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Refrigerant-based propulsion system for small spacecraft

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REFRIGERANT-BASED PROPULSION SYSTEM

FOR SMALL SPACECRAFT

by

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ABSTRACT

The MR SAT spacecraft under development at UMR requires a propulsion system that can be utilized to perform orbital maneuvers and three-axis attitude control to complete its mission objective of conducting spacecraft formation flight. This thesis documents the research, analysis design and development of the cold gas propulsion system that was integrated on the MR SAT spacecraft. The basis of design and safety requirements stemmed from the AFRL University Nanosat Program competition, in which the UMR SAT project placed third out of eleven schools from across the nation. The MR SAT propulsion system was a primary feature as it implements a refrigerant (R-134a) propellant that has never been flown in space. As detailed in this thesis, through engineering modeling and laboratory testing R-134a is demonstrated to be a feasible propellant for small spacecraft. As the R-134a is stored as a saturated liquid in the tank, it was necessary to analyze the thermodynamic properties of the refrigerant and investigate phase changes for its use as a propellant. Also documented is the hardware selected and the integration into the MR SAT spacecraft, along with the laboratory testing that has been conducted. R-134a offers good performance characteristics and this thesis can be used as a design template by other small spacecraft developers who require a safe and inexpensive propulsion system.

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1. INTRODUCTION

1.1. BACKGROUND

Space is a frontier that we must endeavor to explore and learn to conquer for the future of mankind on Earth. The recent challenges humans have overcome and the knowledge gained is unprecedented. The resulting advances in all facets of science and the technology we embrace today are invaluable.

With current access to space so heavily limited by financial burdens, the development of smaller, and more cost effective satellites is a rising trend. The reduced development, launch and insurance costs are far superior when considering that a small satellite still provides a technologically proficient test platform with payloads and missions offering ground breaking research and discoveries. The recent increase in demand and the technological advances related to small spacecraft have driven the need to develop small subsystem components. In particular, the implementation of technically challenging objectives, such as formation flight missions for small spacecraft, has given rise to the need for small, safe and efficient propulsion systems capable of performing orbit and attitude control.

The small spacecraft sector also broadens the range of developers from the traditionally dominating government and large business industries to the small-medium sized businesses and universities. The development of small spacecraft by university-based programs is emerging as a valuable and growing sector of the global spacecraft community. These projects push the boundaries and expand the range of spacecraft advancements, technology and abilities while providing an effective learning platform for the engineers and scientists of tomorrow.

1.2. SMALL SATELLITE CLASSIFICATION

Satellites are generally classified by their mass at launch and divided into broad classes. These classes along with common names and mass ranges are highlighted in Table 1.1.

While these classes of satellites can be vague and the naming convention can be interpreted in a multitude of ways, it is generally considered that small satellites are those below a mini class satellite, i.e. mass < 500 kg. In addition, small satellites are generally perceived as smaller projects with respect to cost and timelines in comparison to traditionally large satellite developments.

Table 1.1 Satellite Classification Sizing [1]

Satellite Class	**Mass Range**
Large	> 1000 kg
Medium	500-1000 kg
Mini	100-500 kg
Micro	10-100 kg
Nano	1-10 kg
Pico	< 1 kg

The term “small satellite” in the context of this paper will be applied to the micro and nano class satellites falling into the mass range 10-100 kg. There is breadth to develop the propulsion system discussed in this thesis beyond this range and be integrated into both smaller and larger systems. Similarly, the term “spacecraft” will be used in place of “satellite” as there is potential scope to extend the use of this propulsion system beyond the limits of Earth orbit.

1.3. UNIVERSITY NANOSAT PROGRAM

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), NASA Goddard Space Flight Center, and the American Institute of Aeronautics and

Astronautics (AIAA), has developed the University Nanosat Program (UNP). This program is intended to promote satellite development, education and knowledge for students in university-based satellite teams. There is a strong emphasis on research and development of small satellites through a practical application of fabrication, integration and testing [2].

The UNP is a two-year cyclic program that involves universities across the United States. The program is based upon a competition format with AFRL and associated personnel reviewing the developments of the teams and satellites over the course of the two years. At the end of the two-year term the teams are required to present an Engineering Design Unit (EDU) and the competition winner will have AFRL UNP present the project to the Department of Defense (DoD) Space Experiments Review Board (SERB) with the intent of securing a launch opportunity through the DoD Space Test Program (STP) [3] [4].

1.4. UNP DESIGN CONSTRAINTS

The design of a propulsion system for a spacecraft in the UNP competition must meet stringent requirements and will undergo a rigorous safety assessment by the AFRL reviewers. Due to the associated safety concerns of a propulsion system and the constraints encountered in the development of a spacecraft by a low budget university group, all designs must adhere to strict guidelines presented in the UNP User's Guide (UG) [3]. The UNP UG is a limited release document which imposes constraints that are based upon the safety required to fly a payload on the Space Shuttle. Although private payloads are no longer flown on the Space Shuttle, the program requires very stringent safety standards for payloads as it is a manned spacecraft. These standards are good guidelines for university-based projects to follow, to ensure their spacecraft will pass scrutiny by launch vehicle providers.

The requirements of the UNP UG that apply directly to a propulsion system primarily stem from the guidelines of the NASA standard 5003 - *Fracture Control Requirements for Payloads Using the Space Shuttle* [5]. NASA standard 5003 classifies a pressurized system as either a "sealed container" or a "pressure vessel" based on the conditions of the fluid being stored. As per the UNP UG requirements pressure vessels are prohibited from use in the competition. Consequently, any pressurized system used must maintain a sealed container classification and maintain a non-hazardous internal environment. In order to be classified as a sealed container, the physical limits given in Table 1.2 may not be exceeded during launch and operation of the spacecraft. These limits are defined in NASA-STD-5003.

Table 1.2 Physical Limits of a Sealed Container [5]

Stored Propellant Property	**Limit**
P - Absolute Pressure	≤ 689.48 kPa (100 psia)
U - Internal Energy	≤ 19,319 kJ (14,240 ft-lbs)

Along with the limitations of a sealed container classification there are additional design guidelines in the UG that are deemed discouraged or prohibited practices. Listed here are the UG-based practices which directly affect the design of a propulsion system:

- It is prohibited to use pyrotechnic devices and/or mechanisms.
- It is prohibited to use toxic and/or volatile fluids or gasses. It is discouraged to use materials that can undergo a phase change during launch or on-orbit.
- It is prohibited to use cast metallic or welded joints.
- It is prohibited to use parts or assemblies for which safety is highly dependent upon the build or assembly process. Examples include composite materials and certain deployment mechanisms. If it is necessary, these processes should be completed or witnessed by aerospace professionals.

While it is a requirement of the competition to adhere to NASA-STD-5003 and UNP UG design constraints, the generic propulsion system developed in this thesis does not need to directly meet these limitations. Regardless, the UNP UG design constraints will be implemented as they provide a benchmark of safety and consistency. The case study presented of the application of this propulsion system in the EDU also meets the UNP UG guidelines.

In order to design and analyze the propulsion system, it is necessary to define a temperature operating envelope. The temperature range of -50 ºC 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 in virtually any low Earth orbit (LEO) and is even beyond the hardware specifications of many onboard systems studied.

1.5. UNIVERSITY OF MISSOURI-ROLLA SATELLITE PROGRAM

The University of Missouri – Rolla (UMR) Satellite program (UMR SAT) is a student design team developing a small satellite with the assistance of faculty and industry mentors. UMR was one of eleven universities invited to participate in the UNP Nanosat – 4 (NS4) competition which concluded in March 2007. UMR finished 3rd in the competition and also received the award for most improved.

UMR SAT is developing a satellite pair to advance studies and knowledge of Distributed Space Systems (DSS) missions. The use of small satellites flying in formation is a relatively recent innovation with many advantages. Utilizing smaller spacecraft in formation can match or outperform the mission objectives of one larger spacecraft often with reduced cost, complexity and risk of mission failure. Formation flight of small spacecraft is a growing area of interest for the U.S. Air Force and for industry partners alike. The Missouri Rolla Satellite (MR SAT) and Secondary Satellite (MRS SAT) are being designed, constructed, integrated and tested to study autonomous formation flight. Figure 1.1 shows a computer generated model of the satellite pair.

With a mission objective to study close formation flight, the UMR SAT requires a propulsion system capable of providing primarily small orbital maneuvers with the capability to also perform launch vehicle ejection tumble (tip off) control and fine tune three-axis attitude corrections. Along with meeting mission requirements, performing efficiently and being financially feasible, the propulsion system must fulfill system and safety requirements as introduced in Section 1.4.

Figure 1.1 MR and MRS SAT On-Orbit After Separation

Both spacecraft will be equipped with attitude determination hardware as well as magnetic coils as a primary attitude control device. The propulsion system can be used as a secondary device to control attitude. Only the larger spacecraft, MR SAT, will be integrated with a propulsion system. During formation flight it is necessary for MR SAT to "follow" the orbit of MRS SAT using the propulsion system to maintain a separated distance of 50 m with a tolerance of ± 5 m.

1.5.1. Mission Objectives. The objectives and modes of operation of the MR SAT mission are focused on the study of close formation flight. Much is to be gained from the study of the orbit and spacecraft dynamics of a close formation satellite system, as well as implementing technologically advanced algorithms for orbit determination and control. The MR SAT test platform requires the development of an inter-satellite wireless communication link and an efficient and safe propulsion system. Due to the limited budget of the UMR SAT program all hardware procurement must be innovative and low-cost [6]. This budget constraint results in "off the shelf" and non-space rated products being used in the design. MR SAT mission success potentially demonstrates the suitability of these unrated hardware components for spaceflight.

Commercially available propulsion systems that could be integrated on MR SAT were researched, however, their price was well beyond the budget of the university developed spacecraft. There is also significant knowledge to be gained by the numerous team members working on in-house design, manufacture and testing.

To ensure success, the MR SAT objectives and goals are to be achieved over the course of the mission as defined by the Modes of Operation. The following Modes of Operation are a program top-level mission sequence, with attention drawn to the modes that require use of the propulsion system.

- Launch Mode: Launch mode covers the mission from spacecraft integration through to launch vehicle separation. During this time it is required that the propulsion system remains inactive and that the propellant be securely and safely stored.
- De-tumble Mode: Once the spacecraft has begun initialization and power up and is in a secure state, the systems can commence functionality to remove tip off slew rates that occurred during launch vehicle separation and restore the spacecraft to an attitude stable state. Attitude control devices will be the primary system to remove slew rates but if stability cannot be restored, the propulsion system can be utilized to perform de-tumble maneuvers. It should be noted that during this mode the two spacecraft are connected as one vehicle, consequently, attitude control must account for the combined mass.

The launch vehicle separation maximum "tip off" rates are expected to be 1 deg/sec [3].

- Separation Mode: With the combined spacecraft in a desired and stable attitude configuration and with all onboard systems ready, the spacecraft can perform their separation sequence. During separation it is undesirable but possible for MR SAT to perform propulsive maneuvers, providing safety of both spacecraft is uncompromised.
- Formation Flight Mode: The formation flight sequence will commence as soon as the spacecraft are separated by a defined clearance, at which point the propulsion system will be utilized to maintain the system formation distance of 50 m with a tolerance of ± 5 m. The time duration of formation flight is dependant upon the propellant consumption, with a goal of at least one orbit. With the depletion of propellant, the mode will conclude, leaving the spacecraft to drift in orbit and other mission objectives, such as wireless communication range testing between the two spacecraft, can begin.

1.5.2. Propulsion Performance Requirements. The design of a propulsion system must be based upon the required on-orbit performance. This includes the required total change in velocity (ΔV), which is required for orbital maneuvers during the formation flight phase. This system must also have sufficient propellant to perform any additional de-tumble and attitude control pulses that will be used to arrange the spacecraft in preparation for formation flight.

There is no specific minimum ΔV requirement that has been set for the MR SAT propulsion system, however, there is a mission objective to perform one orbit of formation flight. This requirement will be justified initially with attitude and orbit simulations and then finally with on-orbit performance. Due to the short life expectancy of MR SAT the consideration of ΔV requirements for correcting orbit perturbations has been accounted for in the orbit simulation that justifies the MR SAT propulsion system performance.

1.5.3. Physical Properties. The spacecraft must also meet mass, dimensions and physical constraints of the UNP UG to qualify for the AFRL competition. Only the physical properties which have an influence on the design of the propulsion system have been highlighted in this section. The satellite has a cylinder static envelope constraint which has a diameter of 474.98 mm (18.7 in) and a height of 474.98 mm (18.7 in) as seen in Figure 1.2. Initial designs of the docked satellite pair fit within the static envelope. As the design progressed it was necessary to void this envelope with the Ground Support Equipment (GSE) tabs. It was also an integration requirement that the propulsion system fit within this envelope limit, however the tank fill and drain valve goes beyond the envelope limit. Also shown is the axis orientation for the spacecraft.

Figure 1.2 Spacecraft Dimensional Envelope

The spacecraft is connected to the launch vehicle through a motorized lightband, low-shock, non-pyrotechnic separation system which is manufactured by Planetary System Corporation. The separation plate is circular and is mated to the circular bottom plate of MR SAT.

The mass requirement of the combined satellite pair is 30 kg (66.14 lbs) with the center of gravity (cg) to be located within 6.35 mm (0.25 in) of the cylinder centerline and within 304.8 mm (12 in) of the spacecraft's bottom plate. This cg requirement only applies when the spacecraft are docked. After separation the cg location will move predominately along the cylinder centerline (Z - axis) with the absence of MRS SAT.

MR SAT individually has a mass of 19.41 kg (42.79 lbs) while MRS SAT has a mass of 9.72 kg (21.43 lbs). The spacecraft pair have a combined mass of 29.13 kg (64.22 lbs) which is below the required 30 kg. The mass data of both spacecraft as well as mass moment of inertia data is presented in Table 1.3.

While there are no direct restrictions in place for the mass, volume and power consumption for the propulsion system, it is imperative that these aspects be considered for all phases of design, and the appropriate subsystems are consulted with all design propositions and hardware acquisitions.

Table 1.3 Spacecraft Mass and Moment of Inertia Information

Spacecraft	**Mass (kg)**	**I_{XX} (kg.mm^3)**	**I_{YY} (kg.mm^3)**	**I_{ZZ} (kg.mm^3)**
MR SAT	19.41	504368733.7	478569729.5	503029730.0
MRS SAT	9.72	145872516.8	198079546.6	132012321.6
Docked pair	29.13	1194307537.2	1148917633.3	701358259.7

1.6. PURPOSE

This thesis presents the design, ground test and performance of a low-pressure cold gas refrigerant propulsion system for use on a micro to nano class spacecraft (10-100 kg). The advantages of a refrigerant-based propulsion system include spatial volume savings and good performance characteristics, which are demonstrated through analysis and preliminary test results. The inherent safety, ease of use and availability of the refrigerant, R-134a, makes it an ideal propellant for university-class satellite projects and future small spacecraft. The R-134a system provides the ability to perform both minor orbital maneuvers as well as three–axis attitude control. The system is intended for use on most small spacecraft, with MR SAT being presented as a case study example in this thesis.

Safety was a major criterion for the design of this propulsion system, and the measures taken to ensure the safety of personnel, launch vehicle and spacecraft have been addressed in this thesis. A Safety Assessment White Paper (SAWP) which addresses the proposed propulsion design is being lead by UMR, and is directed by the author of this thesis with participation from students of Washington University – St Louis, University of Texas at Austin, and guidance from AFRL personnel [7].

R-134a is a safe, non-toxic, non-flammable compound that is well suited for propulsion system development at the university level, provided the necessary laboratory precautions and environmental considerations being presented here are followed.

1.7. THESIS ORGANIZATION

The introductory section of the thesis is followed by nine sections. A brief description of their content is given below:

2. LITERATURE REVIEW - A review of propulsion systems for small spacecraft is undertaken with references and current literature cited. A description of current and former propulsion technologies with specific emphasis on cold gas and saturated liquid systems, and an examination of the use of refrigerants in space is also provided.

3. PROPELLANT SELECTION – An explanation of the methodology and design procedure that was used to define the use of a cold gas propulsion system for MR SAT, and the selection choice of a refrigerant propellant with xenon gas as a backup is given.
4. NOZZLE DESIGN AND PERFORMANCE – An outline of the parameters that determine the design of a nozzle, based on the performance required and obtainable is given. The methodology and computational analysis used to design the MR SAT nozzles and their performance characteristics is also presented. A refined engineering model is also developed and the results discussed.
5. HARDWARE REQUIREMENTS AND SELECTION – A summary of the criteria utilized for hardware selection is given. Included is a description of the hardware selected for the MR SAT propulsion system and the issues involved in this design.
6. SPACECRAFT INTEGRATION – A discussion of the thruster configurations that can be used on spacecraft and the orientation selected for MR SAT is provided. A summary is also included of the concerns and procedures that are involved with propulsion system integration. Using the MR SAT propulsion system as an example, discussion regarding placement and integration for hardware components including thrusters, tubing and tank is given.
7. REFRIGERANT COMPATIBILITY AND MATERIAL SELECTION - Details of the compatibility and outgassing issues that are involved when using refrigerants in space environments, with particular emphasis on seals and sealing agents, are discussed.
8. SYSTEM LOSSES, TESTING AND ANALYSIS – The tested thermodynamic and performance characteristics of the R-134a propulsion system are presented. A particular focus is given to quantify tube flow pressure losses and the computation of the R-134a friction factor to allow complete operating envelope analysis. These results were obtained both through laboratory hardware tests and computed analysis.

9. SAFETY CONSIDERATIONS – The safety issues affecting the design of a propulsion system, which is the primary driver for the AFRL UNP, are discussed. This section includes details of the SAWP that is being undertaken, in collaboration with two other universities in the NS4 competition, for the AFRL. Details of the environmental and legal constraints of using a refrigerant are also given.
10. CONCLUSION – The thesis concludes with a discussion of the propulsion system design integration on MR SAT and its fulfillment of requirements and objectives. A discussion of possible future work and research extending beyond the scope of this thesis is provided.

2. LITERATURE REVIEW

2.1. SMALL SPACECRAFT PROPULSION FLIGHT HERITAGE

The scratchy "Beep-Beep-Beep" was the transmission the world heard when the first satellite orbited Earth on 4th October 1957. The Russian built and launched Sputnik 1 was a 58 cm (23 in) diameter ball weighing in at a mass of 83.6 kg (184 lb), making it the world's first small satellite [8]. Thousands of satellites, both larger and smaller have flown since the groundbreaking milestone of Sputnik 1.

It was not until 1991 that the first small satellite with a propulsion system was flown in orbit. The Defense Advanced Research Projects Agency (DARPA) sponsored the launch of a constellation of seven satellites each weighing 22.7 kg (50 lb) to study DSS communications relay architecture [9]. The spacecraft series were known as MicroSat 1-7 and were launched by a Pegasus launch vehicle on 16th July 1991. They unfortunately did not obtain their desired orbit and faced a reduced life span. MicroSat utilized a cold gas thruster storing gaseous nitrogen (N_2) at a pressure of 41.37 MPa (6000 psi) for orbit and formation station keeping. The expected propellant life was four years, but was never completely utilized as the orbits decayed in January 1992 with all seven spacecraft still operational [10].

2.2. SMALL SPACECRAFT PROPULSION OPTIONS

There are three primary propulsion system types that are currently used on spacecraft; cold gas, chemical and electric. Each propulsion system option offers different levels of performance and advantages and disadvantages. The selection of a propulsion system is dependant on many factors including the requirements of the mission and limitations of the spacecraft design. There are other more advanced and exotic forms of propulsion systems which can be implemented on spacecraft. These however were not feasible in terms of technological development, financial viability and safety and, therefore, were not included in this research.

