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First Level Design And System Design Of Janus Liquid Oxygen-Liquid Methane Lander

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Master's Program in Mechanical Engineering

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FIRST LEVEL DESIGN AND SYSTEM DESIGN OF JANUS LIQUID OXYGEN-LIQUID METHANE LANDER

By

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Abstract

Taking humans to space has always been a fantastic feat, but taking humans to another planet in our solar system is the new goal. Technology has come a long way and with that, so has our knowledge of space and what can be done with current technology to accomplish our goal. These technologies allow for a more efficient space system to transport future astronauts using liquid oxygen and liquid methane (LO_2-LCH_4) as propellants. This combination of propellants, LO_2-LCH_4, brings a variety of benefits. One of the main advantages is that they can be recovered or created from local resources, using in-situ resource utilization (ISRU). This will allow the production of the fuel needed to come back to earth on the surface of Mars, or the space entity being explored, making the overall mission more cost effective by enabling larger usable mass.

At the University of Texas at El Paso (UTEP) MIRO Center for Space Exploration Technology Research (cSETR) in partnership with the National Aeronautics and Space Administration (NASA), research and design of a lander that uses LO_2-LCH_4 is on the move. Janus is a robotic lander vehicle with the capability of vertical take-off and landing (VTOL) which integrates several LO_2-LCH_4 components such as the reaction control engine (RCE).

The following work describes the steps taken to accomplish the design of the first Janus prototype (J-1) which will serve as the learning platform for upcoming prototypes (J-2 and J-3) that will lead to a flight vehicle.

A complete description of the flight profile for the lander will be explained. For this flight profile a MATLAB script was developed to generate plots, which will be used to obtain data. The set of plots developed by the script depicts the flight profile vs time of the lander where height, rotation, velocity, angular velocity, acceleration, angular acceleration, thrust level of the CROME-X engine, thrust level of the RCEs', and the weight of the lander throughout the mission can be

seen. This information was used to determine how much propellant the lander will burn throughout the mission based on the thrust required throughout the mission.

The weight of the propellant required for the mission that was obtained through the script was the placed on the weight budget. The weight budget developed for Janus will be explained in this paper. This weight budget will set a limit on the weight each component has as a limit once each sub-system is complete. This weight limit on each component will ensure a smooth integration to the lander and will keep the lander under a specified weight which will ensure the landers engine (CROME-X) can handle the thrust requirements set by the flight profile. The weight budget will serve for J-2 and J-3 only since there was not weight requirements for J-1 where the testing done will be done on a static thrust stand, therefore no flight oriented equipment was required.

For the static testing (J-1) a set of propellant tanks stands are required to carry the tanks being manufactured. This set of stands will not only carry the tanks, but must also ensure they are safe. A requirements document has been started where a description of the tanks stands operation, interface definition, design loads, failure mode and effects analysis, design requirements and verification criteria. In this paper some of these requirements will be discusses such as the g's of load that the tank stands must be able to withstand in case the stand is dropped or toppled in which case the tank should be unharmed.

A study has begun to define the landers flight configuration, where the goal is to find the best possible way that Janus can be arranged. One of the most crucial components are the tanks and placing them in a certain configuration will affect the dynamics of the lander throughout the mission. This study helps understand how the tanks placement affects and will aid in the decision making of the final orientation.

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Chapter 1: Introduction and Background

1.1 Introduction

Liquid methane (LCH_4) is the new liquid hydrogen (LH_2). NASA and other companies are drifting away from the traditional liquid oxygen-liquid hydrogen propellant due to the many benefits obtained when liquid methane is used as a propellant. These benefits are needed when a mission to Mars is the objective and weight is of critical value.

At the MIRO Center for Space Exploration Technology Research (cSETR) of the University of Texas at El Paso (UTEP), the use for of LO_2-LCH_4 as a propellant for our different projects has been the main research purpose. These projects have been under development for the past recent years for the integration of Janus and Daedalus, which will be used in different scenarios, but will contain identical components as well as variations of the same. These include a LO_2-LCH_4 torch igniter, a 5 lbf reaction control engine (RCE) as well as both a 500 lbf and a 2,000 lbf LO_2-LCH_4 rocket engine.

Janus is a robotic lander that will be able to fly autonomously and have a vertical take-off and landing (VTOL). Daedalus will first be launched using a solid rocket booster provided by NASA then it will serve as a technology demonstrator once it has been placed in sub orbit. Both flight vehicles are being designed at the cSETR and will have two prototypes before each vehicle is completely autonomous, making the design a learning process for everyone involved in the projects. The vehicles will incorporate the same RCE and torch igniter for the first prototypes, but may change as the projects progress. Janus will employ the 2,000 lbf engine and Daedalus the 500 lbf engine as their main propulsion source. Both engines are being developed in house and will be modified if needed for the subsequent prototypes each engine will serve.