Specific impulse (I_{SP}) is a measure of a propulsion system's efficiency, measured in units of seconds. I_{SP} is the ratio of the thrust that is produced to the weight flow rate of propellant. It is a convenient tool for comparing propulsion systems as the thruster size and application are virtually irrelevant when analyzing pure performance in the vacuum of space. The various propulsion technologies and propellants offer different characteristics and the typical I_{SP} ranges are shown in Table 2.1. These various types of propulsion systems, most of which can be utilized on a small spacecraft, are briefly discussed with reasoning given for their exclusion as a consideration as a MR SAT propulsion system option.

Table 2.1 Propulsion Technology Typical I_{SP} Ranges [11], [12]

Propulsion Technology	**Typical I_{SP} (seconds)**
Cold Gas	30 – 70
Liquid (bipropellant)	305 – 460
Liquid (monopropellant)	140 – 240
Solid	260 – 300
Hybrid	250 – 350
Electric	300 – 10,000
Nuclear	800 – 6,000

2.2.1. Cold Gas Propulsion Systems. Cold gas propulsion systems are the simplest and safest propulsion method currently in use. As the name suggests, a gas is stored under pressure in a tank and then released as a cold propellant through a nozzle. It is the pressure of the gas that drives the propellant through the nozzle with the thrust developed from the momentum exchange of propellant exhaust. Although this is a low thrust and low efficiency propulsion means, a reliable system can be developed for low cost. Typical gases used are N_2, helium (He), ammonia (NH_3) and xenon (Xe).

Cold gas propellants can either be stored as a high pressure gas or as a two-phase saturated liquid. A major disadvantage of using cold gas propulsion systems is the high pressure storage, up to 60 MPa (8702 psi), and large volume tanks required to obtain reasonable performance characteristics [12]. One method used to overcome this drawback is to store the propellant in a two phase, liquid-vapor state, where the storage pressure is the propellant saturated vapor pressure, which is often significantly lower.

A successful low pressure cold gas system, demonstrated in the confines of the International Space Station, was the SPHERES spacecraft developed by the Massachusetts Institute of Technology, NASA and DARPA. The spacecraft weighed 3.1 kg, stored 74 grams of carbon dioxide (CO_2) liquid at 5.93 MPa (860 psi) [13]. Gas was exhausted at a regulated pressure of 137-483 kPa (20-70 psi), through 12 micro-solenoid valves and 12 nozzles, generating up to 0.25 N of thrust. Depending on usage the tank had propellant for 20 seconds to 30 minutes of activity and could be refilled upon depletion [14].

The European Space Agency (ESA) designed and developed the 720 kg Cryosat with a mission objective of studying the elevation and thickness of polar ice and sea ice from polar orbit [15]. Unfortunately the Cryosat spacecraft and mission was lost after a launch vehicle suffered an anomaly in the second stage causing separation failure. A second spacecraft Cryosat-2 is being developed for a predicted March 2009 launch with the same mission objectives [16]. The Cryosat spacecraft implemented a cold gas propulsion system with many commercial off the shelf (COTS) components. A single high pressure tank stored 36.2 kg of gaseous N_2 at 27.86 MPa (4040 psi). Attitude control was to be performed with sixteen 10 mN thrusters with four additional 40 mN thrusters used for orbital control. Both thruster sets were nominally supplied with a regulated propellant absolute pressure of 0.13 MPa (18.85 psia) and a maximum flow rate of 0.25 g/s [17].

The earliest forms of on-orbit satellite propulsion systems were cold gas systems utilizing inert gases [18]. As spacecraft developed over the years, their mission complexities and duration grew and the need for more advanced and capable propulsion systems evolved.

2.2.2. Chemical Propulsion Systems. Chemical systems are the workhorse of space industry and have widespread use in all applications from launch vehicles to satellites to manned spacecraft. There are three primary types of chemical systems which are defined by the state of the propellant; liquid, solid and hybrid and are discussed in more detail below.

Liquid propellants are the most common and universal spacecraft propulsion system. Bipropellant systems use a fuel and an oxidizer, which are stored as liquids and combined under pressure to chemically react in a combustion chamber. These systems offer high performance, however, they utilize flammable and often toxic propellants and require large and complex hardware components and therefore have not been considered for development in this study.

Monopropellant systems use a single liquid propellant that reacts with a catalyst to decompose into hot gases that are exhausted producing thrust. These systems are frequently used in small spacecraft for attitude control and orbit maneuvers. They offer high reliability and good performance, however, the hardware development and cost is outside the realm of the intent of this thesis. Additionally, the primary monopropellant hydrazine (N_2H_4) is highly toxic, flammable and dangerously unstable. Other monopropellants such as hydroxylammonium nitrate (HAN) based propellants are safe, however, they lack the research and flight heritage of hydrazine.

While not a small satellite, with a mass of 950 kg (2100 lb), the Landsat series of Earth observing satellites in the 1970s used monopropellant systems with 30.4 kg of hydrazine that decomposed at 1000 °C [18]. A more recent spacecraft to utilize a monopropellant hydrazine system is the planned NASA Lunar Reconnaissance Orbiter (LRO), which is currently under construction and has a scheduled launch of October 2008 [19]. The spacecraft has a dry mass of 1046 kg and will be propelled with almost 900 kg of hydrazine that will be stored as a liquid with a Maximum Expected Operating Pressure (MEOP) of 2.41 MPa (350 psi), but will be pressurized with helium gas that can be pressurized as high as 28.96 MPa (4200 psi) [20].

Solid propellants have not been considered as they utilize combustion to burn their solid material producing a pressurized gas exhaust. Solid propellant motors have a one-time use making them unsuitable for the mission requirements of MR SAT.

Hybrid rocket engines store propellants in different phases, often a liquid fuel and liquid or gas oxidizer. Hybrids have many attractive features such as safety and storability however they still utilize a combustion process and do not have technology readily available for small satellite integration.

2.2.3. Electric Propulsion Systems. The principle of an electric propulsion system is to use an electrical power source to accelerate a propellant in a focused direction generating thrust. Electric propulsion systems, including; Ion engines, Hall Effect thrusters, Field Emission Electric Propulsion (FEEP) and Colloid thrusters, were not considered due to the primitive flight heritage of the technology and the development required for implementation into a university-level spacecraft.

A Pulsed Plasma Thruster (PPT) utilizes an electric charge to ablate and ionize a solid propellant, Teflon. The plasma produced is accelerated electromagnetically producing thrust. PPT's offer simplistic design and long life, however, they only produce small levels of thrust 2 μN – 4.5 mN and can only be pulsed, with the electric discharge [21]. This does not meet the requirements of any prolonged orbital maneuvers requiring continuous thrust.

Resistojets employ an electric heater to vaporize a liquid, or heat a gas, to a higher energy state where it is exhausted through a nozzle to develop thrust. An Arcjet uses the same electrothermal principle however it uses an electric arc to generate and transfer heat to the propellant. Resistojets can operate with a variety of different propellants and have demonstrated proven performance in space on numerous spacecraft. The disadvantages and reasons for exclusion from use on MR SAT are the traditional high power requirements (~100 W), the limited and immature development of miniaturized technology suitable for small satellites, and the technical challenges of development at the low budget university level [21].

SSTL is advancing the miniaturization of resistojet technology with the development of a family of low power resistojet thrusters. These thrusters have proven spaceflight on the ALSAT-1 (2002) and DMC (2003-) series of spacecraft. The resistojet on ALSAT-1 used 3.7 kg of butane stored as a liquid at pressures up to 0.4 MPa (58 psi), and used two redundant heaters each rated to 15W [22]. The system has also been tested for use with the inert gases nitrogen and xenon, such as on the Beijing-1

DMC satellite (also known as China DMC+4) that stores xenon gas at 6 MPa (870 psi) for use through the resistojet [23].

2.3. 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. This section highlights the features of a few spacecraft that utilize a saturated-liquid propulsion system.

SNAP-1 is a nanosatellite developed by SSTL and launched in 2000 carrying a small scale propulsion system that stored liquid butane. The spacecraft had a wet mass of 6.5 kg and stored 32.6 grams of butane at nominal operating conditions of 20 ºC and 0.21 MPa (30.5 psia) absolute, with a MEOP of 0.4 MPa (58 psia) [24]. Thrust was produced by vaporizing the liquid with a 15 Ω resistive heating element which was then exhausted out a valve and nozzle assembly. The propellant was stored in a unique storage device that provided a total volume of 65 cm^3. This was accomplished by utilizing a 1.1 m length of aircraft grade aluminum tubing wound into a triangular coil which was then directly attached to the valve [25].

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 I_{SP} was measured to be 43 s which also suffered in comparison to the theoretical value of 70 s. Additional on-orbit data indicates that the propulsion system provided between 1.9-2.0 m/s in total 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 a total firing duration of 297.1 s [25].

Another satellite developed by SSTL and launched in 1999, was the 325 kg UoSat-12 [26]. This spacecraft utilized two propulsion systems, a standard cold gas system using N_2 and a revolutionary resistojet utilizing the storage of liquid nitrous oxide (N_2O). The N_2O was stored at a vapor pressure of 5.1 MPa (739 psi), and if used as a cold gas, would have an I_{SP} of 66 s, however with the use of the 100 W resitojet the I_{SP} was raised to 127 s and produced a thrust of 125 mN [27]. The N_2O resistojet was flown as a technology demonstrator for orbital maneuvers and produced a total of 10.4 m/s ΔV.

It was also found that the exothermic nature of decomposing N_2O allowed the resistojet to maintain performance at reduced power levels [28].

The N_2 cold gas system on UoSat-12 produced 16.4 m/s of ΔV that was used for both attitude control and orbital maneuvers. The N_2 was stored in three tanks with a total volume of 27 liters [26]. The pressure of the tanks was 20 MPa (290 psi) and was regulated down to a nominal 0.4 MPa (58 psi) for nozzle expulsion [28].

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- DSS spacecraft. Scheduled for flight in June 2007, the 3.5 kg spacecraft will implement a cold gas system storing Sulfur Hexaflouride (SF_6) as a liquid [29]. At 21 ºC, the vapor pressure of SF_6 is 2.17 MPa (315 psi) and MEOP is 3.45 MPa (500 psi). With a 10 ml storage tank, the target performance goals for the system are 50 mN of thrust, with an I_{SP} of 45 s and a total ΔV of 2 m/s [30].

2.4. REFRIGERANTS IN SPACE

Refrigerants in space are currently used primarily in conventional applications such as temperature control fluids in heat management systems. The only means of hardware heat removal in space is through radiation. The use of heat sinks is restricted by size and mass constraints so the use of heat pumps can improve heat removal performance and alleviate these limitations.

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 system. To remove heat from the Freon-21 coolant loop, water boiling, ammonia boiling and heat sink radiators are used. The payload bay doors, that are opened in orbit, house the radiators that provide a surface area of 111 m^2 (1,195 ft^2) and over 1.6 km (1 mile) of Freon tubing [31].

For future spacecraft implementations, a rolling-piston compressor utilizing flow through lubrication has been designed. Without the use of a sump the compressor is able to be used in a zero-gravity environment by allowing the refrigerant and lubricating oil to mix. The preliminary design utilizes two stages to circulate R-134a across a pressure ratio of 69-690 kPa (10-100 psi) and temperature gradient of 55 ºC [32].

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 [33]. Pioneer 10 transmitted signals to Earth for over 30 years including the first close-up images of Jupiter as well as numerous environmental measurements, and was the first spacecraft to travel through the asteroid belt. 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 [34].

2.5. NICHE FOR A NEW PROPULSION SYSTEM

As documented from this review of literature, there is a niche for a new propulsion system that meets the needs of the university-based satellite developers. This thesis describes the procedures in designing and developing a propulsion system that meets the requirements of:

- Low budget, utilizing commercial off the shelf components
- Low storage/operating pressures
- Minimal volume/size envelope
- Proven and easy to implement technology without significant prior research

A cold gas system is a propulsion solution that meets all these requirements and can be implemented by a university-based satellite development team. A cold gas propulsion system is a simple yet highly proven technology that is a safe and manageable propulsion system for a small spacecraft. While system requirements must be fulfilled, the propellant selection criteria must also be met, as detailed in Section 3. MR SAT implements a cold gas propulsion system that utilizes the storage of propellant in a two-phase, saturated-liquid state, with the details of the design and integration described in detail.

3. PROPELLANT SELECTION

3.1. PROPELLANT SELECTION CRITERIA

With the selection of an appropriate propellant, a cold gas propulsion system is a safe and feasible option that allows design, fabrication and testing to be performed by a university-based spacecraft program. For this study, the selection of a propellant compound that met the following criteria was implemented:

- Non-toxic / 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

There are a number of compounds that are available for selection as a propellant that meet these requirements. Those considered in this study include the more traditional noble gases helium (He), neon (Ne), argon (Ar) and xenon (Xe), as well as the mostly inert diatomic nitrogen (N_2) and the stable compound, carbon dioxide (CO_2). Other compounds considered were the refrigerants 1,1,1,2-Tetrafluoroethane and 2,2-Dichloro-1,1,1-trifluoroethane or more commonly R-134a and R-123, respectively. R-123 was later discarded as the Environmental Protection Agency (EPA) Clean Air Act (CAA) defines it as a Class II substance, whose production and sales require appropriate certification and will be illegal after 2015 [35]. More detailed information on EPA regulations and refrigerant use is given in Section 9.

There are a number of other compounds that were initially considered and may have displayed prominent thermodynamic and performance qualities, but were disregarded for other reasons. As an example, hydrocarbons such as butane were not considered because of their flammability. The inorganic compound sulfur hexafluoride

(SF_6) is inert and non-toxic, however, it was not considered in this study as it is the most potent greenhouse gas with a global warming potential that is 23900 times greater than CO_2 as per the U.S. EPA classifications [36].

3.2. PROPELLANT COMPARISON

During the early development of this research, a former UMR SAT team member, Michael Christie, performed an analysis to compare the feasibility of the above propellant compounds. This analysis used the sealed container restrictions of maximum absolute pressure 689.48 kPa (100 psia) and maximum internal energy 19, 319 kJ (14, 240 ft-lbs) to determine performance parameters ΔV and I_{SP} over a range of suitable tank volumes.

The primary requirement of the MR SAT propulsion system is to perform controlled orbital maneuvers during the formation flight mission phase. Consequently, the primary driver for propellant selection was to maximize the obtainable ΔV for a given tank volume while maintaining a sealed container status. It should be noted that this comparison method did not aim to identify the most efficient propellant option (maximum I_{SP}) as shown in the following results. Instead, the propellant selected is the most advantageous in ΔV and meets the needs of the designed propulsion system for a spacecraft of this size.

A conservative approximation for the maximum temperature expected on-orbit was set at 100 °C (212 °F). Using this maximum temperature, the maximum propellant density (kg/m^3) can be determined corresponding to the limit of either maximum pressure or internal energy, whichever occurs first. With the mass of propellant determined, the ΔV was calculated using isentropic nozzle flow relations and the Rocket Equation using a spacecraft mass of 25 kg.

There are many assumptions used in this analysis including the approximation of the propellant being exhausted as a calorically perfect ideal gas at 20 °C (68 °F). At these operating conditions, R-134a is in a pure gas state and the assumption of an ideal gas was valid and allowed comparison to the other gaseous propellants. Isothermal conditions were also utilized, assuming that the stored propellant, tank and hardware maintained a fixed temperature. This is a valid assumption in this analysis given that it is a tool to compare the relative performance of each propellant, not for an absolute analysis of the individual system.

In order to perform an analysis of a two-phase refrigerant, it is necessary to be familiar with the thermodynamic properties. One such thermodynamic property is internal energy (*U*) that quantifies the energy of the molecules in a physical system. It is a combination of the kinetic energy of the particles as well as the potential energy of the attractive forces between molecules. Internal energy can be calculated for a substance using the thermodynamic properties tables. With a single phase substance, the internal energy can be found with two known parameters, such as temperature and pressure.

With a two-phase substance such as R-134a, it is also necessary to define an additional thermodynamic property, the quality (*x*), to define internal energy. Quality is the proportion of vapor mass to total vapor and liquid mass of a two-phase system. A 100% quality indicates that the substance is in a complete superheated vapor state. A quality of 0% indicates that the substance is in a complete liquid state. Quality is a property that can also be calculated for a substance from the thermodynamic tables (and *P-h* diagram) with two known parameters. The results of the analysis for the given propellants are displayed in Figure 3.1.

Figure 3.1 Propellant ΔV Comparison [37]

The maximum ΔV was calculated over a range of tank volumes. As shown, for tank volumes less than 3500 cm^3, R-134a provides the greatest ΔV under the assumed conditions. The sloped section indicates the region where the sealed container pressure constraint defines the maximum storable propellant mass. The flat plateau of the R-134a graph is where the internal energy limit is encountered.

For a given tank volume of 2500 cm^3 (152.6 in^3) the maximum ΔV achievable, while still maintaining a sealed container status is 1.11 m/s with the propellant R-134a. This is followed by xenon at 0.87 m/s and the remaining gases as displayed in Table 3.1. Also tabulated is the calculated I_{SP} for each propellant under the equivalent analysis conditions.

The primary reason for the larger ΔV of R-134a, as opposed to the other gaseous propellants, is due to its high molecular mass and density and consequently the larger momentum transfer during propulsive pulses. It is also desirable to have a controllable and predictable propellant with a low mass flow rate yet high exhaust velocity, which is facilitated with R-134a and its low specific heat ratio (γ). Another point to consider is that the performance comparison made is not affected by the supply pressure of the cold gas, however, it is proportional to temperature. The analysis was performed at 20 °C (68 °F) and would experience performance improvements if this operating temperature were increased.

Table 3.1 Comparison of Propellant Performances [37]

Propellant	**I_{SP} (seconds)**	**ΔV (m/s)**
R-134a	49.9	1.11
Xe	30.8	0.87
CO_2	66.3	0.64
Ar	55.9	0.49
N_2	76.6	0.47
Ne	79.2	0.35
He	176.9	0.15

3.3. REFRIGERANT TWO-PHASE DIMENSIONAL COMPARISON

It is important to exploit thermodynamic properties as well as meet performance and design constraints when selecting a propellant. 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. Figures were used to show the thermodynamic properties of pressure, temperature, and state for four example cases of refrigerant propellant masses in a 2.5 L (2500 cm^3) 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 3.2 shows the dimensional example of the thermodynamic properties and operational conditions profile of R-134a. Figure 3.3 shows the application example of R-123 used for comparison to R-134a.

Four propellant masses (62.5 g, 125 g, 187.5 g, and 250 g) representing a suitable range of realistic density implementations, were chosen for comparison. The expected temperature range used was -20 °C to 100 °C (-4 °F to 212 °F). A 2.5 liter (152.56 in^3) 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. As an example, if a five liter tank is utilized in a system with 500 g of propellant it will exhibit the equivalent thermodynamic properties as the 250 g propellant mass shown in the following figures.

The data that were utilized in this analysis were generated using the education version of Engineering Equation Solver (EES), distributed by McGraw-Hill, 2006. EES calculates the thermodynamic properties of R-134a using a real fluid, high-accuracy, equation of state. This equation of state includes all two-phase properties and can be used in the proximity of the critical point[1]. Viscosity is calculated from a relationship that can be used for the gas-phase state across a temperature range of 230 K to 475 K[2].

The data were generated with a program that uses the inputs of temperature range and fixed volume and mass quantities. The EES generated pressure and corresponding

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[1] R. Tillner-Roth and H.D. Baehr, "An International Standard Formulation for the Thermodynamic Properties of 1,1,1,2-Tetrafluoroethane (HFC-134a) for Temperatures from 170 K to 455 K and Pressures up to 70 MPa", J. Phys. Chem, Ref. Data, Vol. 23, No. 5, 1994.

[2] M. Huber, A. Laesecke, and R. A. Perkins at NIST-Boulder, submitted in January 2003 to Industrial Engineering and Chemistry Research.

quality, or state, of the substance data are output in matrix form and post processed in MatLab to generate the plots.

Figure 3.2 Propellant State for R-134a in 2.5 Liter Tank

Figure 3.3 Propellant State for R-123 in 2.5 Liter Tank

As shown in Figure 3.2, R-134a will be present as either a two-phase saturated liquid at low temperatures or a superheated vapor at elevated temperatures. Figure 3.3 shows that R-123 will remain in a saturated liquid state for the given masses (125 g, 187.5 g and 250 g) over the entire temperature range in a 2.5 liter tank. It also should be noted that the pressure range reached by R-123 is significantly lower than that of R-134a over this temperature range and density profiles.

This is a simple demonstration of the expected state and condition of a two-phase refrigerant propellant over the temperature and pressure envelope used. This shows the importance of considering thermodynamic and two-phase state properties when selecting a cold gas propellant. As shown, R-123 offers more advantageous thermodynamic properties, such as lower pressures over the analyzed environmental conditions. Its use a propellant is limited by the EPA purchase and usage legislations as further discussed in Section 9. This dimensional analysis is for a tank of 2.5 L, however, it can be scaled linearly to other tank volumes and propellant masses.

3.4. REFRIGERANT PROPELLANT SELECTION

The analysis to determine a propellant that meets the criteria and performance parameters required of this research suggests R-134a as the primary choice. A major limitation of small spacecraft is spatial volume, which particularly hinders conventional propulsion systems which utilize large volumetric high pressure tanks. The development of a cold gas propulsion system using the common refrigerant R-134a stored as a saturated liquid has many advantages. The primary benefit of a refrigerant as a spacecraft propellant is its ability to be stored as a saturated liquid at a low pressure. Since the liquid phase has a much higher density than its vapor equivalent it allows for substantially more propellant mass being stored than a pure gas propellant at an equivalent volume and pressure. A relatively low saturation temperature allows a proportion of liquid refrigerant to be heated to a vapor state and to be extracted and used like a traditional cold gas propellant system.

As the development and use of a refrigerant propellant is a novel approach to propulsion systems and requires significant design, test and analysis, a back up system using the traditional cold gas xenon is being implemented. The MR SAT propulsion system hardware was designed to be used with both R-134a as well as the replacement xenon. Considering the confined timeline of the UNP competition, if the safety validation of R-134a could not be completed on schedule the xenon could be easily substituted.

4. NOZZLE DESIGN AND PERFORMANCE

4.1. SPECIFICATIONS FOR DESIGN PARAMETERS AND ANALYSIS

With the refrigerant R-134a propellant selected, it is necessary to expand the scope of the analysis to facilitate the design of hardware such as nozzles. It also allows an in-depth understanding of the performance characteristics of the refrigerant. In a cold gas propulsion system the nozzle is the means by which the propellant accelerates and is exhausted, extracting the fluid dynamic properties of the fluid to produce thrust. The nozzle design analysis shown here is specifically tailored to the requirements of the propulsion system integrated on MR SAT. Similar procedures and analysis can be implemented for the application of any cold gas propulsion system for small spacecraft.

With a tank volume of 2.5 L chosen and the temperature range of -50 ºC to 100 ºC defined, only one additional thermodynamic property is necessary to determine all properties of the stored propellant. The pressure of a propulsion system is the most critical in regards to hardware integrity and safety. In a closed volume system such as a propellant tank, the highest pressure of a fluid will occur at the peak temperature. Defining the maximum thermodynamic properties of the system was conducted at this maximum temperature, 100 ºC (212 ºF). The Maximum Design Pressure (MDP) was set at sealed container limitations with an absolute pressure of 689.48 kPa (100 psia). At this maximum temperature and pressure, the internal energy can be calculated along with density and the corresponding maximum storable propellant mass.

At the limits of these sealed container conditions, the maximum mass of R-134a that can be stored in the tank is 60.523 grams. Using this propellant mass, the nozzle analysis was conducted at the designed operating conditions which correspond to a temperature of 20 ºC and an absolute pressure of 137.95 kPa (20 psia) as set by the regulator.

The operating temperature of 20 ºC was selected as it is an approximate mean temperature that can be anticipated by a small spacecraft on a typical LEO. The temperature of the refrigerant will decrease as propellant is exhausted through the thruster. This is a result of the vapor extraction and the endothermic reaction of the liquid vaporizing to restore saturation pressure in the tank. For the purposes of this analysis, it is assumed that the temperature drop will be limited in the tank. This is justified given that the thrust pulses will be for very short time intervals with sufficiently long time duration between pulses to reduce temperature fluctuations. A tank heater will also be used to maintain temperature. If the temperature of the propellant increases above the expected operating temperature, the performance of the thruster will improve.

The use of a pressure regulator was selected for this system for two primary reasons. First, it reduces the maximum pressure the propellant and hardware will experience downstream from the regulator. As an example, propellant enclosed downstream of the regulator at 137.95 kPa (20 psia) and 20 ºC will experience a maximum pressure of only 178.2 kPa (25.85 psia) if the temperature were to rise to 100 ºC. This gives an additional safety margin on the pressure requirements, or lowers strength requirements of hardware components downstream of the regulator. The second advantage is the more consistent thrust levels that can be achieved. When a pressure regulator is utilized, the pressure, and consequently thruster performance, remains constant as the tank pressure fluctuates. The regulator selected and described in Section 5.6, allows the remainder of the propellant to be released when the tank pressure reaches regulated pressure, maximizing propellant usage.

The expected operating pressure of 137.95 kPa (20 psia) was chosen during the propellant selection analysis performed by UMR SAT team member Michael Christie. With an anticipated MDP limit of 689.48 kPa (100 psia) and an expected operating (storage) pressure of only 508.5 kPa (73.75 psia) the regulator pressure has to be relatively low to still be advantageous. However, as regulated pressure is lowered the thrust produced decreases. The choice to regulate to 137.95 kPa (20 psia) was made as a compromise between thrust consistency and steadiness while still maintaining reasonable thrust magnitude. The actual regulator selected for MR SAT has an output pressure, set by the manufacturer, at 170.30 kPa (24.7 psia, 10 psig). This revised pressure setting, however, was not implemented in this early analysis.

In this analysis, it was unrealistically assumed that no pressure losses occurred between the regulator and the nozzle inlet. Consequently, it was assumed that the flow from tank to nozzle is isentropic, in which there is no energy transfer. In reality, this is not the case as later design phase tests confirm there are significant pressure losses in feed lines. However, during this preliminary analysis stage, the isentropic assumption is justified given that the total ΔV, which is the primary performance driver of the system for orbital maneuvers, is independent of inlet pressure. However, a linear inlet pressure decrease affects thrust in a negative fashion, so the duration of thruster pulses must be increased to compensate. For attitude control, this increases the time required for a correction, consequently increasing propellant consumption.

In conducting a performance analysis, the R-134a gas is assumed ideal. In reality, gases can significantly deviate from the behavior of ideal gases around the saturation region and their critical point. This imposes a complication in the analysis of a refrigerant propulsion system with the intention of utilizing the two-phase state for storage and the vapor as a cold gas propellant. In order to quantify the deviation from ideal-gas behavior and compare R-134a to other gases, the compressibility factor, Z, can be calculated. Comparison of gases can be performed when the fluid is normalized with respect to their corresponding critical temperature (T_{CR}) and pressure (P_{CR}). The resulting reduced temperature (T_R) and pressure (P_R) terms, as shown in Equations 4.1 and 4.2, can be used to determine the compressibility factor using the Nelson-Orbit compressibility chart. For all gases, the Z factor is approximately equivalent at the same T_R and P_R conditions [38].

$$T_R = \frac{T}{T_{CR}} \qquad [4.1]$$

$$P_R = \frac{P}{P_{CR}} \qquad [4.2]$$

The compressibility factor of R-134a at the design operating conditions of 20 psia, 20 ºC, T_R = 2.9 and P_R = 0.034 is approximately 1.0. For an ideal gas Z = 1, it is therefore appropriate and valid to assume R-134a will behave like an ideal gas in this analysis.

For the design of the nozzle, it was necessary to take into consideration the structural integration of the thrusters. The outer diameter of the nozzle structure was limited to the maximum diameter of the valve, 6.35 mm. This ensured that the nozzle/valve assembly could be easily integrated with the MR SAT structural side panels. Further information can be found in Section 6.2 on nozzle integration. With the total exterior diameter of the nozzle set at 6 mm, it was then necessary to limit the nozzle exit outer diameter at 5 mm allowing a 0.5 mm wall thickness for structural rigidity. This dimensional limit constrains the exit area (A_e) of the nozzle for flow calculations.

In order to calculate the ΔV in the analysis, it is necessary to specify the mass of the satellite. The mass of MR SAT to be maneuvered by the thruster is estimated to be 25 kg. The specific heat ratio of R-134a was calculated using EES at the operating conditions. The specifications for the design parameters used in the analysis of the nozzle are presented in Table 4.1 below.

Table 4.1 Analysis Parameters used for Nozzle Design

Propellant mass	m_p	60.52 grams
Nozzle inlet temperature	T_c	20 ºC (68 ºF)
Nozzle inlet absolute pressure	P_c	137.9 kPa (20 psia)
Specific heat ratio	γ	1.127
Nozzle exit diameter (maximum)	D_e	5×10^{-3} m
Nozzle exit area (maximum)	A_e	1.9635×10^{-5} m^2
Spacecraft mass (estimate)	m_o	25 kg

4.2. NOZZLE DESIGN PERFORMANCE ANALYSIS

The previous section defined nozzle specifications based on thermodynamic and dimensional constraints. The remaining nozzle parameters to design are the area expansion ratio (AR = A_e / A^*) and consequently the throat area (A^*) and diameter (D_t). The total ΔV is the primary driver for propulsion system requirements and was calculated for a range of suitable nozzle AR. Both ΔV and I_{SP} are functions of inlet temperature, specific heat ratio and the nozzle pressure ratio (PR = P_e / P_c), where P_e is the fluid pressure at nozzle exit. PR is numerically calculated from the AR. The governing equations are based upon the rocket equation and nozzle flow equations as shown in Equations 4.3- 4.10 with further information shown in the Appendix [12], [39], [40].

Sonic velocity:

$$a_0 = \sqrt{\gamma R T_0} \tag{4.3}$$

Characteristic velocity:

$$c^* = \frac{a_0}{\gamma \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma+1}{2\gamma-2}}} \tag{4.4}$$

The PR was numerically calculated using the Newton's method with the function:

$$\frac{A_e}{A^*} = AR = \sqrt{\frac{\left(\frac{\gamma - 1}{2} \right) \left(\frac{2}{\gamma + 1} \right)^{\frac{(\gamma+1)}{(\gamma-1)}}}{PR^{\left(\frac{2}{\gamma}\right)} \left[1 - PR^{\frac{(\gamma-1)}{\gamma}} \right]}} \tag{4.5}$$

and the derivative function:

$$\frac{dAR}{dPR} = \frac{\sqrt{\left(\frac{\gamma-1}{2} \right) \left(\frac{2}{\gamma+1} \right)^{\frac{(\gamma+1)}{(\gamma-1)}}} \left[PR^{\frac{(\gamma-1)}{\gamma}} \left(\frac{2}{\gamma} + \frac{(\gamma-1)}{\gamma} \right) - \frac{2}{\gamma} \right]}{2PR \left(PR^{\frac{(\gamma-1)}{\gamma}} - 1 \right) \sqrt{- PR^{\frac{2}{\gamma}} \left(PR^{\frac{(\gamma-1)}{\gamma}} - 1 \right)}} \tag{4.6}$$

Mass flow rate:

$$\dot{m} = \frac{A^* P_c}{c^*} \tag{4.7}$$

Specific Impulse:

$$I_{SP} = \frac{c^* \gamma}{g_0} \sqrt{\left(\frac{2}{\gamma - 1}\right)\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1 - PR^{\frac{\gamma-1}{\gamma}}\right]} \qquad [4.8]$$

Velocity change:

$$\Delta V = g_0 I_{SP} \ln\left(\frac{m_0}{m_0 - m_p}\right) \qquad [4.9]$$

Force:

$$F = A^* P_c \gamma \sqrt{\left(\frac{2}{\gamma - 1}\right)\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1 - PR^{\frac{\gamma-1}{\gamma}}\right]} + P_e A_e \qquad [4.10]$$

All variables are defined in Table 4.1, except *R,* which is the gas constant for the propellant being analyzed and g_0 is the Earth gravitational constant.

The ΔV and I_{SP} produced over a range of AR inputs are shown in Figure 4.1. As can be seen, the I_{SP} and accordingly ΔV have an asymptotic behavior that approaches a limit with increasing AR. In this analysis ΔV and I_{SP} are functions of PR only, which is numerically determined from AR. The asymptotic behavior corresponds to the relationship of PR approaching zero as AR increases. In the vacuum of space, the ideal scenario is to have total flow expansion, PR = 0, resulting in no pressure gradient at the nozzle exit.

From a ΔV perspective, it is beneficial to have a higher AR, which corresponds to a smaller throat area and diameter. For example, an AR of 100 is achievable with a throat diameter of 0.5 mm (0.0197 inches).

The propulsion system is intended not only for orbital maneuvers but also attitude control where it is more important to study the thrust performance of the nozzle. The thrust of the nozzle is a function of propellant properties, as well as the mass flow rate ($\dot{m}$) which is driven by A^*. The thrust produced from the nozzle as a function of AR is shown in Figure 4.2.

Figure 4.1 Performance Parameters for Nozzle Area Ratio

Figure 4.2 indicates that as AR increases, throat area asymptotically decreases, and the thrust production is also asymptotically decreased. The decreased thrust performance is due to the reduced throat area, which decreases $\dot{m}$ through the nozzle. Thrust is reduced even though the smaller A^* drives the PR closer to zero, maximizing exhaust velocity. The asymptotic behavior is a consequence of the A^* value which is a reciprocal relationship of the linearly varying input AR.

Along with thrust production, it is also advantageous to analyze the predicted time of thrust. Figure 4.3 displays the total thruster exhaust time that is achievable as a function of AR. As shown, the total time of thrust is based on $\dot{m}$ which has a linear relationship with A^*. Smaller A^* restricts $\dot{m}$, reducing propellant consumption and consequently increasing duration of propulsion.

Figure 4.2 Range of Thrust Production for Nozzle

Figure 4.3 Range of Thruster Total Exhaust Duration for Nozzle

For attitude control maneuvers, it is necessary to find a compromise between both the asymptotic thrust magnitude as well as linear thrust duration, to design a nozzle that meets all requirements. As an example, an AR = 110 produces ~51 mN thrust and a total exhaust time of 10 minutes. If the AR is increased to 140, the thrust produced drops 21.5% to ~40 mN, yet total time increases 27.5 % to 12.75 minutes.

4.3. NOZZLE GEOMETRY

With the knowledge and understanding of the performance of the possible nozzle geometries, it is possible to define the final design for MR SAT. The primary emphasis of the propulsion system is to maximize ΔV and consequently the duration of the formation flight phase. A high AR achieves maximum ΔV, however, this reduces the thrust production which is more critical for attitude control maneuvers. A compromise has to be made to meet both requirements as well as the geometric considerations, which are addressed in this section.

The nozzle is manufactured by Micro Aerospace Solutions (MAS) of Melbourne, Florida, who have a history of developing thruster systems for microsatellites. Technical questions and nozzle concepts were discussed with engineers at MAS to assist in the design of the MR SAT nozzles.

The previous analysis was performed as a function of AR for clarity and graphical displays. In reality, it is necessary to consider the throat area and diameter that corresponds to these AR values. The maximum exit diameter has been set at 5 mm for structural integration reasons as discussed in Section 6.2. As previously mentioned, it is highly advantageous to maximize the exit area and consequently increase the PR and performance; for these reasons the exit diameter has been set at the maximum limit of 5 mm. The corresponding throat areas for the AR range analyzed is: AR of 50, D_t = 0.707 mm and AR of 150, D_t = 0.408 mm.

When designing the nozzle AR, consideration for the throat area and its construction must be made. The probability of impurities and condensation or even propellant freezing, inducing blockages, is increased with a small throat diameter. Liquid droplets in the flow of a larger nozzle can cause losses as great as 5%, however it is anticipated that a nozzle of this reduced size would be more substantially affected by two phase flow and the losses may in fact be greater [40].

Another aspect to consider is the construction of the nozzle and the tolerances and accuracy achievable by the manufacturer. MAS is capable of machining to an accuracy of 0.001 inches or 0.0254 mm. Consequently, a 0.5 mm throat diameter could vary as much as ± 0.0254 mm, which results in an AR range of 90.56-110.99. The smaller the throat diameter becomes the larger the resulting AR range becomes, which reduces the accuracy of the theoretical prediction of the nozzle.

It is also important that the nozzle can be manufactured with minimal imperfections as these can reduce performance. A smooth nozzle surface minimizes losses from friction and convective heat transfer. Boundary layer effects caused by wall friction in the nozzle can reduce effective exhaust velocity by 0.5 to 1.5 %, but the losses are anticipated to be amplified by the small geometry of the MR SAT nozzle [40].

The shape of the nozzle is also important and requires consideration during the design phase. The de Laval or convergent-divergent nozzle is the standard rocket configuration which utilizes a contoured converging inlet that smoothly joins through the throat to a bell-shaped diverging cone. The bell-shape configuration is advantageous as it constrains the flow lines to remain in the axial direction, reducing divergence losses and maximizing thrust. A disadvantage is that the length and mass of a bell is larger than a cone nozzle with an equivalent AR.

In the inlet converging section of the nozzle, the geometry is not particularly important as long as the flow is subsonic (a desired condition). The flow can be turned easily with minimal pressure losses and attain Mach 1 at the throat and accelerate supersonically in the diverging section. The converging section of the MR SAT nozzle will consist of a cone shaped inlet that joins directly to the inlet tubing from the valve. Additional minor losses in pressure, thrust and exhaust velocity can be anticipated for this nozzle design due to the small area ratio between inlet tubing and throat area. These losses are taken into account in the refined engineering model, with inclusion of the pressure losses anticipated in the feed lines.

Accurately manufacturing a 5 mm bell shaped diverging nozzle for MR SAT was considered excessively difficult at this time. Like most small-scale nozzles, a straight sided cone was chosen as the diverging section, as it was simpler and easier to manufacture. A length constraint based on the dimensional envelope for MR SAT as shown in Section 1.5.3, also favored this configuration.

A cone shape design meets requirements, however, is not ideal as there are inefficiencies associated with this nozzle configuration. The sharp edge of the throat where the cones meet interacts with the hypersonic flow, generating shocks and causing performance losses. There are also losses created from the divergent flow lines in a cone shaped nozzle. For theoretical analysis, a conical correction factor (λ) can be used to quantify the ratio of gas momentum of a diverging flow to an ideal axial flow. This correction factor is related to the divergent half angle (α) and is implemented in the analysis in Section 4.4.

Incorporating the thermodynamic performance analysis results, as well as the structural integration limitations and geometric considerations, and after consultation with MAS personnel, the resulting nozzle designed for MR SAT has the following properties: AR = 100, D_e = 5 mm, D_t = 0.5 mm.

These parameters maximize ΔV for orbital maneuvers while still providing sufficient thrust for attitude control as verified with current attitude determination and control (ADAC) subsystem simulations. The compromise of thrust, $\dot{m}$ and resulting total time of thrust exhaust is acceptable. Converging and diverging nozzle components will be cone shaped and meet at a point at the throat to simplify manufacturing. The diverging nozzle exit has a 30 º half angle providing a compact nozzle length for structural integration, meeting geometric flight envelope requirements. To highlight this effect with an example, a half angle of α = 30 º requires a 3.897 mm length of the divergent section of the nozzle. If the angle is reduced to α = 15 º the length increases 4.5 mm to 8.397 mm. A drawing of the nozzle with all geometric properties is displayed in Figure 4.4.