This thesis will narrate the development of the Janus robotic lander at its first stage, J-1. All the requirements for the main and sub systems were set at this point as well as the first design concepts which will be shown.

Janus will be designed on campus and the will be tested in UTEP's Technology Research and Innovation Acceleration Park (tRIAc) in Fabens, Texas 33 miles from UTEP's main campus. This area was recently acquired by UTEP in partnership with El Paso County to build three facilities where students will be able to learn and test space and energy research innovation.

1.2 Background

1.2.1 The New Propellant, LO_2-LCH_4

The liquid oxygen-liquid methane combination as a propellant has never been used before in any aerospace mission, but since this kind of methane based propellant has more benefits when compared to hydrogen or kerosene it is worthy of research to adapt or design current and new landers and rockets to use this kind of fuel. Liquid methane is denser than liquid hydrogen and can be stored at a more manageable temperature, which allows for smaller tanks. Since liquid methane and liquid oxygen have around the same cryogenic temperature this means less insulation is required therefore more affordable tanks can be used. Having less insulation and tank weight also allows for more payload to be carried by the rocket. This is particularly important since Mars missions are long and payload capabilities are critical for mission success (Newton, 2017).

Liquid propellants are the newest attraction for lower stage rocket engines. Lower stage rocket engines fire once the solid rocket boosters have been detached. Liquid propellant usually has high-energy output, but for this high output the propellant must be at cryogenic temperature. Cryogenic propellants have one major issue, that if allowed to reach Earth's ambient temperature they would become gasses, which can still be used as propellant but not with the same properties

cryogenics have. Therefore, this type of liquid propellant must remain at an extremely low temperature ranging from -250°F to -500°F to main its liquid state.

Before liquid methane-liquid oxygen was ever given a chance to be a propellant, liquid hydrogen-liquid oxygen (LH_2-LO_2) and refined kerosene-liquid oxygen (LO_2-RP-1) were the propellants of choice for any type of rocket that would require a high thrust output. Even though these propellants have proven to be useful in past missions, they do not serve very good with new mission requirements such as being able to produce the propellant on site or being able to store it safely and easily. LH_2-LO_2 has a very high specific impulse (ISP) of 455 s., which is better than any other rocket propellant ever used, in fact it has been used successfully by numerous rocket engines such as the Space Shuttle Main Engine (SSME) and Saturn V's J-2 engine. This propellant can also be retrieved on site through ISRU, but the problem with this propellant is its temperature. Liquid hydrogen must be stored at -423°F to keep it from evaporating. Rockets fueled with liquid hydrogen need to have a considerable amount of insulation from every possible heat source, such as rocket engine exhaust and air friction during flight through the atmosphere. Insulation should also be placed to protect the tanks from radiation and heat emitted by the sun. Once liquid hydrogen absorbs heat it can quickly evaporate which requires the storage to have some sort of ventilation to prevent an explosion from the increasing pressure in the tank due to evaporation. Another main problem with this type of propellant is that in can leak through any kind of minute pore in welded seams the tank might have when it was welded together (Zona, 2010). In addition, liquid hydrogen causes hydrogen embrittlement, where hydrogen atoms alloy themselves into their metal containers, and so weaken the structure. At high pressures, this can be catastrophic, which is another reason methane has been chosen over hydrogen for Janus. Refined kerosene (RP-1) has a higher density when it is compared to liquid hydrogen (LH_2) therefore requiring smaller tanks. The

liquid oxygen-refined kerosene (LO_2-RP-1) propellant has a density advantage, but has a lower ISP of 358 seconds. It still provides a very high thrust output and unlike hydrogen RP-1 is not a cryogenic fluid which would require tanks to be kept at extremely cold temperatures. This would allow for the propellant to be stored for longer duration of time before evaporation would occur and removes the tank complexity of the system in the vehicle. The problem with RP-1 relies on the fact that it is very difficult to obtain the required materials to produce it on Martian or Lunar soil through ISRU. Unlike RP-1 and LH_2, liquid methane (LCH_4) possesses some advantages and lacks the main disadvantages from the propellants previously described like being able to be obtained through ISRU as well as having a higher density compared to LH2 which allows for smaller and simpler tanks. Even though LCH_4 is a cryogenic propellant it has a much lower boiling point of -259 °F compared to LH_2 at -423 °F. This temperature is relatively close to the boiling point of LO_2 at -297 °F at gives the opportunity to have the same cooling and insulation system in both the propellant (LCH_4) and oxidizer (LO_2) tanks. Although LO_2-LCH_4 does not have the same ISP of LH_2-LO_2 at 455 s., it still has an ISP of 369 seconds. This is larger than LO_2-RP-1 at 358 seconds.