MAS can manufacture nozzles from stainless steel, aluminum, or polyetheretherketone (PEEK). PEEK is an engineered thermoplastic which has excellent thermal stability as well as fatigue resistance and superior chemical resistance. Although it offers good thermal characteristics and is lightweight the use of PEEK was not pursued and it was chosen to use the more conventional small nozzle material, stainless steel.

This option simplifies connection methods to the stainless steel valves as well as offering equal thermal expansion, a concern in this region of the propulsion system where highly varying temperature fluctuations are anticipated.

Figure 4.4 Nozzle Design Draft Dimensions

4.4. REFINED ENGINEERING MODEL

With the geometry of the nozzle defined, it is possible to refine the engineering model and analyze the performance of the system with higher fidelity to include expected losses and provide more realistic results. The analyses to this point have utilized a refrigerant that maintains a sealed container status in the storage tank. This low pressure condition severely limits the performance of the system. ADAC/Orbit simulations indicate that the requirement of conducting formation flight for a minimum of one orbit can not be obtained without increasing the propellant on MR SAT. In order to store more propellant it is necessary to violate the sealed container status and request a waiver from AFRL to permit the propellant tank to function as a pressure vessel.

This engineering model examined three different maximum pressure regimes: the sealed container limit of 689.48 kPa (100 psia) as well as the elevated 1378.96 kPa (200 psia) and 2068.44 kPa (300 psia). These pressure ranges were selected as they greatly improve the performance characteristics yet can still be considered a low-pressure system when compared to current propulsion systems used on spacecraft. These pressure regimes can also be implemented with the current MR SAT hardware components and still meet factors of safety.

The conservative temperature range of -50 ºC to 100 ºC was retained to ensure safety margins are maintained. Using the maximum temperature and pressure, the mass of propellant for each storage condition can be calculated. The mass and thermodynamic properties are highlighted in Table 4.2. The superheated temperature indicates the temperature required for the saturated liquid to become a single-phase, superheated gas when heated (i.e. in Pressure Regime 3, the propellant will have to be heated to 45.5 ºC to become a single-phase, superheated vapor).

Table 4.2 Example R-134a Tank Storage Conditions

Pressure Regime	**Maximum Tank Pressure at 100 ºC [kPa (psia)]**	**Internal Energy [kJ]**	**Propellant Mass [grams]**	**Superheated Temperature [ºC]**
1	689.48 (100)	18.765	60.523	15.4
2	1378.96 (200)	39.788	130.795	36.3
3	2068.44 (300)	64.200	215.873	45.5

This engineering model implements correction factors to improve the accuracy of the theoretical ideal performance. The factors incorporated in this analysis are based upon the data ranges contained in [40]. These factors are obtained experimentally by comparing real system test results with ideal theoretical analysis. The correction factors used here were not calculated from testing results of the MR SAT propulsion system; instead they were selected to be the middle of the specified ranges.

λ – A conical nozzle correction factor was implemented to account for the losses that are a result of the divergent flow in the nozzle. The conical correction factor can be applied directly to the momentum term of the thrust equation, (not the pressure term). The divergent half angle for this nozzle is $\alpha = 30\,^{\circ}$, indicating that only 93.3% of the ideal exhaust velocity can be achieved.

$$\lambda = \frac{1}{2}\left(1 + \cos\alpha\right) \qquad [4.11]$$

ζ_v – The velocity correction factor is directly related to the energy conversion efficiency, which quantifies the ratio of the kinetic energy per unit of flow of an actual nozzle flow to an ideal nozzle flow. The range of ζ_v is between 0.85 and 0.99 with an average of 0.92. An estimated correction factor $\zeta_v = 0.9$ was used for this analysis. It was applied to I_{SP} with the relationship

$$I_{SP\_ACTUAL} = \zeta_v \, I_{SP\_IDEAL} \qquad [4.12]$$

ζ_d – The discharge correction factor quantifies the ratio of the mass flow rate of a real nozzle flow to that of an ideal nozzle flow. The range of ζ_d is between 1 and 1.15, as the resulting mass flow rate of a real nozzle flow increases. An estimated correction factor $\zeta_d = 1.08$ was used for this analysis. It can be related to $\dot{m}$ with the relationship

$$\dot{m}_{ACTUAL} = \zeta_{\mathrm{d}} \dot{m}_{IDEAL} \qquad [4.13]$$

ζ_F – The thrust correction factor is used to correct the actual thrust produced, which is less than the ideal thrust. ζ_F has values in the range 0.92 – 1.00, but can be calculated with I_{SP} and mass flow rate correction factors with the relationship $\zeta_F = \zeta_d \times \zeta_v$. A correction factor $\zeta_F = 0.972$ is calculated for this analysis. The thrust relationship is determined by

$$F_{ACTUAL} = \zeta_F F_{IDEAL} \qquad [4.15]$$

The analysis also considered the pressure losses that occur in the feed lines from regulator to nozzle. This can be a significant loss in real systems and should be accounted for. Section 8 has further information on actual system losses experienced during testing. For simplification a direct pressure loss has been incorporated, where the nozzle inlet pressure was selected to be 6.8948 kPa (1 psi) below the regulator pressure of 170.30 kPa (24.7 psia, 10 psig). This factor takes into consideration losses from friction and interference of feed lines, bends, fittings and valves. The 6.8948 kPa (1 psi) pressure loss value has been selected based on pipe pressure losses that have been experienced with testing R-134a in the UMR laboratory. Fluid temperature for this analysis was set at 20 °C as this resembles a realistic desired temperature.

The final factor implemented to the analysis is an estimated margin that only 90% of the initial mass of propellant is utilized for propulsive expulsion. This has a direct impact on the ΔV calculation only. This margin accounts for the unreliable low pressure propellant expulsion as the tank empties and any additional losses that may be associated with undesired phase changes. At 20 °C, when the tank pressure reaches the regulated pressure of 170.30 kPa (24.7 psia, 10 psig), 17.82 grams of propellant remains in the tank. This corresponds to 8.26 % of the total initial propellant stored in Pressure Regime 3. The pressure regulator is capable of functioning when the tank pressure drops below its preset output, however, this factor of 90% usage was selected as it generates more conservative results. The calculated ΔV has a linear sensitivity to this mass margin with a mild derivative. If the real system does suffer significant propellant waste expenditure, the ΔV will suffer a reduction, with a relationship $\frac{d\Delta V}{dm_p} = 3.76$ for Pressure Regime 3. The lower the pressure regime limit the lower this sensitivity derivative becomes.

The analysis conducted is based upon theoretical modeling, and although it is higher fidelity, it is still not without limits in accuracy. In order to implement the thermodynamic, energy and momentum theory of rockets, it is still necessary to make the following assumptions:

- Isentropic flow in the nozzle
- Isothermal fluid in tank and propellant lines
- Refrigerant remains in the gaseous state and obeys the perfect gas law
- No shock waves or discontinuities in the nozzle flow
- Nozzle boundary layers are ignored and flow is axially uniform (1-D flow)
- Propellant flow is constant with no open/close transient effects

A number of these assumptions are justifiable under the condition that the propulsive maneuvers are limited to short durations with significant pauses in between, and that tank heating is implemented. Other losses are accounted for with the correction factors used to generate performance predictions that are more representative of a real system. Table 4.3 highlights the predicted performance parameters that are achievable with the MR SAT nozzle design.

Table 4.3 Engineering Model Predicted Performance Parameters

Isp	44.09 sec
Thrust	62.79 mN
$\dot{m}$	0.1481 g/s

Table 4.4 highlights the performance parameters that are achievable for the three pressure regimes that were calculated in the engineering model. As shown, there are significant advantages in increasing the maximum allowable tank pressure. Increasing the maximum tank pressure threefold to 2068.44 kPa (300 psia) increases the ΔV by a factor of 3.57 to 3.374 m/s from the 689.48 kPa (100 psia) ΔV of 0.943 m/s. The thrust duration represents the capable total time of propellant exhaust, based on $\dot{m}$, and is only achievable if the temperature remains constant.

Table 4.4 Predicted Performance for Three Pressure Regimes

Max Tank Pressure at 100 °C [kPa (psia)]	**ΔV (m/s)**	**Total Thrust Exhaust Duration (mins)**
689.48 (100)	0.943	7.10
1378.96 (200)	2.041	15.34
2068.44 (300)	3.374	25.31

The results offer a more accurate representation of the performance capabilities of the true MR SAT propulsion system. For comparison, the pressure loss experienced downstream of the regulator was increased to a conservative 68.948 kPa (10 psi) below regulated pressure. In effect, the nozzle inlet pressure is only 101.3525 kPa (14.7 psia). The propellant temperature was also reduced to a more conservative 15 °C and the results are shown in Tables 4.5 and 4.6. As shown the effect on ΔV and ISP is relatively small, however the mass flow rate and thrust encounter a more significant penalty.

Table 4.5 Conservative Predicted Performance Parameters

Isp	43.71 sec
Thrust	37.37 mN
$\dot{m}$	0.0889 g/s

Table 4.6 Conservative Predicted Performance for Three Pressure Regimes

Max Tank Pressure at 100 ºC [kPa (psia)]	**ΔV (m/s)**	**Total Thrust Exhaust Duration (mins)**
689.48 (100)	0.935	11.34
1378.96 (200)	2.024	24.52
2068.44 (300)	3.345	40.46

5. HARDWARE REQUIREMENTS AND SELECTION

5.1. BASIS FOR HARDWARE DESIGN

A primary limitation of the MR SAT propulsion system, along with numerous other spacecraft developers, is funding to purchase hardware. Flight proven and space qualified hardware by nature is extremely expensive. Designing and manufacturing custom “one-off” components can quickly raise costs beyond a university-based budget. It is, therefore, often necessary to purchase inexpensive commercial off the shelf (COTS) items that are not certified for spaceflight.

It is still possible to utilize these COTS components in the spacecraft provided they meet the necessary safety and integration requirements and thorough testing is performed. While not being space rated, these components may have been designed and manufactured to other industry and military specified standards that hold some level of accreditation toward space worthiness, if not at least safety.

The most important aspect when selecting hardware components for a spacecraft propulsion system is safety. It is necessary to mitigate catastrophic hazards that arise as a result of hardware failure. There is heightened fear of a propulsion system failure because of the stored energy of a pressurized fluid and its dynamic and active nature that changes with environmental variations and usage. Along with the UNP UG and the advice of UNP officials, the 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*. Details of this document are included in subsequent sections where individual hardware components are discussed.

A general requirement of the UNP 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 is a measure of the release of gas by a non-metallic 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 the UNP), are a maximum CVCM of 0.1% and a TML of 1.0% or less. The hardware components and their integration to the MR SAT bottom plate are shown in Figure 5.1.

Figure 5.1 MR SAT Propulsion System Hardware Components

The components that constitute the MR SAT propulsion system include a tank, that is used to store the propellant. The tank also requires special fittings along with a fill and drain valve which is used during ground operations. The system requires valves which are used as a safety inhibit in the propulsion system as well as the mechanism to control and time thruster pulses. Two pressure transducers will be used in the system to sense the propellant pressure in the tank as well as in the thruster feed lines. A pressure regulator is used in the system to maintain a lower and fixed propellant pressure to the thrusters. The propellant will be piped throughout the system with stainless steel tubing utilizing Swagelok fittings. The system will also incorporate heaters for active thermal control and Multi Layer Insulation (MLI) as a passive thermal control device. The details of each hardware component is described in the following sections.

5.2. TANK REQUIREMENTS AND CONSIDERATIONS

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 selection of the tank for MR SAT was constrained by the requirements of both UNP and UMR SAT system requirements.

The UNP requires that the tank meet the following requirements that are sourced from the UNP UG, NASA NSTS 1700.7B, Military standard 1522A - *Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems* as well as direct communication with UNP representatives. The requirements include:

- Factor of safety greater than 5 (Burst : MDP)
- Structural fatigue test diagnostics
- Leak before burst failure
- Metal construction (No composites or over-wrapped tanks)
- Constructed/welded by certified manufacturer
- Space certified and tested highly preferred
- Flight history preferred

The additional UMR requirements for MR SAT integration are:

- 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

The primary tank requirement corresponds to a pressure factor of safety relating – Burst : MDP. The proof pressure is the maximum nondestructive pressure obtained during testing. Burst pressure is the pressure at which the tank will fail.

A requirement of the tank for MR SAT integration is the propellant management device (PMD). A PMD fitted tank is used to contain and control propellants that will be stored as a saturated liquid, such as R-134a. The intent of the PMD is to reduce liquid sloshing during maneuvers as well as allowing only vapor extraction, a highly desired feature. The PMD consists of internal baffles and screens that provide additional surface area and reduces liquid movement. The extra surface area also increases the interaction and attraction of liquid through surface tension. In the zero gravity environment of space this causes the liquid globules to adhere to the internal PMD and promotes only vapor extraction to occur.

Preliminary tank investigations involved discussions with UNP personnel, professors and students on the propulsion design teams of other universities participating in Nanosat-4. Shawn Miller, another UMR SAT propulsion team member, contributed significantly to tank research and considerations.

It was decided early in research that it was not feasible to design and develop a spacecraft tank at UMR due to the significant time and experience required, as well as the precise manufacturing and testing required. Vendors marketing space qualified tanks as well as more contemporary tank manufacturers were contacted and the results of the investigation are summarized here.

ATK PSI is a manufacturer of propellant and pressure tanks for the aerospace industry. A number of the ATK range of tanks are space qualified and have spaceflight heritage. The PMD integrated models are not suited for small spacecraft as they were well beyond mass and dimensional limitations. There were no “off the shelf” space qualified tanks available, which greatly increases cost and lead time if a tank were to be ordered. A high altitude aerospace tank was available for purchase, rated to extremely high pressures (41.37 MPa, 6000 psi, proof pressure). However, it was heavy, 3.4 kg (7.4 lb), and was too large with a maximum dimension of 419.1 mm (16.5 inch) and consequently its use was not pursued.

Early in the investigation, communication was initiated with Carleton Technologies Inc. manufacturers of lightweight composite pressure vessels for the aerospace sector. A suitable tank was available that could be implemented on MR SAT in terms of pressure, volumetric and dimensional requirements. Later UNP correspondence however, strongly advised against any composite or over wrapped tanks for the NS4 competition. In addition, the Carleton COTS tanks do not feature a PMD and it was also found that the resin used did not meet outgassing requirements. Carleton Technologies Inc. was willing to manufacture the tank with a space qualified resin. This option was not pursued as it would have required configuration changes, special lot setup and acceptance testing significantly inflating the tank expense.

In an attempt to source and compare non-aerospace tank vendors, Catalina Cylinders who produce high quality aluminum cylinders and Luxfer Gas Cylinders who produce seamless aluminum tanks were contacted. Tanks were available that met volume, mass and dimensional requirements and were priced significantly lower than their aerospace counterparts. These tanks, however, do not feature a PMD and are not manufactured to any space qualification, or appropriate military or Department of Transportation standards. For these reasons, the choice of one of these non-aerospace tanks was not pursued, as the tank is a vital component of the system and there can be no compromise on safety.

5.3. TANK SELECTION

The search for a tank was not limited to the United States. Marotta UK Ltd. was contacted initially in October 2005 and lead to the delivery of the BSS01 tank in January 2007. Discussions on design and implementation were carried out during that time period to negotiate requirements, international trade regulations and budget. The tank purchase did require United Kingdom Department of Trade and Industry “End-User Undertaking” documentation, however it was free of USA International Traffic in Arms Regulations (ITAR), which would have significantly increased administrative paperwork and delivery time.

The BSS01 tank is a model designed for small spacecraft propulsion systems intending to store saturated liquids. These tanks are currently in LEO on four SSTL developed Disaster Monitoring Constellation (DMC) satellites. The first satellite Alsat-1 was launched in November 2002 and UK-DMC, NigeriaSat-1 and BilSat-1 were launched in September 2003. The spacecraft used the tanks to store saturated liquid butane propellant that was stored at a maximum absolute pressure of 400 kPa (58.0 psia) at 40 ºC and released through a 50 mN resistojet [41].

The tank was selected and purchased as it meets the requirements of both the UNP competition and the MR SAT mission. As well as having space qualification and flight heritage, the tank features a PMD system that utilizes mesh baffles and insert disks as well as an impurity filter. The tank has a mass of 1.4769 kg (3.256 lb) and has a maximum dimension of 370.6 mm (14.59 in). The tank is intended for butane storage with a MDP of 400 kPa (58.0 psi). Consultation with Marotta engineers indicated that the tank could be used for the refrigerant R-134a with a MDP of 1378.96 kPa (200 psia) and the tank was proof tested to at least that pressure. The tank is shown in Figure 5.2 and test data of pressure testing and all tank properties are displayed in Table 5.1.

The BSS01-01 tank delivered for MR SAT was purchased at a reduced price due to the fact that it was an extra tank manufactured in a previous batch. The tank also has a construction imperfection that occurred during assembly: at each end of the tank is a 25 mm across-flat (A/F) cut out that can be used for integration, and the delivered tank has a misalignment of approximately 37 degrees with respect to each A/F cut out. This has no effect on the functionality of the tank and is not used for MR SAT integration.

Figure 5.2 Tank (with Heater Attached) Integrated in MR SAT

Table 5.1 MR SAT Tank Properties and Configuration – Marotta BSS01-01 [42]

Mass of empty tank – measured	1.4769 kg (3.256 lb)
Volume – measured	2.459 L
Temperature range	-40 °C to 75 °C
Proof pressure – measured	1.62 Mpa (235 psi)
Burst pressure (minimum recorded – leak not burst) [3]	9.8 Mpa (1421 psi)
Factor of safety (burst pressure : proof)	6.05
Leak rate (He – 0.81 Mpa) – measured	2×10^{-10} std. cm^3/sec
Mesh baffles (PMD)	Aluminum alloy
Insert disks (PMD)	Aluminum
20 micron filter	Stainless steel
Purchase price	$9800 USD

[3] S. J. Edwards, “RE: Marotta tank enquiry,” Email correspondence with author, Tuesday, February 07, 2006 6:49 AM.

5.4. FILL AND DRAIN VALVE AND GROUND CONNECTION

The Marotta BSS01 tank has a designated fill end that requires a valve that can be utilized for filling and draining. Marotta also develops a fill/drain valve, VC02-007 that can be integrated with the BSS01 tank. The VC02 valve is space qualified and has flight heritage on SNAP and the DMC series of spacecraft. The VC02 valve has a complimentary coupling connection that is used during ground operations, designated IN-CA01. With the purchase of the tank the VC02 and IN-CA02 parts were provided at a heavily discounted price. The fill and drain valve with end cap and integrated on the tank in MR SAT is shown in Figure 5.3.

Figure 5.3 Fill and Drain Valve with End Cap

The primary seal of the VC02 fill/drain valve is achieved with an internal spring driven poppet. A secondary seal is provided with an external cap. The VC02 valve is designed to pressures of 62 MPa (9000 psi), however the MR SAT valve was proof tested to only 4.7 MPa (680 psi) as this was sufficient for MR SAT requirements. The leak rate tests performed on the MR SAT valve are for both the sealed configuration and the ground connection configuration. The details of the VC02 fill and drain valve are shown in Table 5.2.

Table 5.2 MR SAT Fill/Drain Valve Properties – Marotta VC02-007 [43]

Mass - measured	42 grams
Temperature range	-40 °C to 70 °C
Proof pressure - measured	4.7 Mpa (680 psi)
Designed proof pressure	62 Mpa (9000 psi)
Leak rate (He – 0.4 Mpa) - measured	2 x 10^{-7} std. cm^3/sec
Leak rate coupled (He – 0.4 Mpa) - measured	1 x 10^{-5} std. cm^3/sec

The IN-CA02 ground connection was manufactured for MR SAT with a straight input tube with a ¼ inch Swagelok adaptor for laboratory attachments at UMR. Figure 5.4 shows the ground half coupling device IN-CA02 which features the Swagelok fitting on the inlet tube and the tank valve connector probe on the other. The ground connector is shown in the clean bag it was delivered in and also features kapton tape on the inlet and outlet ends for cleanliness and protection.

Figure 5.4 Tank Ground Connection

It was required to design a connection piece that mated the tank inlet (7/16” -20 UNJF – 3A) to the fill/drain valve connection (1/8 inch BSP Parallel with ‘O’ ring). After consultation with Marotta a piece was designed with assistance from Swagelok. A draft view of the connection is shown in Figure 5.5. This connection was manufactured by Swagelok and is made entirely of stainless steel.

Figure 5.5 Special Fitting – Tank Inlet to Fill/Drain Valve

It was also necessary to design a custom fitting to connect the tank outlet to the feed lines and remainder of the system. This reduced the cumbersome dimensions of using two standard connections, which was originally considered, and allowed tank integration within the tight confines of the hexagonal prism structure as shown in Figure 5.6. Reducing the connections is also a requirement as it reduces the possible sources of leaking in the system. This connection was also manufactured by Swagelok and is made entirely of stainless steel.

Figure 5.6 Special Fitting – Tank Outlet to Feed Lines

5.5. ISOLATION AND CONTROL VALVES

The valves will serve two primary functions in the 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 required to have one of the three inhibitors as a “fail-safe” valve where it will close in the absence of an open electrical signal. The first valve is the isolation valve and isolates the propellant in the tank from the remainder of the system. It is most practical to use this first valve from the tank as a “fail-safe” fitted isolation valve. The second valve is located downstream of the regulator and is the second independent interruption in the flow lines. The third inhibitors are the valves at each individual nozzle assembly. A schematic of the propulsion system and valve locations are shown in Figure 5.7.

Figure 5.7 Schematic of Propulsion System with Valves Shown

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. The electrical circuitry for the valves is being designed and manufactured accordingly by the Command and Data Handling (C&DH) subsystem with vendor assistance.

For simplicity and ease of manufacture, it was decided that each of the ten valves would be identical and all are consequently "fail-safe" and will close without electrical signals. The search for MR SAT valves began with well known propulsion system vendors. Moog was contacted and offered cold gas solenoid valves, model 51E190, at a cost of $15 000 each and with a lead time of nine months, which was outside the MR SAT budget. Vacco Industries was contacted and was willing to develop a partnership to manufacture hardware designed by the UMR SAT team. This was not considered for MR SAT as this would not allow feasible build, test and integration under the UNP time restrictions.

Micro Aerospace Solutions uses a COTS micro-dispense solenoid valve for their systems and offered a valve and nozzle package to UMR SAT that was subsequently selected for integration. The valve selected for MR SAT is the INKX0507800A and is a special purpose model manufactured by the Lee Company. The valve is not space rated nor believed to have been used in space, but it was specified by the MIT SPHERES program for use. The MAS valve/thruster system is currently undergoing vacuum testing by MAS as part of a space-qualification process. The valve constructed as a thruster sub assembly is shown integrated onto the MR SAT side panel in Figure 5.8. The internal workings of the valve are displayed in Figure 5.9.

Figure 5.8 Valve with Nozzle Integrated on Side Panel

Figure 5.9 Valve Internal Design [Lee Co.]

There were initial concerns that the operating pressure of the valve would decrease with elevated temperatures. The Lee Co. indicates a reduction to 0 - 0.207 MPa (0 - 30 psi) at 150 °C (300 °F). Discussion with MAS indicated that a smaller, less capable model valve has been extensively tested by MAS with N_2 and hydrogen peroxide cold gas systems operating at 820.37 kPa (120 psi), without failure, alleviating some concern. These tests involved the exterior wall temperatures of the valve reaching as high as 260 °C (500 °F) due to friction heating and prolonged solenoid use [44]. This problem is considered minor, given the testing that has been conducted, and the relatively low pressure that the MR SAT valves will be operating under. However, as a precaution complete and thorough valve laboratory testing will be conducted by the MR SAT propulsion team to ensure functionality across the entire operating envelope.

The valve is constructed of Stainless Steel 316 and uses an ethylene propylene diamine monomer (EPDM) seal, which is a durable, high density rubber. The eight nozzles, manufactured by MAS will each be attached to a valve. The inlet side of the valve will be attached to a Swagelok fitting for connection to the feed lines. For further information on the thruster design, refer to Section 6.2. The two isolation valves downstream of the tank and regulator are respectively fitted with 1/8 inch Swagelok fittings on both inlet and outlet for attachment to feed lines.

The maximum average open/close power is 0.75 Watts with lower power required to keep the valve in the open position. The valve requires a 24 V spike for actuation which can be achieved from the MR SAT standard 5 V bus with special circuitry. A pulse width modulation (PWM) circuit with a 50% duty cycle is being utilized to reduce power consumption. It is also required as stated by NASA NSTS 1700.7B that system components such as valves have an ultimate factor of safety of at least 2.5. Specifications of the valve including the required pressure factors of safety are summarized in Table 5.3; for additional details refer to reference [45].

Table 5.3 MR SAT Valve Specifications - Lee Co. [45]

Mass	7 grams
Proof pressure (Lee Co. rating)	5.17 MPa (750 psi)
Burst pressure (Lee Co. rating)	7.76 MPa (1125 psi)
Rated thermal environment	-18 °C to 70 °C
Open response time - 689.48 kPa (100 psig)	0.25 ms
Close response time - 689.48 kPa (100 psig)	< 3.0 ms
Actuation Voltage	24 V spike
Actuation power (maximum average)	0.75 W

5.6. PRESSURE REGULATOR

The pressure regulator is used to take the fluctuating tank pressure and reduce it to a lower and more consistent pressure for the thruster. There were a range of pressure regulators that were researched during the MR SAT propulsion design, however there was only one that met requirements. Space qualified regulators such as the Moog model 50E741 was priced at $50 000 and had a lead time of twelve months which was well beyond the budget and time frame of MR SAT. It was, therefore, required to source COTS regulators that did not possess space qualification but could function in a vacuum environment.

A feature of numerous regulators is a reference pressure port and vent hole. This poses a significant issue as this is incompatible in vacuum environments and discouraged from use in the UNP. Models with this feature were disregarded from consideration, such as the Beswick range of piston regulators. Another vendor, Tescom, was contacted to pursue the use of their 04 series regulators. The 04 series regulators are inexpensive, however, they have a pressure adjustment device which was strongly discouraged by UNP personnel even though it featured a tamper resistant assembly. For this reason, along with the pressure port concerns, the Tescom regulators were not considered for MR SAT use.

Swagelok also produces precise compact regulators for use in fluid systems and was finally selected as the pressure regulator vendor for MR SAT integration. The model HFS3B regulator was chosen, as its compact and inline design allows for easy integration. It is preset to the desired output pressure and is a completely sealed unit meeting specification requirements.

During production in the factory, the regulator is charged with an inert gas to obtain the preset output pressure. It was necessary to set the regulator output to a gauge pressure of 68.9 kPa (10 psig) which is equivalent to an absolute pressure of 170.3 kPa (24.7 psia). Also during production, the regulator was fitted with 1/8 inch Swagelok connectors on inlet and outlet for direct integration with the MR SAT propulsion system feed lines. This was uniquely selected over the split nut connection which is standard on the regulator. The HFS3B model regulator is shown integrated in MR SAT in Figure 5.10.

Figure 5.10 Pressure Regulator

Another important feature, which was confirmed by Swagelok engineers prior to purchase, was the highly desired, over-pressure security of a "leak before burst" system. NASA NSTS 1700.7B requires an ultimate factor of safety of at least 2.5 on system components such as regulators. This is achieved with the HFS3B which is capable of inlet pressure ranges up to 6.89 MPa (1000 psig). Additionally, the HFS3B is capable of functioning when the inlet (tank) pressure drops below the 68.9 kPa (10 psig) regulated pressure. During this scenario, the outlet pressure is equivalent to the tank pressure. The properties of the HFS3B are highlighted in Table 5.4.

Table 5.4 MR SAT Pressure Regulator Specifications - Swagelok HFS3B-WU5-P10

Preset outlet pressure	170.3 kPa (24.7 psia)
Mass - measured	176 grams
Temperature range	-40 °C to 70 °C
Inlet pressure range	Vacuum to 6.89 MPa (1000 psig)
Operating temperature range	-23 °C to 65 °C
Orifice Size	3 mm (0.12 in)
Flow Capacity	100 std. L/min
Leak rate (He)	1×10^{-9} std. cm^3/sec

5.7. PRESSURE TRANSDUCER

A pressure transducer converts the displacement of a strain gauge, under pressure, into an electrical signal that can be computed as a pressure reading of the fluid. The decision was made that two pressure transducers would be required for the MR SAT propulsion system. One would be placed at the tank outlet to continuously monitor the pressure in the tank. This is required as it will provide data on the tank operating conditions along with supplementary thermal control sensing. The second transducer would be placed downstream of the regulator. This position is necessary for performance calculations as it allows the fluid pressure to be measured regardless of operating temperatures and accuracy of the preset regulator. While it is beneficial to use additional pressure transducers for redundancy or for more accurate propellant property measurement, mass and costs restricted the design to two transducers.

The pressure transducers selected for MR SAT integration are the COTS model AS17A manufactured by Honeywell/Sensotec. The AS17A is a flight rated model tailored specifically for aerospace applications that require an absolute pressure reading. Consequently, they are also relatively compact and low mass without compromising durability and accuracy. NASA NSTS 1700.7B requires an ultimate factor of safety of at least 2.5 on system components such as transducers. The AS17A meets this requirement as it is designed to sense pressures up to a maximum of 68.9 MPa (10,000 psia). The details of the Sensotec AS17A transducer can be found in Table 5.5.

Table 5.5 MR SAT Pressure Transducer Specifications – Sensotec AS17A

Pressure range	0 to 1378.96 kPa (0 to 200 psia)
Mass - measured	140 grams
Operating temperature range	-54 °C to 121 °C
Material	Stainless Steel
Pressure Port	7/16-20 UNF
Electrical Connection	PTIH-10-6P

Correspondence with Sensotec engineers indicated that the AS17A could be manufactured with a custom connector for MR SAT integration. This was not pursued as it would result in additional design and testing and increased costs, when an adaptor could be more easily substituted. The MR SAT transducers are factory set to read an absolute pressure range of 0 to 1378.96 kPa (0 to 200 psia). At the time these transducers were ordered, the MDP was limited to 689.48 kPa (100 psia), so this value was chosen to offer a factor of two margin over MDP. The transducer without electrical connector is shown in Figure 5.11 integrated in MR SAT.

Figure 5.11 Pressure Transducer Integrated

5.8. LINES AND CONNECTIONS

The selection of lines and connectors was assisted by the knowledge and experience of the UNP officials. They suggested the use of Swagelok trademark connections, for their easy and secure assembly and low leak rates over other connections such as “AN” flare fittings. Aluminum ¼ inch outer diameter (OD) tubing was originally selected as the feed lines to connect the tank through the system to the eight thrusters located around MR SAT.

Since Swagelok connectors were not available in aluminum, it was decided to utilize stainless steel (SS) tubing at the smaller 1/8 inch OD. This smaller OD tube counteracts the increased mass of using SS (43.156 grams/m) and also allows easier integration into the tight confines of MR SAT. Other advantages of the SS tubing are the increased strength, reduced thermal loads and reduced potential leak sources as the entire system is assembled with SS. NSTS 1700.7B requires an ultimate factor of safety of at least 4.0 on all pressurized lines and fittings. Swagelok 1/8 inch fittings and stainless steel seamless tubing is capable of working pressures up to 58.6 MPa 8500 psig maximum pressure, well above the required MDP.

5.9. HEATERS AND MLI

Heaters were implemented on the tank and on the feed line to regulate the thermodynamics of the system and improve propulsive performance. As previously mentioned, when a thruster is pulsed the tank temperature will decrease as the liquid propellant vaporizes to maintain saturation pressure in the tank. The primary function of the tank heater is to prevent and counteract this temperature loss, improve system response, and ensure the propellant maintains its optimum thermodynamic properties.

The heaters selected for use on MR SAT are provided by Minco and are composed of a heating element with a polymide film (Kapton) insulator and an aluminum backing for mounting. All materials, including the adhesive, meet outgassing requirements. At full power, the tank heater has a rating of 3.63 Watts while the line heater is rated to 1.06 Watts. The line heater attached to the stainless steel tubing integrated in the MR SAT system is shown in Figure 5.12.

In order to efficiently utilize the heaters and conserve any heat loss due to radiation, the tank will be wrapped in MLI. MLI consists of numerous layers of insulation carefully constructed to provide a thermal blanket. Mantech, NASA Goddard Space Flight Center will supply MLI for the MR SAT propulsion system. The heaters along with the monitoring of conditions with thermal sensors and pressure transducers will be implemented as an active thermal control loop. The MLI is used as a passive thermal control feature.

Figure 5.12 Heater Attached to Line

6. SPACECRAFT INTEGRATION

6.1. THRUSTER CONFIGURATION

The thruster placement and their configuration in the satellite defines the control authority the propulsion system has for attitude corrections. It also governs the efficient use of propellant for orbital and attitude maneuvers. MR SAT requires complete three-axis attitude control capability. To achieve this requirement, a minimum combination of thrusters and their arrangement can be used. It is also necessary to align the thrusters with respect to the cg. It is desirable to position the thrusters on the extremity of the satellite as this increases the perpendicular distance from the cg, increasing the torque exerted and the effectiveness of the system for attitude rotational control.

Early in the MR SAT propulsion system design, trade studies were conducted on the thruster configurations that could be utilized. Twelve, ten, nine and eight thruster configurations were considered. The greater the thruster numbers, the greater the efficiency of the system in minimizing propellant usage during attitude corrections. This is because during an attitude rotational maneuver it is not necessary to pulse a control thruster as well as an opposing thruster. If only a control thruster is pulsed, the spacecraft will experience an attitude rotation but also a linear motion. The secondary opposing thruster, that ultimately must have a force directed through the cg, isolates the attitude rotation. With a twelve thruster system, it is necessary to always pulse two opposing thrusters, however they are used collectively, doubling the effective torque and halving the pulse time. The increased complexity, costs and additional hardware, internal routing and integration in a small satellite severely limit large number thruster configurations. For this reason, it was chosen to implement a simpler and more cost effective eight thruster configuration in MR SAT as shown in Figure 6.1.

This MR SAT eight thruster configuration is not easy to implement with the hexagonal structure. Significant research and collaboration with the Integration and Structure subsystems were conducted to integrate the propulsion system into MR SAT. UMR SAT team members Lori Ziegler and Noah Ledford provided significant inputs into propulsion integration and structural modifications.

Figure 6.1 MR SAT Hardware Components - With Panels and Thrusters

Because the propulsion system will be utilized when MRS SAT is undocked, it was necessary to position the thrusters relative to the cg of only the MR SAT spacecraft. Small errors in thruster alignment are inevitable and can be accounted for with accurate measurements integrated into the control software, as well as being actively monitored and adjusted on-orbit. The magnetic coils used for attitude control will also be utilized to assist in correcting induced errors in attitude due to small thruster misalignments.

The nominal on-orbit attitude is MR SAT positioned with the z-axis normal to the orbital plane as shown in Figure 6.2. This will align all thrusters within orbit plane. The advantage of this configuration is that it allows any of the thrusters to be used to perform an in-plane maneuver to maintain formation, reducing attitude corrections, fuel consumption and time in preparing the spacecraft attitude for a maneuver.

As there are tight constraints on the cg location, it was assumed for thruster integration that the cg is approximately in the geometric center of the MR SAT structure. Consequently, the thruster integration is intended to align the nozzle thrust vectors with the geometric center. The center is with respect to the structural side panels, not the honeycomb solar panels which extend beyond the structure in the Z axis direction. It is very common to add ballast weight to slightly shift the cg to a more desired position and this may be implemented on MR SAT.

Figure 6.2 On-Orbit Formation Flight Attitude Configuration of MR SAT

6.2. THRUSTER SUBASSEMBLY AND INTEGRATION

The thruster subassembly incorporates the valve which has the nozzle attached to the outlet and a Swagelok connector on the inlet line. This subassembly is manufactured as one piece by MAS and will be referred to as the "thruster." Given the eight thruster orientation and the hexagonal shape of MR SAT, it was necessary to integrate the thrusters in two primary positions as shown in Figure 6.1. In the first position are thrusters which are aligned perpendicular to Panels 1 and 4, the second position being in the corners where Panel 2 meets 3 and 5 meets 6. Integrating this configuration required two separate thruster subassemblies to be designed and manufactured. The five thrusters (1-4 and 7) that are integrated perpendicular to the panels are referred to as "flat thrusters." Figure 6.3 shows the flat thruster and its dimensions. The three thrusters in the corners (5, 6 and 8) are referred to as "corner thrusters." Figure 6.4 shows the corner thruster and its dimensions.

The primary difference between the two thrusters is the orientation of the Swagelok fitting. The thrusters are designed in such a way that the Swagelok fitting will be utilized as an alignment and attachment point to the side panel. On the flat thruster, the nozzle is aligned perpendicular to a side on the hexagon section of the fitting. On the corner thruster, the nozzle is aligned with a point of the hexagon section of the fitting. The valve is also supported by a hole that it protrudes through in the side panel. This support hole was the limitation for the nozzle outer diameter, as described in Section 4.3, to be no greater than the valve diameter, as it was required to fit through this side panel hole. It is most advantageous to position the nozzle and valve on the exterior of the side panels as there is extremely limited space within the spacecraft. The combined length of the nozzle and valve was critical on the flat thrusters to ensure that the dimensional envelope was not breached. However, this envelope restraint has since been relaxed, through discussions with UNP personnel.

Presenting the prototype spacecraft and thrusters to a UNP design review allowed consultation with officials about thruster attachment methods. It was decided to attach the thrusters to the panels with a space rated zip tie around the fixed Swagelok hexagon. There is also a casting compound, Arathane 5753 which is used for additional support as well as providing some dampening between thruster and panel. Testing indicated that a misalignment between the rotating nut of the Swagelok fitting and the fixed hexagonal section that is zip tied, would not compromise the secure attachment.

Figure 6.3 Draft View of Flat Thruster

Figure 6.4 Draft View of Point Thruster

There was also concern that the thin valve tubing (1.6 mm OD) would not support the weight of the valve and the nozzle. This would also contribute to vibrations induced by launch and thrust pulses. For these reasons, it was decided to encase the valve inlet tubing (between the Swagelok fitting up around the 90° bend to the valve) with an outer tube sleeve. This tube sleeve was already deemed necessary to attach the fine valve tubing to the Swagelok fitting. Secondly, all eight thruster valves will have additional support from the side panel hole into which they are inserted. This essentially removes all loads from this tube bend and reduces the concerns of vibrations. The hole size selected is 6.8 mm and allows for the Arathane 5753 casting compound to be used as a securing agent that will fill the gap between valve and hole wall.

It is also necessary for the aluminum honeycomb, which is separated from the panel with 15.5 mm spacers to be modified for thruster integration. Custom holes and cut outs have been implemented for the thrusters to protrude through with significant clearance. Early designs for valve support holes using custom honeycomb inserts were discarded for a number of reasons. First, the honeycomb is not designed to bear any load. Also this may induce cross vibration and increase the likelihood of misalignment issues. Consideration was also made for design concepts of additional brackets that attach directly to the panel, to house and support the protruding valve. These were superseded by the preferred hole supports, due to complexity and dimensional integration issues.