LO_2-LCH_4 was chosen as the propellant for Janus as the main source of propellant not only because of this, but because other components such as the RCE would utilize this same propellant which would enable us to use the same source to power both the 2,000 lbf engine and the RCE as well. This advantage also helps the lander be lighter since only one source of propellant is needed therefore requiring less tanks. If two different kinds of propellant were needed for the different sub systems in the lander more tanks would be required to store the different propellants therefore making the lander heavier.

1.2.2 In-situ Resource Utilization (ISRU)

Methane can be produced from local sources on Mars using in-situ resource utilization (ISRU) technology, it makes it the best option. ISRU will enable the production of the fuel needed to come back to earth on the Martian surface making the overall mission more affordable since only the required propellant to get to Mars will be loaded in the tanks unlike other missions where fuel for the return was needed as well.

The ISRU is based on the recently launched Sabatier system which was originally developed by Nobel Prize-winning French chemist Paul Sabatier during the 1900s. This process uses a catalyst that reacts with carbon dioxide and hydrogen, both byproducts of current life-support systems onboard the space station, to produce water and methane. This interaction closes the loop in the oxygen and water regeneration cycle. In other words, it provides a way to produce water without the need to transport it from Earth (Administrator, 2017). For this process frozen water found in Mars is electrolyzed producing hydrogen and oxygen. The hydrogen obtained would then be combined via Sabatier process with the carbon dioxide found in the atmosphere to produce water and methane. These materials can then be used as propellant and a drinkable water source. The International Space Station (ISS) uses this same principle to obtain these elements. ISS was venting excess carbon dioxide and hydrogen overboard before they sent hardware in 2010 to adapt ISS to this new procedure where they could produce water with those excess chemicals (Administrator, 2017). A brief description of the process is shown in the figure below.

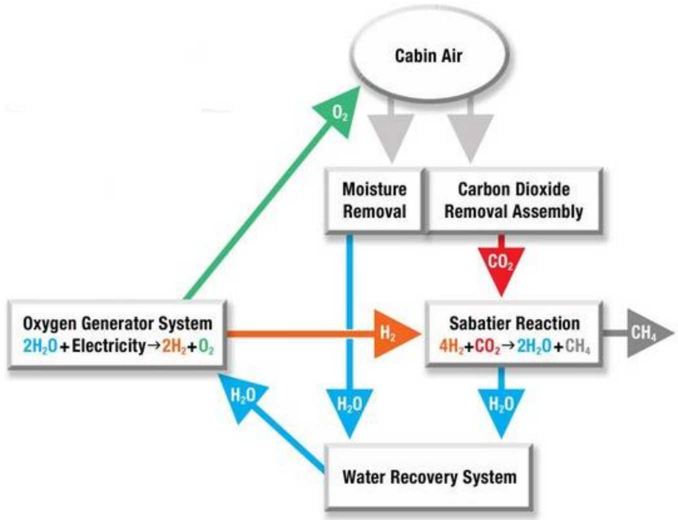

Figure 1: ISS Oxygen and Methane Regen. System

Another great benefit of ISRU is having the capability to not only produce propellant, but to also use existing materials at any planet or asteroid to produce different resources such as a solar cell. Solar cells have long been a suggestion of a kind of resource that can be produced through ISRU. Lunar and Martian soil have silicon, aluminum, and glass, which are three of the primary materials required for solar cell production (Landis, 2007). In fact, the native vacuum on the lunar surface provides an excellent environment for direct vacuum deposition of thin-film materials for solar cells (P.A. Curreri, 2006).

1.2.3 Previous LO2-LCH4 Propellant Engines

1.2.3.1 RS-18 Lunar Ascent Engine

In 2005 NASA Exploration Systems Architecture Study (ESAS) proposed that the crew exploration vehicle (CEV), lunar surface access module (LSAM), ascent stage propulsion and service module that were going to be used in the mission that would take humans back to the moon should be powered by a pressure-fed LO_2-LCH_4. Unfortunately, that mission was later cancelled in 2013, but the research and testing done paved the way for new LO_2-LCH_4 engines to be

developed. The Figure below shows a simple schematic of how the RS-18 engine works, unlike other engines where cooling of the nozzle is done before the engine is fed, this engine is fed directly from the propellant and oxidizer tanks.

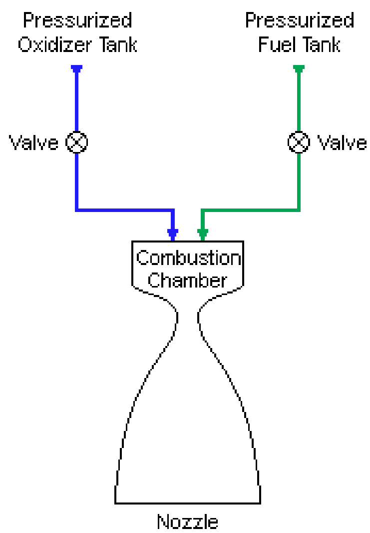

Figure 2: RS-18 Pressure Fed Engine Diagram

The RS-18 was the first engine to be re-structured to fit the necessities of LO_2-LCH_4. It was a NASA agreement with Pratt and Whitney Rocketdyne which enable these changes. Before these changes the RS-18 had been used as the Apollo Lunar Excursion Module (LEM) ascent engine and it used a combination of NTO and a 50/50 blend of Hydrazine and Unsymmetrical Dimethylhydrazine (UDMH). Although the RS-18 was performing less than optimally because it was not originally developed to work on LO_2-LCH_4, it worked. The first advantage of this propellant over monomethyl hydrazine (MMH) and nitrogen tetroxide (NTO) was weight. The LSAM ascent module ended up saving 1,000-2,000 lbm even though it was larger than the current design at the time. Safety and performance was a main requirement for this change to take place where LO_2-LCH_4 landed above the typical hypergolic propellants that had been used in the US

human space program propulsion system (Melcher & Allred, 2009). These propellants are known toxic substances, which not only add risk to handling during ground feeding into tanks but also presented a risk for crew members as the mission progressed. Unlike these propellants in was found that LO_2-LCH_4 had no such toxicity levels and the combustion products were cleaner as well (Melcher & Allred, 2009).

Test stand 401 (TS-401) was chosen to test the RS-18 at simulated altitude conditions. Located at the White Sands Test Facility (WTSF), the test stand had an integration system known as the Large Altitude Simulation System (LASS) that was capable of simulating altitude conditions of around 122,000 ft (37 km) at the engines thrust and flowrates required for testing. For this testing not only was the engine tested, but other sub components like a spark-torch igniter and a pyrotechnic igniter. The test stand allowed for three different igniters to be set at the same time to test for different ignition methods (Melcher & Allred, 2009). These igniters used the same propellant as the engine either in a gaseous or liquid state which could be obtained in a vehicle directly from the main propellant tanks in a lander.

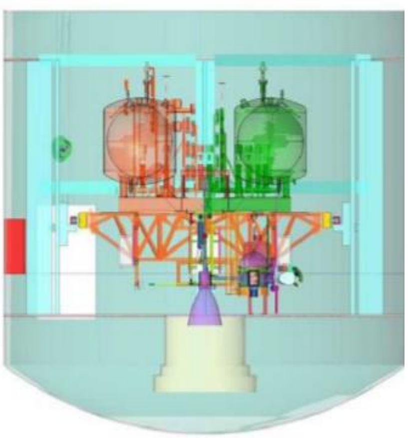

Figure 3: RS-18 on TS-401 CAD

Figure 4: RS-18 on TS-401

The RS-18 had three successful ignition tests at TS-401 under vacuum conditions. The temperature the engine reached could possibly damage engine hardware since the engine was not originally designed for this kind of propellant and tests only lasted for less than 1 second (Melcher & Allred, 2009). The simulated altitude conditions were of 103,000 – 122,000 ft. Although the tests only lasted for less than a second it was enough to obtain steady state measurements, mass flow measurements, ISP and C* efficiency, which is a measure of the energy available from the combustion process (Braeunig, 2012).

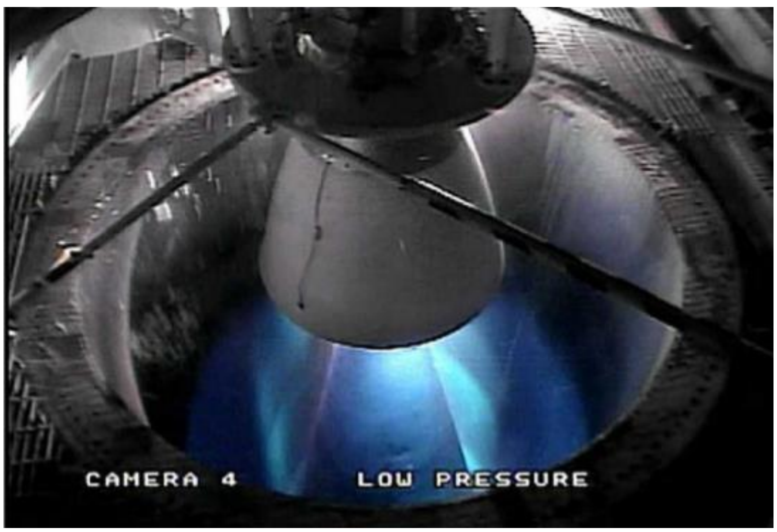

Figure 5: Hot Fire Testing of RS-18 Under Vacuum Conditions

1.2.3.2 Armadillo Aerospace LO2-LCH4 Engine

While the RS-18 was being developed and tested Armadillo Aerospace was also working on their own engine in conjunction with NASA Johnson Space Center (JSC). Their engine was a 1,500 lbf LO_2-LCH_4 engine that would first be tested at the Armadillo facilities in Caddo Mills, TX and would be later tested under conditions like those of the RS-18 at TS-401. This engine was

installed on lander that would serve as its own test bed therefore allowing for static, tethered, and autonomous flight testing. The Armadillo engine was the first in its class to ever complete a free