The flat thrusters on Panel 1 are positioned in the middle of each side section. Thrusters 2 and 4 are attached directly to the side panel with the valve supported by a hole as shown in Figure 6.5. The thruster's Swagelok hexagon is positioned in the middle of the side bar on the panel so that when the zip tie is secured there is an equal force in the lateral direction. Thrusters 1 and 3 come close to the top plate brackets so it was necessary to customize their attachment. The bracket will be extended to have a secure attachment plate for the Swagelok hexagon rather than using the side bar of the panel as show in Figure 6.6. The valve is supported here in the curve of the isogrid cutout instead of with a hole in the panel.

Figure 6.5 Thruster 2 on Panel 1

Figure 6.6 Thruster 4 on Panel 1 - with Extended Support Bracket

Thruster 7 is located in the middle of panel 4 and is attached directly to the isogrid webbing with a zip tie. The valve is also supported by the panel, however it was necessary to increase the thickness of the webbing around that isogrid node so that the hole in the node could be increased to fit the valve. Thruster 7 is positioned in the geometric center of all the thrusters on the opposing panel 1. Figure 6.7 shows thruster 7 and its integration to panel 4.

Figure 6.7 Thruster 7 on Panel 4

The corner thrusters required an integration design approach similar to that of the flat thrusters. The thruster is attached to the side panel with zip ties and the valves are additionally supported by holes in the panel. The thruster is assembled so that the Swagelok fitting lies flat with the panel with the nozzle protruding at the desired 60 ° from the panel. This allows the thrust vector to be aligned through the geometric center of the spacecraft. It was initially envisioned that the thruster would sit neatly in between the two meeting panels and attached to both. Although this aligns the thruster directly through the geometric center of the spacecraft, the design was changed for the following reasons. There were concerns about vibrations between the two panels as well as the thruster being treated as a load path between the panels under stress; both being undesirable conditions. In addition, the width of the panels has been reduced so that there is now a larger gap in between adjacent panels, removing the option to connect to both.

As an alternative, the corner thrusters are attached to only one panel and are as close to the edge, where the panels meet, as possible. This reduces the thrust vector misalignment to 3.2 mm from the geometric centerline. The corner thrusters on opposing sides of the satellite are off the geometric centerline in the same direction, meaning that they are still aligned with respect to each other. The result is that the thrust control authority is still equivalent during attitude correction pulses, however there is a slight misalignment for orbital maneuvers.

After discussions with Lori Ziegler, (UMR SAT Integration Subsystem Lead) on integration procedures it was chosen to attach Thrusters 5 and 6 on Panel 2 and Thruster 8 on Panel 6. The batteries are also one of the heaviest items on board the satellite and it was estimated that the likelihood of the cg being toward that direction was greater.

In order to integrate these three corner thrusters, the panel was modified to allow zip tie attachment as well as to include valve support holes. The thruster feed lines interfered with brackets so it was necessary to extend the panel inward or outward to place the thruster off center from the panel side bar. It was decided to go outward as this placed the thruster closer to the geometric centerline as well as moving the thruster and lines away from the internal components on the panel. The thruster and panel modifications with hole support are shown in Figure 6.8. It was also necessary to create an additional hole in the panel to allow the zip tie to pass through securing the Swagelok fitting with equal lateral force.

Figure 6.8 Thruster 8 on Panel 6 - System Integrated and CAD

6.3. TANK AND SYSTEM INTEGRATION

The propellant tank, being the most prominent feature of the propulsion system, is positioned on the bottom plate of MR SAT. It is positioned here primarily for integration reasons, as the tank will not fit parallel to the Z axis, and because of the MRS SAT attachment mechanism (Qwknut) on the upper surface. It is most desirable to place the tank as close to the cg as possible as this reduces the movement of the cg as propellant mass is released. This is not possible on MR SAT due to integration limitations, but the movement of cg will be primarily in the Z-axis and can be compensated for in the control software. In addition, the mass of propellant is small relative to the entire spacecraft, and will have minimal effect on cg movement. As an example, the pressure regime of 2068.44 kPa (300 psia) can store a maximum of 215.873 grams of propellant, which is only 1.08% of a 20 kg satellite.

Custom tank mounts were created to support the tank from the hemispherical ends with attachments to the MR SAT bottom panel. This allowed clearance for tank heaters and MLI to be attached to the cylindrical section of the tank. This design also allowed the transducers, first inhibit valve, and regulator to be supported with a beam across the mounts. This was utilized as there is sufficient volume available in the spacecraft center, however, there are no attachment methods to support these components. After the regulator, the feed lines continue to a cross fitting where the lines split off to all eight thrusters. The propulsion system is shown in Figure 6.9.

The tubing is supported by the Swagelok fittings which are zip tied and potted directly to the panels in discrete locations. The tubing is primarily routed to clear subsystem components, but also aims to minimize overall length and bends. Only standard 30, 45, 60 and 90 degree bends are used to simplify manufacturing and minimize manufacturing errors. Feed line losses are unavoidable, however, lines were arranged to create equivalent flow lengths to corresponding thrusters. When two corresponding thrusters are pulsed simultaneously, the flow losses and thrust produced would then be approximately equal, reducing maneuver-induced errors.

On the inlet end of the tank, a fill and drain valve is attached. During ground operations, access to this valve is vital for both filling and if necessary, draining the tank. It was necessary to modify Panels 1 and 6 along with their respective honeycomb panels with cut outs. These modifications allow the ground support crew to access the screwed cap on the fill and drain valve as well as attach the ground coupling device.

Figure 6.9 Propulsion Sub Assembly

7. REFRIGERANT COMPATIBILITY AND MATERIAL SELECTION

7.1. R-134a COMPATIBILITY

One significant concern with cold gas propulsion systems is their tendency to leak. Propellant leakage significantly jeopardizes the mission life of a low pressure, small satellite propulsion system, where 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 where ever possible in a spacecraft propulsion system due to compatibility and outgassing concerns. As an alternative, all metal components are preferred, such as the Swagelok fittings that utilize a two-ferrule mechanical grip and seal feature.

Some hardware components cannot function without a non-metal seal. When selecting a seal compound for these components it is necessary to account for two important factors. First, the seal must be compatible with refrigerants, particularly R-134a in this case. 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. Secondly, the seal compound must meet outgassing requirements in a vacuum environment, which may also lead to the same degenerative effects, and potentially seal failure.

There are numerous sources that publish chemical compatibility data, so it is necessary to validate the authenticity of the source as the information can often vary or be incomplete. 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.

Chemical compatibility data of certain compounds is often not available as the process requires significant research and testing. In addition, the compatibility of compounds such as refrigerants is often not available. This is due to a number of reasons, including the fact that refrigerants can be manufactured as a combination of compounds, without a precise chemical makeup.

Compatibility of plastic or elastomer compounds is also heavily dependant on the application environment and the thermodynamic conditions. In the presence of additional compounds, plastics can react differently, influencing their compatibility rating. In addition, certain compounds display varying levels of compatibility and there is often no clear distinction between declaring two substances compatible or incompatible. For these reasons, it is important when investigating a compound for compatibility, as well as outgassing, that the precise material, manufacturer, model name and specifications are sought. This will ensure that the compound being considered for use has the properties that are desired and will not compromise propulsion design safety.

Dupont Fluoroproducts, the manufacturer of "Suva" 134a, produces an MSDS that states R-134a is incompatible with alkali or alkaline earth metals, such as powdered Al, Zn and Be. Dupont also publishes additional technical data sheets for their product which indicate that R-134a is chemically stable with steel and aluminum.

7.2. VALVE SEAL COMPATIBILITY AND MATERIAL SELECTION

There have been significant issues encountered in selecting a compound to use for the seals in the micro-solenoid valves. The Lee Co. manufactures the valve with a standard Viton® seal. Dupont Performance Elastomers manufacture the fluorelastomer (FKM), which has the trademark name Viton®. R-134a is not compatible with Viton® and, as published by Dupont, experiences a severely unacceptable change when they come in contact.

It was, therefore, necessary to find a replacement seal that would be compatible with R-134a as well as meet outgassing requirements. The Lee Co. also moulds valve seals with Kalrez®, ethylene propylene diene monomer (EPDM) rubber and silicone. Kalrez® is a trademark perfluoroelastomer (FFKM) manufactured by Dupont Performance Elastomers. It is also incompatible with fluorinated refrigerants such as R-134a and was not considered for use.

EPDM is a rubber compound that is manufactured by a number of vendors and goes by trademark names such as Nordel®, Royalene® and Vistalon®. Dupont manufactures Nordel®, which they indicate is compatible with R-134a. Silicones are an inorganic polymer that contains the organic element silicon. The silicone family of compounds come in numerous compositions with varying properties. The moldable silicone that Lee Co. uses as a valve seal is Silastic® 24020-V, manufactured by Dow Corning. Silicone in general is considered compatible with R-134a, however, no published data could be found on this specific silicone model.

The Lee Co. sent samples of both the EPDM and silicone for compatibility testing with R-134a. The samples were accurately measured in geometry, thickness and mass. They were then placed in a sealed container of R-134a which consisted of both liquid and vapor phases. After a month the samples were removed, measured and weighed. The EPDM had not changed and showed no indication of incompatibility. The silicone sample results were inconclusive and it is necessary to perform additional compatibility testing.

The silicone test has yet to be performed again to confirm this compatibility result. The R-134a used in the experiment was a COTS product that claimed to have no additional products such as lubricating oils, which is an important consideration when performing compatibility testing. Commercial compatibility testing was also investigated and was an option if considered necessary. It was not, however, pursued for these products due to time and cost limitations. The NASA technical standard NASA-STD-6001, *Flammability, odor, offgassing, and compatibility requirements and test procedures for materials in environments that support combustion,* February 9, 1998 was used as a reference when these compatibility tests were conducted.

It was also necessary to compare the outgassing properties of both the EPDM and silicone seal options. The EPDM used by the Lee Co. is a proprietary product, so the particular model or manufacturer, along with the outgassing data for the compound, could not be disclosed. A search on the NASA material outgassing database, lists EPDM seal products with outgassing figures as high as (TML 26.38 %, CVCM 14.62 %) and as low as (TML 3.42 %, CVCM 0.94 %) which is still above requirements. There were initial concerns with these outgassing figures, although MAS indicates that EPDM has been used in space applications. Outgassing testing sites were sourced both on campus and externally to locate a facility that could quantify the proprietary EPDM outgassing data and validate it for MR SAT integration.

Silicones and specifically the Dow Corning Silastic® were also searched on the NASA database and were found to have varying outgassing values. Some models met outgassing requirements, but the specific model 24020-V, was not listed on the NASA database. The Lee Co. indicated that the silicone they use is not rated as low outgassing, so with the initial test revealing incompatibility, this silicone option was not pursued.

Other low outgassing options were also presented to the Lee Co. for seal consideration and production. General Electric (GE) Silicones were contacted as they produce low outgassing silicone rubber compounds. GE manufactured RTV 567 with catalyst RTV 567B, as well as LVG342 (RTV1673LV). Both were considered, however, attempts by the Lee Co. to mould this silicone for the seal application failed. Nylons, such as the Dupont manufactured Zytel® resins offer refrigerant compatibility and can have low outgassing rates. Zytel® model (70G33HRLUG59D14) has outgassing rates that meet requirements, however, these were not pursued by the Lee Co.

The valve seal option remains under consideration with the EPDM compound requiring further testing and verification. There is no limit to its use in the laboratory, however, the unknown vacuum outgassing rates of the material are a concern which will not allow the MR SAT propulsion system to be launched as it stands. Outgassing facilities that meet both appropriate testing requirements and budget constraints are currently being sought.

7.3. SYSTEM COMPONENTS AND COMPATIBILITY ISSUES

The pressure regulator is another system component that requires a non-metal seal that comes in contact with the R-134a propellant. The Swagelok regulator uses an internal poppet that is used to generate a seal. This feature is manufactured with the polymer polychlorotrifluoroethylene (PCTFE) as standard, however, discussion with Swagelok engineers indicated that the poppet could also be made with polyimide.

Polyimide is a polymer that comes in a variety of compounds. Dupont produces Kapton® film which is a polyimide product used in space applications. Vespel® is the trade name of another Dupont polyimide product which can offer low outgassing properties. Dupont data indicates that there are no test results available to classify

Vespel® as compatible with R-134a. In addition, Swagelok was unable to identify the exact type of polyimide poppet used in the pressure regulator.

PCTFE is a homopolymer manufactured by Daikin under the trade name Neoflon®. PCTFE compounds offer extremely low outgassing properties that are strongly favored by the aerospace industry. Compatibility data with R-134a was not published by Dupont, however, one source indicates that contact of PCTFE with halogenated compounds, of which R-134a is one, can cause slight swelling. The regulator has been purchased with the PCTFE seal, however, this compatibility issue has not been directly addressed but has been deemed acceptable, with the following precautions. The pressure regulator is situated downstream of the first inhibit valve, so it will not have contact with R-134a until the propulsion system has reached operational status on-orbit. Additionally, the regulator has only a very short life expectancy on-orbit, as the propellant consumption will be rapid. This reduces the time the propellant has contact with the poppet, reducing the chance of any adverse effects. Thorough laboratory testing will also be undertaken with R-134a, to ensure the regulator performs its desired function without fault for the desired time duration.

The connection between tank and fill/drain valve is a custom connection designed with the assistance of Swagelok. The original design featured an “AN” seat insert that was made of nylon resin PA66. PA66 is not advertised as a low outgassing resin, and Swagelok investigated alternative seal materials. It was decided that the most cost effective mitigation was to develop the connection with an “all metal” design that has since been manufactured and integrated into the MR SAT propulsion system.

8. SYSTEM LOSSES – TESTING AND ANALYSIS

Theoretical analysis still has its deficiencies and will not account for all parameters and losses that occur in a real system. A fundamental step in the design process of a spacecraft propulsion system is to perform laboratory testing. One area of testing that can be easily implemented and whose results offer significant benefits is fluid dynamics. Losses that occur from wall friction, tube bends, fittings and valves can all be quantified in the laboratory. These losses have a direct impact on the fluid dynamic parameters that adversely affect the performance of the nozzle flow.

Using the refrigerant R-134a allows for easy and user-friendly testing as it is safe and readily available. As identified in Section 9, it is still necessary to implement handling procedures and laboratory ventilation to ensure personnel safety. The propulsion system testing has been conducted primarily by UMR SAT propulsion team member Chris Norgren. For more detailed information on testing and results, refer to the UMR SAT internal document 04-008 – *Propulsion System Testing Summary*.

The initial testing scheme quantified the pressure loss along a straight length of tubing. The shear stress that occurs with the interaction between fluid flow and tube walls is the greatest contributor to losses in a propulsion system. The objective of this test was to quantify the coefficient of friction (C_f) and related friction factor (f), which is a fluid dynamic property that can be used to determine fluid flow losses due to shear stress. By determining the friction factor for a set of known testing conditions, it is possible to numerically correlate shear stress losses for any tube length and for any thermal operating envelope. The coefficient of friction for R-134 is not well publicized for the gaseous phase, however, there are many sources for liquid and two-phase tube flow as it is more advantageous to the refrigeration industry.

Testing was first conducted using air as the working fluid to confirm the functionality and accuracy of the testing apparatus. The testing apparatus implemented for this preliminary testing scheme was basic laboratory equipment that was available in the UMR Mechanical and Aerospace department.

A schematic of the air testing apparatus is shown in Figure 8.1. The air testing apparatus used a 26.5 L (7 gallon) tank that could be connected directly to the laboratory's 689.48 kPa (100 psig) air supply. The tank outlet, which features a pressure gauge, is connected directly to a manual valve for on/off control. A manually adjustable pressure regulator is also used with an additional pressure gauge placed downstream of the regulator to measure flow pressure at tube inlet. The test tubing is made up of three sections of 1/8 inch (OD) Swagelok stainless steel tube. To ensure the flow in the test section was fully developed a development section (a) and a terminating section (b) was implemented. Sections a and b are both 177.8 mm (7 inch) in length. Two straight test sections of tubing of lengths 200 mm (7.875 inch) and 406 mm (16 inch) were used. The test section has a differential pressure gauge that measured a full scale deflection of 34.47 kPa (5 psi). All tubing is connected with Swagelok fittings.

Figure 8.1 Testing Apparatus Schematic for Air

To conduct the R-134a testing, it was required to alter the hardware apparatus to supply the refrigerant working fluid. R-134a was purchased in 340 gm (12 oz) containers and were connected directly to the on/off control valve. The tubing test section and hardware apparatus downstream of the valve were equivalent to the air testing apparatus. Due to the significant endothermic nature of releasing R-134a it was necessary to immerse the refrigerant source container in a water bath that was regulated at a temperature of 40 °C. The temperature was manually controlled with a thermometer and heating element. This temperature was selected as it was safe, easy to maintain and produced good flow conditions for testing. The water bath more efficiently transferred heat to the cooling source container of R-134a during testing. A schematic of the R-134a testing apparatus is shown in Figure 8.2.

Figure 8.2 Testing Apparatus Schematic for R-134a

This testing provides the ability to quantify the pressure losses of R-134a flow in 1/8 inch stainless steel tubing. These results are still limited to test sections of these lengths and with similar thermodynamic conditions. A quasi one-dimensional (1D) flow solver was created to analyze the tube flow and numerically determine the friction factor based on testing results. This program can then be used, with an extrapolated friction factor estimate, to analyze tube flow for any operating condition or hardware configuration. The quasi 1D FORTRAN program numerically solves the thermo and fluid dynamic differential equations of the flow through a straight tube. The program numerically steps through the discretized tube solving the equations of continuity, momentum, energy and gas equation of state as shown in Equations 8.1 – 8.4.

Continuity:

$$\frac{d\rho}{\rho}+\frac{du}{u}+\frac{dA}{A}=0 \qquad [8.1]$$

Momentum:

$$\frac{dP}{\rho}+udu=-\frac{\tau_w cdx}{\rho A}=-\frac{\rho_1 u_1 C_f cdx}{2\rho A}=-\frac{\rho_1 u_1 fcdx}{8\rho A} \qquad [8.2]$$

Energy:

$$C_P dT+udu=\delta q_{convection}=\frac{C_f C_P cdx}{2A}(T_W-T_t) \qquad [8.3]$$

Equation of State:

$$\frac{dP}{P}=\frac{d\rho}{\rho}+\frac{dT}{T} \qquad [8.4]$$

Substituting Equations 8.1 – 8.3 into Equation 8.4 the system of differential equations is reduced to one equation with one unknown, velocity (*u*).

$$du=\frac{\left(\dfrac{\rho_1 u_1^2 C_f cdx}{2AP}+\dfrac{\delta q}{TC_P}\right)}{\left(\dfrac{1}{u}+\dfrac{u}{TC_P}-\dfrac{u\rho}{P}\right)} \qquad [8.5]$$

It is then possible to solve the flow parameters with back substitution of the governing equations to determine the conditions at the current axial position using:

$$u_i=u_{i-1}+du \qquad [8.6]$$

$$\rho_i=\frac{\rho_{i-1}u_{i-1}A_{i-1}}{u_i A_i} \qquad [8.7]$$

$$T_i=T_{i-1}+\frac{1}{2C_P}\left(u_{i-1}^{\ 2}-u_i^{\ 2}\right)+\frac{\delta q}{C_P} \qquad [8.8]$$

$$P_i = \rho_i T_i R \tag{8.9}$$

$$M_i = \frac{u_i}{\sqrt{\gamma R T_i}} \tag{8.10}$$

As seen in the equations, the quasi 1D solver incorporates all fluid and thermo dynamic properties of the fluid including convective heat transfer from the wall, compressibility effects and the wall shear stress. The program is initiated with inflow conditions that are based upon experimental data from laboratory testing. With an input pressure drop over a known length of tubing, the program was reiterated to determine the friction factor that produced a pressure drop to match test data. The average velocity at the test section inlet ($\overline{V}_{AVG\_INLET}$) was approximated by measuring the mass flow rate ($\dot{m}$) of the propellant. The mass flow rate is assumed constant throughout the test section tubing. With both air and R-134a, the mass flow rate was recorded over several time intervals and then converted to an average velocity using the inlet temperature and pressure as well as the tube inner area.

$$\overline{V}_{AVG\_INLET} = \frac{\dot{m}}{A\rho_{NLET}} = \frac{\dot{m}RT_{NLET}}{AP_{INLET}} \tag{8.11}$$

The friction factor was determined over a range of regulator pressures that resemble suitable low pressure propulsion systems. Based on this experimental data, a relationship between friction factor and the regulated pressure as well as the nondimensional, Reynolds number, has been established. Using this relationship a friction factor was then determined for a new set of flow conditions in a new length of test section tube, 174.625 mm (6.875 inch) long. The quasi 1D program was used to recalculate the estimated pressure drop. The pressure drop calculated was then compared to experimental test pressure drops over this length of tubing. This test determined the correlation and accuracy between the experimental system and the friction factor trend lines. The results of this test and the error between predicted pressure drop and the experimental pressure drop are displayed in Table 8.1. The experiment was conducted in a laboratory where the apparatus was initially at room temperature, 22.5 °C, and the water bath temperature was 40 °C.