Figure 6: Armadillo Engine and Lander

flight using LO_2-LCH_4 as propellant utilizing a dual-bell nozzle and a pyrotechnic igniter at altitude. To take less time to change in between engines at the test site it was decided that only the nozzle would be swapped to make the testing easier and cheaper since only one engine would be required. The lander the engine was on was also the first self-pressurized throttling LO_2-LCH_4 propellant lander to take flight. Future testing was conducted at WSTF where the rocket was fired under vacuum and sea level conditions. Ten hot fire tests were conducted, where different scenarios such as altitude and different engine nozzles were tested. The dual-bell nozzle ended up having better results where it obtained an ISP of 133-227 seconds under vacuum conditions for 17 second runs. The testing was run at mixture ratios varying from 1.5 to 2.0 at ambient and simulated altitude as well. Furthermore, the testing proved that LO_2-LCH_4 was a viable source of propellant while using the torch and pyrotechnic igniter under vacuum and sea level conditions.

1.2.3.3 Raptor Engine

The private industry has gained momentum over the past 10 years developing their own engines and coming up with new ideas for new rockets and rocket engines as well. Space X

recently developed a new engine that would be used strictly to support the company's Mars technology development program.

The Raptor engine, as they named it, is a LO_2-LCH_4 engine that started its development in 2009. The engine is to be pressurized utilizing dual turbopumps for the propellant and oxidizer. The original engine was first thought to be a LH_2-LO_2, but recent research and development proved that an interplanetary mission would require the use of LO_2-LCH_4 due to weight restrictions and ISRU, therefore SpaceX modified the engine to fit the needs for this propellant. According to Elon Musk, CEO of SpaceX, the Raptor engine will have and ISP of 334 seconds at sea level and 382 seconds under vacuum conditions. The engine will perform at 4,350 psi chamber pressure and the thrust output will be of 685,000 lbf at sea level and 787,000 lbf under vacuum conditions. The engine uses regenerative cooling where the nozzle has small tubes running through its core where cryogenic propellant is passed through to keep the engine from melting. The engine was test was announced in 2014 and components of the engine were tested at NASA Stennis Space Center (SSC) in Mississippi. Modifications to the E2 test stand had to be done to meet the large propellant and oxidizer flowrates that this engine required. In 2016 the first successful tests for this engine were performed at their main site in McGregor, Texas. The raptor engine will be mounted in a rocket like that of Falcon 9, an existing rocket which has been used lately to transport payload to ISS. According to SpaceX a variation of this engine would be mounted on a lander vehicle oriented towards the colonization of Mars.

Figure 7: Hot-Fire test of Raptor Engine

1.2.3.4 BE-4 Engine

Just like SpaceX, Blue Origin has also been developing new interplanetary technologies and amongst those is the BE-4 LO_2-LCH_4 Engine. Blue Origins main goal is to lower the cost of lower earth orbit travel for civilians to experience what only astronauts have until today. Through this idea they are developing a reusable space vehicle, New Shepard, that uses the BE-3 as its main engine, which is a LO_2-LH_2 propelled engine capable of producing 110,000 lbf of thrust. Along with this idea Blue Origin has also been developing New Glenn, a heavy lift vehicle that could potentially take payload and astronauts to low Earth orbit and be used for interplanetary missions as well. The BE-4 would be the engine used for New Glenn, but it would also be a candidate to form part of United Launch Alliance's (ULA) Vulcan vehicle. The Vulcan is being built around the BE-4 and would cost as much as %40 less than its competitor the AR-1, which is an engine by Rocketdyne that uses kerosene as their main fuel. The BE-4 engine can give 550,000 lbf thrust and is fed liquified natural gas (LNG) which is a natural gas composed mostly of methane. Although the BE-4 engine has just recently been tested in October 19, 2017 there is still no available data than can verify the expected thrust and ISP the engine will output.

Figure 8: Blue Origin's BE-4 Engine

1.2.4 Landers

1.2.4.1 Mars Ascent Vehicle

Taking humans to mars has been the hardest challenge humanity has ever attempted and with new challenges technology evolves. A new lander is being developed by NASA for future interplanetary missions such as mission to Mars. Mission to Mars promises humans will land on Mars by 2020 and their first unmanned launch to be in 2018 using the most powerful rocket ever built, the Space Launch System (SLS).

The Mars Ascent Vehicle (MAV) will be NASA's newest crew lander sin the Apollo. It will be capable of carrying 4 crew members along with approximately 550 lbs of cargo. The MAV will deliver crew members from the SLS down to Martian soil and back, hence its two stages. Stage one will have four 100kN LO_2-LCH_4 engines with an ISP of 360 seconds, while the second stage will be composed of a single identical engine and sixteen 445N RCE thrusters. The second stage will be used for a controlled descent on to Martian soil and will be reused on its ascent back to the SLS thereafter being detached and powering the first stage for the remaining trajectory. NASA had various options for the return from Mars to the SLS vehicle where one was to land an ISRU module that would begin producing propellant before the crew arrived in the MAV and would the re-fuel before leaving Mars. The other option was to land two MAV's one with the crew and payload and another one with a full tank for the ascent back to SLS. Both

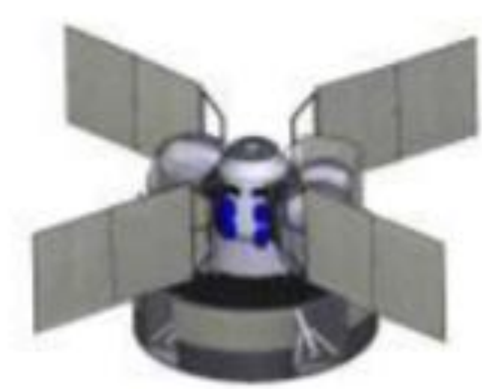

Figure 9: MAV with Deployed Radiators for Propellant Production

options required a 550 lb return cargo to be assumed for return samples and storage containers as well.

1.2.4.2 Morpheus Lander

NASA's Morpheus project was a lander developed to test the capability for unmanned vertical takeoff and land. It was originally designed to serve as a vertical testbed for future advance spacecraft technologies. This testbed would provide and easy access integrated flight system that would reduce the costs for testing. Another main goal for Morpheus was to pave the way for future technologies that haven't been applied to any kind of rocket before such as an LO_2-LCH_4 engine and an automated hazard detection and avoidance technology to reduce possible human error and reduce risk of damage to payload, landers, and fatal accidents as well. Morpheus project and the Autonomous Landing and Hazard Avoidance Technology (ALHAT) provided the technological foundations for the evolution of components needed to move humans beyond low Earth orbit.

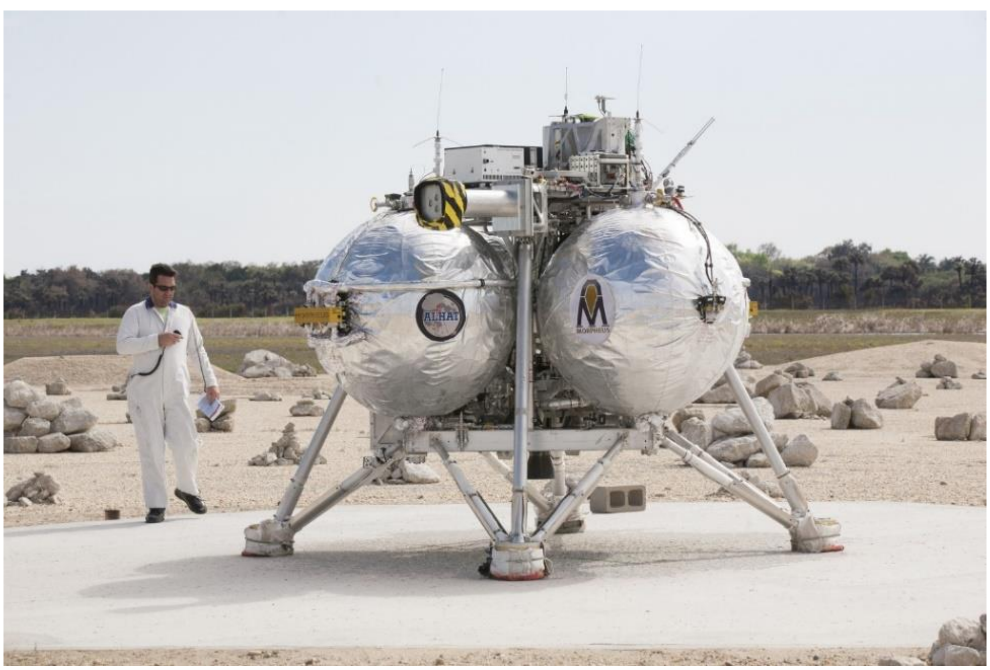

Figure 10: Morpheus Lander

The development of Morpheus's propulsion and RCE's took place at NASA's JSC. The primary objectives for the propulsion system was for it to be an engine with a short development and production to avoid high costs. Morpheus is a 4 tank LO_2-LCH_4 lander that allows the usage of both engine and RCE to be fed from the same tanks. It can carry 2,100 lbs of propellant and the engine can be fed either through a pressure fed or blow down system. Morpheus had different engines, name HD1 to HD5 successively, power the lander. All the engines were to have the same base model, but after testing the first engine iterations were made which drastically changed the engine's model and thrust output.