As shown, the pressure losses in the pipe flow were fairly accurately modeled with this testing scheme. The pressure in the MR SAT system is regulated to 170.30 kPa (24.7 psia, 10 psig) which for the testing conditions produces a pressure drop of 12.9 kPa (1.875 psi) over the 174.625 mm (6.875 inch) length of tubing. This testing scenario does not incorporate any flow constricting devices such as valves and nozzles. In order to apply this analysis and data it is necessary to correlate the friction factor for the MR SAT hardware configuration.

Table 8.1 Friction Factor Determination Accuracy for R-134a Tube Flow [46]

Regulated Pressure (psig)	**Mass Flow Rate Average (g/s)**	**Friction Factor Average**	**Predicted Pressure Drop (psi)**	**Experimental Pressure Drop (psi)**	**True Value % Error**
22	1.023	0.01116	3.225	3.425	5.84
20	0.978	0.01024	2.965	3.175	6.60
18	0.889	0.01153	2.774	2.975	6.75
15	0.789	0.01152	2.456	2.600	5.55
10	0.655	0.01034	1.807	1.875	3.61
5	0.552	0.00850	1.298	1.300	0.13

The longest tube length from regulator to nozzle is 796.8 mm for thruster three. The mass flow rate will be constricted to 0.1481 g/s by the nozzle throat. The estimated pressure loss for this condition is 472.034 Pa (0.0685 psi).

This loss estimate does not take into consideration the tube bends and additional hardware components that the flow encounters to reach the nozzle. As an intermediate step it is possible to numerically account for the losses using a dimensionless constant known as the equivalent length (L_e/D). The equivalent length is used to represent the losses for a given fitting or bend and is calculated from published experimental data. It is used by adding an equivalent length to the numerical analysis, essentially increasing the total length of the tube flow. The values for L_e/D used here are based upon the data presented in the text, *Introduction to Fluid Mechanics* [47], and should only be used as a representation of typical hardware data. Further testing is planned to measure losses that occur in tube bends with R-134a. Typical Equivalent lengths, L_e/D for fittings and bends are shown in Table 8.2.

Table 8.2 Dimensionless Equivalent Lengths (L_e/D) for Certain Line Hardware [47]

Flow condition	L_e/D
Gate valve – Full open	8
Tee – Flow through run	20
Tee – Flow through branch	60
Bend - 90 degrees	30
Bend - 45 degrees	16

Using these equivalent lengths, a more accurate pressure loss experienced by the propellant between regulator and thruster can be calculated. Using the same example of thruster three there are two valves, four fittings and eight bends encountered downstream of the regulator. This equates to an equivalent length $L_e/D = 388$ and subsequently a length extension of $L_e = 591.312$ mm. The total length of tubing for thruster three, used in the estimation is $L = 1.38811$ m. The resulting pressure loss estimate is calculated to be 4285.09 Pa (0.6215 psi). The parameters used to calculate this pressure loss with the quasi 1D solver are shown in Appendix B2.

9. SAFETY CONSIDERATIONS

9.1. SAFETY ASSESSMENT WHITE PAPER

Ensuring safety is of paramount concern and a priority when designing a spacecraft, particularly a propulsion system. Safety extends from ground operations, testing and integration all the way through launch to operation on-orbit. Concerns for personnel must be addressed along with laboratory facilities, launch vehicle and its primary payloads as well as the spacecraft itself and onboard subsystems. In addition to the safety measures discussed in this thesis, the Safety Assessment White Paper (SAWP) has been developed to mitigate any concerns of using a two-phase, refrigerant propellant in a small spacecraft. The concept of the SAWP was originally initiated by UNP officials with the intent to be used as a supplementary document that would be used with a SERB review.

The SAWP serves two primary purposes. Firstly, it addresses the concerns of using a two-phase propellant system on a small spacecraft, with a particular focus of attention on the refrigerants R-134a and R-123. It also aims to show that a two-phase refrigerant propellant can be stored as a pressure vessel status, (beyond sealed container limits), and still be deemed safe. This is a valid assumption, as shown in the SAWP, provided suitable measures are taken by the developer during design, analysis, testing and implementation.

The SAWP contains a complete safety assessment which is based upon the fundamentals of a Failure Modes Effects Analysis (FMEA). The safety assessment identifies potential hazards at the component and process level and gives a risk assessment based on the consequences of failure. It classifies the class of the hazard as tolerable, critical or catastrophic depending on the outcome of the failure. Also identified are the conditions/events that could trigger the hazard as well as possible mitigation procedures.

9.2. EPA REGULATIONS AND LEGAL USE OF REFRIGERANTS

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 U.S.A, the use and release of all refrigeration compounds is governed by the Environmental Protection Agency (EPA) Clean Air Act (CAA). The CAA is in place to protect the environment from ozone-depleting compounds (ODC) and limit greenhouse gas contributions and prevent human induced global warming. Title VI of the act, specifically Sections 604, 605, 608, and 612 addresses the regulations on the release of refrigerants.

Chlorofluorocarbons (CFCs) and hydrochlorofluorocarbons (HCFCs) are two classes of compounds that refrigerants can belong to. It is required, as per CAA Title VI, that all CFC and HCFC refrigerants in air-conditioning and refrigeration equipment be recovered during any intentional release. The CAA classifies CFCs as a Class I substance and HCFCs as a Class II substance. Since January 2000, it has been illegal to produce and sell Class I substances. The production and sale of Class II substances will be illegal no earlier than January 2015 and is currently restricted to licensed technicians, vendors and buyers. The refrigerant, R-123, is a Class II HCFC and is being replaced by other more environmentally friendly refrigerants.

Hydrofluorocarbons (HFCs) are an additional class of haloalkanes that contain no chlorine molecules, which are known to deplete the ozone layer. CAA Title VI dictates that HFCs are the environmentally friendly replacement for CFCs and an alternative to HCFCs. HFC refrigerants are still regulated with similar guidelines as CFC and HCFC refrigerants with regard to recovery and containment.

R-134a is a HFC, and consequently an alternate refrigerant that is friendlier to human health and the environment. A primary advantage of R-134a is that it has no restrictions on sales and can be purchased in small quantities “off the shelf.” This advantage has allowed laboratory testing to be implemented so readily and easily. The CAA dictates that it is illegal to release into the environment a refrigerant if the application it was used for is a heat transfer fluid (refrigerant cycle). Used as a spacecraft propellant, there are no legal requirements, however, there are ethical considerations when intentionally releasing the propellant. The quantities released are so small and the intent is for educational and research purposes which do not directly fall under a category of the CAA. On a similar note, the intentional release of propellant on-orbit is certainly not accounted for in the CAA. During laboratory testing, the amount of refrigerant intentionally released is extremely small in comparison to the amount released daily in the U.S.A. and the world by both private users and industry.

9.3. PHASE CHANGE IDENTIFICATION

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. A preliminary study of the system has been conducted to ensure that undesired phase changes do not occur. 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 release. So, the necessary hardware to achieve this must be incorporated. This section discusses the areas of a propulsion system where a phase change is likely to occur.

9.3.1. Phase Change Occurrence – Tank. The tank is intended to store the propellant in a two-phase 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 vapor. 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 and phase changes are addressed in the SAWP.

9.3.2. Phase Change Occurrence – Lines and Hardware Components. During maneuvers, the propulsion system is required to release only gas for optimum performance and efficiency. There are two possible scenarios for propellant extraction and line travel, both with a different phase change location. 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. The MR SAT propulsion system implements the first scenario where the tank utilizes a PMD system to extract only vapor. It is necessary that downstream of the tank the propellant remains in the vapor phase.

Given this scenario is implemented, it is important that the propellant remains a vapor downstream of the regulator. This is especially true if there is the possibility of a propellant remaining stored for extended periods of time in the lines. Due to the temperature variations expected on-orbit, if the stored propellant in the lines experiences a significant temperature drop there is the possibility of a phase change to liquid. Current simulations indicate that the propellant will not be stored downstream of the regulator for extended periods of time and spacecraft temperature would have to significantly drop for a phase change to occur at the regulator pressure of 170.30 kPa (24.7 psia, 10 psig).

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. This condition has been considered but safely discounted, as the propellant velocity is very high throughout the system, significantly reducing the static pressure and increasing the fluid temperature by friction. This in turn moves the refrigerant propellant thermodynamic properties farther away from a possible phase change.

The concern of a phase change from liquid to solid has not been considered as the temperature required for R-134a to perform this solidification is beyond the bounds of the temperature envelope. At atmospheric pressure, the freezing temperature for R-134a is -96.6 °C (-142 °F) [48].

9.4. 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 two-phase propellant. This section describes possible methods and actions that can be implemented for inducing and mitigating propellant phase changes.

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. The tank conditions are to be monitored with thermal sensors and a pressure transducer. The transducer will only monitor vapor pressure, so for a two phase propellant there is no direct method of monitoring propellant mass and quality without accurate measurement of propellant consumption. The tank also utilizes a PMD which contains the liquid propellant so that all phase changes occur only inside the tank.

While it is important to monitor the tank propellant conditions, it is also helpful for propulsion performance reasons to implement heaters and insulation to the tank to increase the energy of the vapor. Not only does this improve the performance of the system but also assists in preventing undesired phase changes to liquid farther along the system. Insulation will be implemented on the tank as MLI reducing heat loss to radiation.

As with the tank, there are benefits in monitoring the thermodynamic properties of the propellant in the lines and hardware of the system. Monitoring can be used primarily for performance characteristics but also for safety reasons. Monitoring in the MR SAT system will incorporate an additional pressure transducer and thermal sensors downstream of the regulator.

It is also possible to implement both heating and insulation to the lines and other system hardware to prevent undesired phase changes and for performance enhancement. This will be achieved on MR SAT with resistance heaters that are wrapped around the feed line upstream of the regulator. This placement ensures that any propellant outside of the tank will be heated and enter the regulator and the rest of the system as a vapor. Thermal coatings that absorb inward heat radiation and reduce outward heat radiation were also considered by the MR SAT thermal subsystem, however they will not be implemented due to budget constraints.

9.5. LATENT HEAT CONSIDERATIONS

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 the state and the fluid temperature remains constant. As an example, during a heating process the energy (enthalpy change) associated with a change of state from solid to liquid is known as the latent heat of fusion. Similarly, 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 its boiling point, with units of kJ/kg. Refrigerants are intended to 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) [49]. The strong endothermic nature of the latent heat of vaporization of refrigerants has been 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 in temperature. From a safety view point, this 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 deplete 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 undergoes thorough analysis and laboratory testing. This will ensure the propellant properties and conditions are known and best utilized and the performance levels are maintained.

10. CONCLUSION

This thesis documents the procedures of developing a small spacecraft cold gas propulsion system. This includes the entire design process from research to analysis, design, manufacture, integration and testing. The result is a design template that was utilized for the MR SAT propulsion system and discussed as a case study in this thesis. The methodology and techniques developed can be implemented by other small spacecraft developers searching for a safe, low cost propulsion system.

The system designed for MR SAT meets the requirements of the AFRL UNP as well as the stringent requirements of payloads intending to fly on the NASA Space Shuttle. The propulsion system has been designed to meet the sealed container requirements, however, in order to meet mission objectives it is necessary to increase the tank pressure into a pressure vessel status so that more propellant can be stored.

The propulsion system designed and implemented on MR SAT is a cold gas system that can be implemented for both orbit maneuvers as well as three-axis attitude control. This is achieved with eight thrusters that are geometrically placed around the spacecraft. The propellant of choice is refrigerant R-134a, which will be stored in two phases on the spacecraft. Utilizing a two-phase propellant allows substantially more mass to be stored in the liquid phase, maximizing mission life, while still allowing the vapor to be extracted and used as a conventional cold gas. The system implements a flight proven tank designed for saturated liquid propellants with an internal PMD.

A primary advantage of R-134a propellant is that it can be safely and easily implemented for testing in the laboratory, making it ideal for university-based developers. The thermodynamic and fluid dynamic properties offer good performance characteristics. Engineering models and laboratory testing have been performed and included in this thesis to validate expected on-orbit performance parameters.

Extended areas of research beyond the scope of this thesis can be performed for the MR SAT propulsion system. This primarily includes extended laboratory testing of hardware, from component level, subsystem level through to complete integrated system tests in the spacecraft. It is also necessary to perform thrust measurement testing on the nozzles to confirm the results of the engineering model. With the completion of these tests, the system can be completed validated and quantified for its implementation as the propulsion system for MR SAT. The completed propulsion system integrated into the MR SAT structure in the UMR clean room is shown in Figure 10.1.

Figure 10.1 Propulsion System Integrated in MR SAT Structure

APPENDIX A.

MATLAB PROGRAMS USED FOR PROPULSION SYSTEM THERMODYNAMIC AND FLUID DYNAMIC ANALYSIS AND R-134a PROPELLANT PERFORMANCE MODELING

A1. Nozzle Design

```
% MASTERS RESEARCH - Nozzle Design from AR
%
% This program calculates the Performance Parameters (Delta V, Isp, Thrust
% and continuous thrust duration) for a range of nozzle geometries. The
% results are graphically represented as a function of Area Ratio.
%
% The operating design point along with other constraints used are:
% # 60.52 mass of propellant for 100 psia tank conditions
% # regulated pressure of 137.95 kPa, 20 psia
% # temperature of 20 C
% # nozzle exit diameter 5 mm
%
% Uses the function plotter(AR, dV, Isp) to plot a dual axis figure
%
% Carl Seubert
% July 2006, Nov 06, Feb 07
%

clc, clear all, close all
format compact, format long g

Mo = 25             ;% Spacecraft mass (kg)
Vol = 0.0025        ;% Tank Volume(m3)
g = 9.81            ;% gravity (m/s2)
mass = 60.523e-3    ;% propellant mass @ 100 psia (kg)

% ANALYSIS CONDITIONS
Tc = 293.15     ;% Maintained Temperature (K) [20 C]
Pc = 137.95e3   ;% Regulated pressure absolute (N/m2) (20 psia)

Tcr = 101.05    ;% Critical Temperature (C) [374.2 K]
Pcr = 4.06e6    ;% Critical Pressure absolute (N/m2) [588.9 psia]
Pr = Pc/Pcr     ;% Reduced Pressure for Compressibility factor
Tr = Tc/Tcr     ;% Reduced Temperature for Compressibility factor

% R134a - Ideal Gas Properties
gam = 1.127     ;% Specific Heat Ratio @ analysis conditions [EES]
M = 102.03      ;% Molar Mass (kg/kmol) [Wong]
Ru = 8314.51    ;% Universal Gas Constant (J/kmol.K)
R = Ru/M        ;% Gas Constant (J/kg.K)
a0 = sqrt(gam*R*Tc)     ;% sonic velocity

% Characteristic Velocity - Humble p. 139
cstar = a0/(gam*(2/(gam+1))^((gam+1)/(2*gam-2)));
```

```
% Integration/Structural Limitations on Nozzle Exit
De = 5e-3           ;% Exit diameter (m)
Ae = pi*De^2/4      ;% Exit Area (m2)

% NOZZLE Parameters as a function of AR
for i = 1:149
    AR(i) = i+1 ;
    Dt(i) = sqrt(4*Ae/(pi*AR(i)))   ;% Throat diameter
    At(i) = Ae/AR(i)                ;% Throat area
    PR = PRfromAR(gam, AR(i))       ;% Call PR function
    PRs(i) = PR                     ;% Store PR
    Pe = PR*Pc                      ;% Exit Pressure (N/m2)

    % Specific Impulse (seconds)
    Isp(i) = (cstar*gam/g)*sqrt((2/(gam-1))*(2/(gam+1))^((gam+1)/(gam-1))*(1-PR^…
              ((gam-1)/gam)));

    % Change in Velocity (m/s)
    dV(i) = g*Isp(i)*log(Mo/(Mo-mass(1)));

    % Force (N)
    F(i) = At(i)*Pc*gam*sqrt((2/(gam-1))*(2/(gam+1))^((gam+1)/(gam-1))*(1-PR^…
           ((gam-1)/gam))) + Pe*Ae;

    % mass flow rate (kg/s)
    mdot(i) = At(i)*Pc/cstar;
end

% PLOTTING

plotter(AR, dV, Isp)

figure
plot(AR(50:end), F(50:end).*1000, '*-')
xlabel('Nozzle Area Ratio AR (A_e / A_t)' )
ylabel('Thrust (mN)'), grid on

figure
plot(AR(50:end), (1/60)./(mdot(50:end)./mass(1)), 'r*'), grid on
xlabel('Nozzle Area Ratio AR (A_e / A_t)' )
ylabel('Total Thruster Exhaust Duration (minutes)')
```

A2. Subroutine – Pressure Ratio from Aspect Ratio

```
function Pressure_Ratio = PRfromAR(gam, AR)
% Calculates the Pressure Ratio (Pe/Pc) in a nozzle
% for a given Area Ratio (Ae/A*) using a Newtons method
% based numerical solver.
%
% Carl Seubert
% February 2007

% simplification variables
top = sqrt( ((gam-1)/2)*(2/(gam+1))^((gam+1)/(gam-1)));
a = 2/gam;
b = (gam-1)/gam;

PR = 1e-4   ;% initial estimate
i = 1       ;% counter
diff = 1    ;% difference variable

while (abs(diff) > 1e-15) & (i <100)
    % PR function
    fP = AR - top/sqrt(PR^a*(1-PR^b));

    % PR derivative
    dfP = top*(PR^b*(a+b)-a) / ( 2*PR*(PR^b-1)*sqrt(-PR^a*(PR^b-1)));
    diff = fP/dfP   ;% difference calculation
    PR = PR - diff  ;% adjust PR solution
    i = i + 1       ;% prevents solution divergence
    if i == 100, fprintf('solution could not converge, AR:%1.0f', AR), end
end

Pressure_Ratio = PR;
```

A3. Subroutine – Data Plotter

```
function plotter(AR, dV, Isp)
% Plots delta V and Isp vs Area Ratio
% Carl Seubert
% November 2006

figure
[AX,H1,H2] = plotyy (AR, dV, AR, Isp);
```

```
set(H1,'marker','*')
AXIS([0 150 0 1.42]), grid on
xlabel('Nozzle Area Ratio (A_e / A_t)' )
set(get(AX(1),'Ylabel'),'String','Delta Velocity (m/s)')
set(get(AX(2),'Ylabel'),'String','I_S_P (seconds)')
```