The first engine, HD1, was designed to have an output of 2,700 lbf of thrust at sea level and had an impinging injector, which injects fuel in a jet like stream that collides with other propellant jet like streams therefore causing some spraying effect where propellant droplets get smaller therefore getting a better engine combustion. The HD1, like all HD models, was a fuel film cooled (FFC) engine. The fuel for cooling is first passed through separate manifold which allows the lander to have a different flowrate to those the engine requires. The engine performed as desired with an ISP of 190 seconds, but damage was done to the main injector and combustion chamber due to a combustion instability detected during testing. The HD2 was the same engine used for HD1 except repairs were done to the injector and combustion chamber. The repairs didn't work out as expected and leakage was found in the injector which led the team to completely abandon the engine. The major design change

Figure 11: HD3 Engine with Tunable Acoustic Configuration

in engine HD3 was its thrust output. The engine was designed to have a higher thrust output of 4,200 lbf. The team learned from the previous tests done in HD1 and included an acoustic cavity to dampen combustion instabilities in the new engine. The acoustic cavity had a variable position acoustic cavity ring as well as a modified combustion chamber to prevent any damage. The cavities worked, and the engine was tested 13 times before a second replica of the engine mas mounted on the lander. For the testing the engine was fired at thrusts ranging from 21-60% of its capacity at Armadillo Aerospace test stand and was later mounted on Morpheus where it was tested at 70-75% of its capacity. Modifications were later done to HD3 and HD4 became the new engine. HD4 was designed to have the same thrust output as HD3 of 4,200 lbf. The engine was flown in multiple Morpheus configurations, the first one of them being the 1.5a. Using this configuration, the first flee flight attempts were demonstrated at Kennedy Space Center (KSC) in 2012. The test ended in a catastrophic result due to navigation and control data loss. For Morpheus model 1.5b a higher thrust output was required hence the new engine HD5. Due to problems with film cooling and unstable ignition, the injector from model 1.5a was salvaged from the wreckage and outfitted with a "large-throat" chamber that made the engine put out a max thrust of 5,400 lbf. The engine was renamed the HD4-A-LT. The engine was used to fly Morpheus 1.5b successfully during 2013 and 2014. A second replica with a large throat configuration was built and named HD4-B-LT. The engine has not been tested as of today.

Figure 12: HD4 Engine Tested at Stennis Space Center

To test Morpheus, it underwent three configurations. The first of them was a static test where the lander was suspended using a crane while being strapped to the ground using chains, restraining the lander from any movement. The test was focused on testing the engine's integration as well as the gimbal system that would direct and correct Morpheus's navigation path. For the second iteration of testing, Morpheus was released from its chains while still being tethered. This test was now focused on testing the landers Guidance and Navigation Control (GNC) system. The test allowed the lander to move freely therefore testing the GNC (Braeunig, 2012)was safe to test. Hovering, rotation and translation tests were conducted without any real risk of crashing the lander. The last test was a fully autonomous flight that took place at KSC. During this test the lander model 1.5a crashed due to a signal loss, therefore leaving the lander "blind". The test ended in a catastrophic crash and explosion hence the model 1.5b and new engine models. Model 1.5b with engine HD4-A-LT conducted the first successful autonomous flight of the smartest lander ever designed. The lander can study terrain before its landing therefore choosing the best place available on a 100 x 100 m. hazard field. The field was a simulation of the type of terrain that could be found on Martian or Lunar soil.

Figure 13: Morpheus model 1.5b During Autonomous Flight

The tests ultimately proved the reliability of the LO_2-LCH_4 propulsion system and paved the way for newer technologies to be adapted on future rockets. Morpheus not only helped engineers at NASA but also helped the cSETR Janus project. Janus resembles project Morpheus

in more than one aspect such as its mission and several components. Engineers at NASA have been aiding throughout the project's planning and development giving their insight and expert opinion as the project progresses.

Chapter 2: Top Level System Specifications of Janus

2.1 Janus Design Configuration

A major conflict with regards to the way the tanks would be set up once Janus was designed is the tank position. The tank position not only alters how the lander will look, but also how it will behave during flight. In previous iterations Janus had been defined as a rocket-like structure where only two propellant tanks would be implemented along with a pressurant tank as well. This decision was taken while talks with NASA had suggested the cSETR would obtain the propellant tanks from previous Morpheus missions, which would allow the center to use the money dedicated for tanks elsewhere. After research was done, it was decided that Morpheus tanks could not be used due to the weight and size of the tanks, which would require an engine with a higher thrust capacity than that of CROME-X rated at a theoretical 2000 lbf. New ideas for a model were brought up and a similar design was chosen which required custom made tanks that would have the required volume to store propellant.