A4. Tank Storage Conditions

```
% MASTERS RESEARCH - R-134a Tank Storage Conditions
%
% Calculates the maximum mass of R-134 propellant storable in a 2.5 L tank
% for a given pressure of 100, 200, 300 psi gauge
% Engineering Equation Solver EES is utilized for thermodynamic properties.
%
% Sealed Container Requirements
% Pmax = 689.48e3 ;% Max Tank Pressure (N/m2)
% Tmax = 373.15   ;% Max Temperature (K) (100 C)
% Umax = 19310    ;% Max Internal Energy (J)
%
% Carl Seubert
% July 2006, Nov 06, Feb 07
%
clc, clear all, close all
format compact, format long g

Vol = 0.0025    ;% Tank Volume(m3)

% Using EES thermodynamic properties for 100, 200, 300 psi and 100 C
rho = [24.209, 52.318, 86.349]      ;% Density (kg/m3)
u = [310.048, 304.2, 297.4].*1e3    ;% Specific internal energy (J/kg)
v = 1./rho          ;% Specific volume (m3/kg)
mass = rho.*Vol     ;% Max mass storable (kg)
U = u.*mass         ;% Internal energy for max mass (J)

for i = 1:3
    fprintf('\r%1.0f00 psi Maximum Pressure - R-134a Conditions:', i)
    fprintf('\rMass of propellant in tank: %4.3f g', mass(i)*1000)
    fprintf('\rInternal Energy: %4.3f kJ ', U(i)/1000)

    Tsat(i) = -143147*v(i)^5 + 15551*v(i)^4-65449*v(i)^3 + 13690*v(i)^2 ...
        - 1582.7*v(i) + 335.196 ;% temperature saturation occurs
    fprintf('\rTemperature saturation occurs (full): %4.1f C, %4.1f K\r'...
        , Tsat(i) - 273.15, Tsat(i))
end
```

A5. Refined Engineering Model

```
% MASTERS RESEARCH - R-134a High Fidelity Performance Analysis
%
% Calculates the performance parameters with correction factors and losses
% for three pressure scenarios: 100, 200, 300 psi gauge
% Engineering Equation Solver EES is utilized for thermodynamic properties.
%
% Carl Seubert
% July 2006, Nov 06, Feb 07

clc
clear all
format compact, format long g

% Tank Storage conditions [100, 200, 300 psia]
Vol = 0.0025                    ;% Volume (m3)
rho = [24.209, 52.318, 86.349]  ;% Density (kg/m3)
mp = rho.*Vol                   ;% Max propellant mass storable (kg)
Mo = 25                         ;% Satellite mass (kg)
g = 9.81                        ;% acceleration due to gravity (m/s2)

% ANALYSIS CONDITIONS
Tc = 293.15     ;% Maintained Temperature (K) [20 C]
Preg = 170300.5 ;% Regulated pressure absolute (N/m2) [24.7 psia]

% Assumed Pressure at Nozzle inlet (N/m2) -1 psi =  [23.7 psia]
Pc = 163405.7;

% Assumed Pressure at Nozzle inlet (N/m2) -10 psi = [14.7 psia]
%Pc = 101352.5;

% R134a - Ideal Gas Properties
gam = 1.127      ;% Specific Heat Ratio @ analysis conditions [EES]
M = 102.03      ;% Molar Mass (kg/kmol) [Wong]
Ru = 8314.51    ;% Universal Gas Constant (J/kmol.K)
R = Ru/M        ;% Gas Constant (J/kg.K)

% NOZZLE GEOMETRY
AR = 100        ;% Nozzle Area Ratio (Ae/At)
Dt = 0.5e-3     ;% Throat Diameter (m)
At = pi*Dt^2/4  ;% Throat Area (m2)
Ae = AR*At      ;% Exit Area (m2)

alpha = 30*pi/180           ;% Divergent half angle (rad)

% Correction Factors
```

```
lambda = 0.5*(1+cos(alpha)) % nozzle angle correction factor
zetav = 0.9                 ;% Velocity correction factor
zetad = 1.08                ;% discharge correction factor
zetaF = zetav*zetad         ;% thrust correction factor

PR = PRfromAR(gam, AR)  ;% Pressure ratio (Pe/Pt)
Pe = Pc*PR              ;% Nozzle exit Pressure (N/m2)

a0 = sqrt(gam*R*Tc)     ;% sonic velocity (m/s)

% Characteristic Velocity - Humble p. 139
cstar = a0/(gam*(2/(gam+1))^((gam+1)/(2*gam-2)));

% Performance Characteristics
% mass flow rate (kg/s)
mdot = zetad*At*Pc/cstar

% specific impulse (sec)
Isp = zetav*(cstar*gam/g)*sqrt((2/(gam-1))*(2/(gam+1))^((gam+1)/(gam-1))...
      *(1-PR^((gam-1)/gam)))

% Force (N)
F = zetaF*(lambda*At*Pc*gam*sqrt((2/(gam-1))*(2/(gam+1))^((gam+1)/...
      (gam-1))*(1-PR^((gam-1)/gam))) + Pe*Ae)

% Total Thrust time (min) [assume: temperature constant]
time = mp'./mdot/60

% Delta V (m/s) - 90% of propellant mass utilized
dV1 = g*Isp*log(Mo/(Mo - 0.9*mp(1) ))
dV2 = g*Isp*log(Mo/(Mo - 0.9*mp(2) ))
dV3 = g*Isp*log(Mo/(Mo - 0.9*mp(3) ))

% mass of propellant in tank when Regulated pressure reached
mreg = Preg*Vol/R/Tc;

% percentage of propellant mass when Regulated pressure reached
mreg*100./mp';
```

APPENDIX B.

FORTRAN PROGRAM – QUASI 1D SOLVER USED TO MODEL THE THERMODYNAMIC AND FLUID DYNAMIC FLOW PROPERTIES OF R-134a PROPELLANT THROUGH SYSTEM LINES AND NUMERICALLY DETERMINE FRICTION FACTOR RELATIONSHIPS

B1. FORTRAN Program – Quasi 1D tube flow

```
Program newtubes
!
!       Carl Seubert & Chris Norgren
!
!       April 2007
!
!       QUASI 1D Solver - For R-134a propellant loss analysis
!
! This program solves four differential equations, four unkowns to
! calculate the changing fluid parameters through a constant area duct.
! The four differential equations are solve simulatneously in a matrix
! format. The parameters are T, P, rho, and U
! These are calculated at discrete locations through the duct.
! The inflow properties are determined with a .txt input file
! The output data is presented to the screen as well as being written to a file

double precision :: DiffMat(4,1), BB(4,1), AA(4,4), INVA(4,4)
double precision :: length, A, gam, R, Cp, P1, T1, M1, ID, mu, Re1
double precision :: f, fu, fl, Cf, Tw, U1, rh1, c, dx, pi, mdot, j=0, dq
double precision, Dimension(10000000) :: U, rho, T, P, M, Tt, Pt, du, X, Area, Drag,
       Pdrop, Re
Integer :: k=1

! Open input file and store inflow variables
open(10, file='INPUT5.txt')
rewind (10)

read(10,*) length      ! tube length (m)
read(10,*) P1          ! pressure (N/m2)
read(10,*) mdot        ! mass flow rate (kg/s)
read(10,*) f           ! friction factor
read(10,*) ID          ! inner diameter
read(10,*) gam         ! Fluid Specific heat ratio
read(10,*) R           ! Gas Constant(J/kgK)
read(10,*) mu          ! Fluid viscosity (Ns/M^2)
read(10,*) T1          ! Fluid Inlet Temperature (K)
read(10,*) Tw          ! Tube Wall Temperature (K)

close(10)

! Duct and Inflow fluid parameters
pi = 3.14159265359
A = pi*(ID**2)/4              ![m2]
Cp = R*(gam/(gam-1))          ![J/kgK]
rh1 = P1/(R*T1)               !INITIAL DENSITY [KG/M^3]
```

```
U1 = mdot/(rh1*A)              !INITIAL VELOCITY [M/S]
M1 = U1/sqrt(gam*R*T1)         !INITIAL MACH
Re1 = rh1*U1*ID/mu             !INITIAL REYNOLDS
c = sqrt(4*A*pi)               !TUBE CIRCUMFERENCE [M]

! DETERMINE COEFFICIENT OF FRICTION
Cf = f/4                       !wall shear stress coefficient

! differential length interval (m)
dx = 0.000001

! store the initial conditions into arrays
U(k) = U1
rho(k) = rh1
T(k) = T1
P(k) = P1
M(k) = M1
Tt(k) = T(k)*(1+(gam-1)*(M(k)**2)/2)                       !TOTAL TEMP [K]
Pt(k) = P(k)*((1+(gam-1)*(M(k)**2)/2)**(gam /(gam-1)))      !TOTAL PRESSURE [N/M^2]


! open a data file and write the initial parameters to this file.
open (unit = 11, file = 'final2.txt', status = 'replace', action = 'write', position =
       'rewind')
write (11,100) 'length','Vel','Temp','Press','Mach','Dens','Ttl Temp','Ttl
       Press','Drag','Pdrop','Re'
100 format(a,t20,a,t40,a,t60,a,t80,a,t100,a,t120,a,t140,a,t160,a,t180,a,t200,a)

write (11,101) 0, U(k), T(k), P(k), M(k), rho(k), Tt(k), Pt(k),0,0,Re1
       101
format(g14.7,t20,g14.7,t40,g14.7,t60,g14.7,t80,g14.7,t100,g14.7,t120,g14.7,t140,g14.7,t16
       0,g14.7,t180,g14.7,t200,g14.7)

do k = 2,  floor(length/dx)

               ! Convective heat transfer
               dq = Cf*Cp*(Tw-Tt(k-1))*c*dx/(2*A)

               ! Matrix of differential equations   (LHS)
               AA(1,1) = 0
               AA(1,2) = 1/rho(k-1)
               AA(1,3) = 1/U(k-1)
               AA(1,4) = 0
               AA(2,1) = 1/rho(k-1)
               AA(2,2) = 0
               AA(2,3) = U(k-1)
               AA(2,4) = 0
               AA(3,1) = 0
```

```
    AA(3,2) = 0
    AA(3,3) = U(k-1)
    AA(3,4) = Cp
    AA(4,1) = 1/P(k-1)
    AA(4,2) = -1/rho(k-1)
    AA(4,3) = 0
    AA(4,4) = -1/T(k-1)

    ! inverse the matrix
    call inverse(4, 4, AA, INVA)

    ! Solution vector (RHS)
    BB(1,1) = 0
    BB(2,1) = -(U(k-1)**2)*Cf*c*dx / (2*A)
    BB(3,1) = dq
    BB(4,1) = 0

    DiffMat = matmul(INVA, BB)

    P(k) = P(k-1) + DiffMat(1,1)
    rho(k) = rho(k-1) + DiffMat(2,1)
    U(k) = U(k-1) + DiffMat(3,1)
    T(k) = T(k-1) + DiffMat(4,1)

    if (T(k) < 1) then ! check for negative Temperature
       print *, "Temperature under limit"
   print *, T(k), k*dx
    end if

    ! calculate the new Mach Number
    M(k) = U(k) / sqrt(gam*R*T(k))

if (M(k) > 1) then ! check for choking
       print *, "Flow choked at position:", k*dx
   print *, "Mach Number at position:", M(k)
   print *
   exit
    end if

    ! Total temperature
    Tt(k) = T(k)*(1+(gam-1)*(M(k)**2)/2)

    ! Total Pressure
Pt(k) = P(k)*((1+(gam-1)*(M(k)**2)/2)**(gam /(gam-1)))

! calculate the total drag through the duct
    Drag(k) = A*(rho(k-1)*U(k-1)**2 - rho(1)*U(k)**2 + P(k-1) - P(1))
```

```
        ! Pressure Drop (psi)
        Pdrop(k) = (P(1)-P(k-1))*0.000145037738

        !Reynolds Number
        Re(k) = rho(k-1)*U(k-1)*ID/mu

        ! write each 1000th paramter to a output data file
        j=j+1
        if (j == 1000) then
            write (11,200) (k*dx), U(k), T(k), P(k), M(k), rho(k), Tt(k),
                Pt(k), Drag(k), Pdrop(k), Re(k)
            200 format(g14.7,t20,g14.7,t40,g14.7,t60,g14.7,t80,g14.7,t100,
                g14.7,t120,g14.7, t140,g14.7, t160,g14.7, t180,g14.7,
                t200,g14.7)
            j=0
        end if

end do

! write the exit conditions into the output data file
write (11,300) ((k-1)*dx), U(k-1), T(k-1), P(k-1), M(k-1), rho(k-1), Tt(k-1), Pt(k-1),
    Drag(k-1), Pdrop(k-1), Re(k-1)
300 format(g14.7,t20,g14.7,t40,g14.7,t60,g14.7,t80,g14.7,t100,g14.7, t120,g14.7,
    t140,g14.7, t160,g14.7, t180,g14.7,t200,g14.7)

! Print the variables to the screen
print *, "DROP (psi)", Pdrop(k-1)
print *, "REYNOLDS", Re(k-1)
print *, "PRES", P(1),         P(k-1)
print *, "VEL ", U(1),         U(k-1)
print *, "DENS", rho(1), rho(k-1)
print *, "TEMP", T(1),   T(k-1)
print *, "MACH", M(1),   M(k-1)
print *, "Drag Force:   ", Drag(k-1)
print *, "Friction factor:",f

close(11) ! close file

Contains

subroutine inverse(n, sz, A, AI)
  implicit none
  integer, intent(in) :: n  ! number of equations
  integer, intent(in) :: sz ! dimension of arrays
  double precision, dimension(sz,sz), intent(in) :: A
  double precision, dimension(sz,sz), intent(inout) :: AI

!      PURPOSE : COMPUTE INVERSE WITH REAL COEFFICIENTS  |AI| = |A|^-1
```

```
!
!      INPUT  : THE NUMBER OF ROWS  n
!               THE DIMENSION OF A, sz
!               THE REAL MATRIX  A
!      OUTPUT : THE REAL MATRIX  AI

  integer, dimension(n) :: ROW             ! ROW INTERCHANGE INDICIES
  integer, dimension(n) :: COL             ! COL INTERCHANGE INDICIES
  double precision, dimension(n) :: TEMP   ! INTERCHANGE VECTOR
  integer :: HOLD , I_PIVOT, J_PIVOT       ! PIVOT INDICIES
  double precision :: PIVOT                ! PIVOT ELEMENT VALUE
  double precision :: ABS_PIVOT, NORM1
  integer :: i, j, k

  NORM1 = 0.0D0;
  ! BUILD WORKING DATA STRUCTURE
  do i=1,n
    do j=1,n
      AI(i,j) = A(i,j)
      if( abs(AI(i,j)) > NORM1 ) then
        NORM1 = abs(AI(i,j))
      end if
    end do ! j
  end do ! i
  ! SET UP ROW AND COL  INTERCHANGE VECTORS
  do k=1,n
    ROW(k) = k
    COL(k) = k
  end do ! k

  ! BEGIN MAIN REDUCTION LOOP
  do k=1,n
    ! FIND LARGEST ELEMENT FOR PIVOT
    PIVOT = AI(ROW(k), COL(k))
    I_PIVOT = k
    J_PIVOT = k
    do i=k,n
      do j=k,n
        ABS_PIVOT = abs(PIVOT)
        if( abs(AI(ROW(i), COL(j))) > ABS_PIVOT ) then
          I_PIVOT = i
          J_PIVOT = j
          PIVOT = AI(ROW(i), COL(j))
        end if
      end do ! j
    end do ! i
    ABS_PIVOT = abs(PIVOT)
```

```
    ! HAVE PIVOT, INTERCHANGE ROW, COL POINTERS
    HOLD = ROW(k)
    ROW(k) = ROW(I_PIVOT)
    ROW(I_PIVOT) = HOLD
    HOLD = COL(k)
    COL(k) = COL(J_PIVOT)
    COL(J_PIVOT) = HOLD

    ! CHECK FOR NEAR SINGULAR
    if( ABS_PIVOT < 1.0D-52*NORM1 ) then
      do j=1,n
        AI(ROW(k),j) = 0.0D0
      end do ! j
      do j=1,n
        AI(COL(k),j) = 0.0D0
      end do ! j
      print *, 'redundant row (singular) ', ROW(k)
    else
      ! REDUCE ABOUT PIVOT
      AI(ROW(k), COL(k)) = 1.0 / PIVOT
      do j=1,n
        if( j .ne. k ) then
          AI(ROW(k), COL(j)) = AI(ROW(k), COL(j)) * AI(ROW(k), COL(k))
        end if
      end do ! j
      ! INNER REDUCTION LOOP
      do i=1,n
        if( k .ne. i ) then
          do j=1,n
            if( k .ne. j ) then
              AI(ROW(i), COL(j)) = AI(ROW(i), COL(j)) - &
                                   AI(ROW(i), COL(k)) * AI(ROW(k), COL(j))
            end if
          end do ! j
          AI(ROW(i), COL(k)) = - AI(ROW(i), COL(k)) * AI(ROW(k), COL(k))
        end if
      end do ! i
    end if
    ! FINISHED INNER REDUCTION
  end do ! k
  ! END OF MAIN REDUCTION LOOP

  ! UNSCRAMBLE ROWS
  do j=1,n
    do i=1,n
      TEMP(COL(i)) = AI(ROW(i), j)
    end do ! i
    do i=1,n
```

```
      AI(i,j)= TEMP(i)
    end do !i
  end do ! j
  ! UNSCRAMBLE COLUMNS
  do i=1,n
    do j=1,n
      TEMP(ROW(j)) = AI(i,COL(j))
    end do ! j
    do j=1,n
      AI(i,j)= TEMP(j)
    end do ! j
  end do ! i
end subroutine inverse

End Program newtubes
```

B2. Example Input Code – input.txt

```
1.38811        ! Max Tube length (m)
170300.5       ! Fluid Inlet Pressure (N/m2) absolute
0.0001481      ! Fluid Mass flow rate (kg/s)
0.01           ! Friction Factor
0.001524       ! Tube inner diameter (m)
1.127          ! Fluid Specific heat ratio
81.49          ! Gas Constant(J/kgK) [R134a 81.49, Air 287]
1.167e-5       ! Fluid viscosity (Ns/M^2) [R134a 1.167e-5, Air 1.73e-5]
293.15         ! Fluid Inlet Temperature (K) [20 C]
293.15         ! Tube Wall Temperature
```

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VITA

Carl Reiner Seubert was born in Sydney, Australia on March 1, 1982 to Reiner and Julianne Seubert. He has two younger sisters, Melissa and Cherie. Carl graduated from St. Ives High School in December 2000 and enrolled at the University of Sydney, Australia in March 2001. He graduated with a degree of Bachelor of Engineering Aeronautical (Space) with Honors Class I in November 2004. In August 2005 he enrolled at the University of Missouri – Rolla, U.S.A. He graduated in May 2007 with a Master's of Science degree in Aerospace Engineering. While at the University of Sydney, Carl wrote an undergraduate thesis entitled, *Attitude Determination and Control of the CASSat Cubesat*. He was also an undergraduate teaching assistant and student representative for the school of Aerospace, Mechanical and Mechatronic Engineering and was a member of Mars Society Australia. While at the University of Missouri-Rolla, Carl received the Outstanding Graduate Teaching Assistant Award as well as the Academy of Aerospace and Mechanical Engineers Student Excellence Award. He served as an officer and treasurer of the International Students Club, was a student representative for the council of graduate students and a member of AIAA and the engineering honor society Tau Beta Pi.