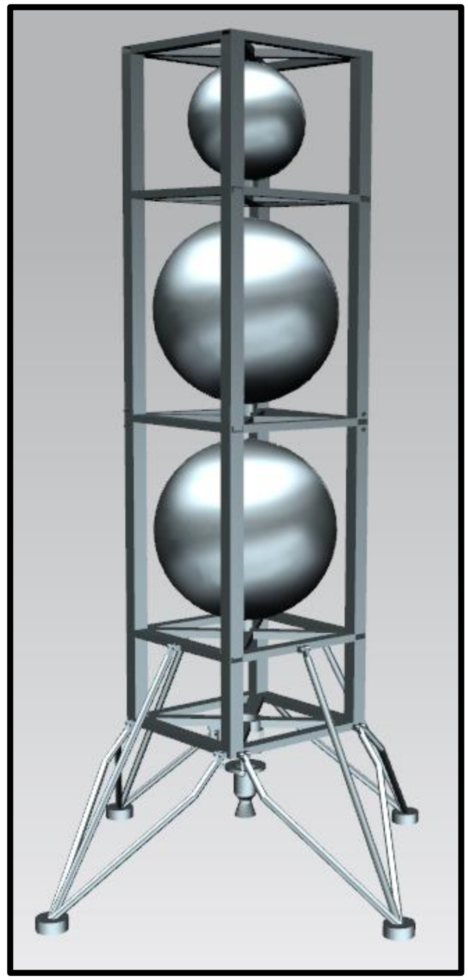

Figure 14: First Janus Lander Design Iteration with Morpheus Tanks

Recently, research was done once again to find the best possible configuration for the tanks which would allow the lander to be maneuverable, cost efficient and light for the CROME-X engine to be utilized. The first iteration of the lander was discarded due to its height, which is shown in Figure 22. The height of the lander made the structure heavy which surpassed the

capabilities of the engine to conduct a successful mission. The height of the lander also implied assembly complications since heavy load cranes would be required to assemble each of the tank modules.

The best option for Janus resulted in a 4-propellant tank configuration, where there will be two liquid oxygen and two liquid methane propellant tanks. All tanks will be on the same plane which will shorten the lander to around 5 ft in height. Since the tanks are at the same plane the lander will be lighter than its predecessor due to a decrease in structural weight. The lander at this configuration has a lower center of gravity the thrust is closer to this point which requires a larger correction in the thrust vector angle to correct the direction of the lander. A disadvantage this design presents is its equilibrium. Unlike the 1st iteration where the lander was theoretically symmetric the new model now has 4 tanks, which weigh the same when empty, but when filled with the required propellant for the mission have a different total weight. This happens because of the different densities LO_2 and LCH_4 have at their cryogenic state where LO_2 is heavier. This weight difference makes the lander have different moments of inertia (MI) in its X and Y axis therefore making the lander unstable during flight. A solution to this problem was found and a study was conducted to assess the feasibility of it.

By placing the tanks at different distances from the Z axis the moments of inertia in the X and Y axis change, therefore by placing the LO_2 tank at a specific distance closer to the Z axis would make the moments of inertia in both X and Y equal (see Figure 15 for lander coordinate system). The first study conducted had the tanks placed at a distance where the lander would be in equilibrium when the tanks where full (beginning of mission). For this study 4 spherical masses were assumed as the propellants and 4 hollow spheres as the tanks while the propellant tank was not included in the study since it was assumed it would be at the center of the lander which would

not influence the moments of inertia. Tanks and propellants were assumed to be concentric. Two of the concentric spheres had a symmetric distance A and the other two a symmetric distance B with respect to the Z-axis (see Figure 15).

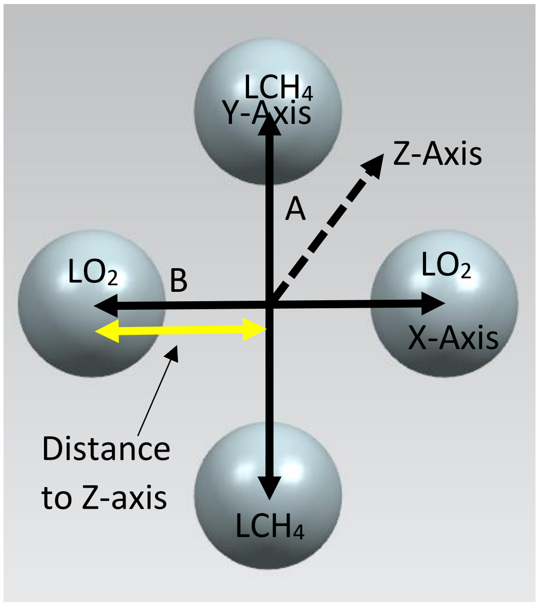

Figure 15: Example Orientation of Tanks

The propellant masses are those of the two propellants the tanks would be carrying at three different stages of the mission which are: full tank (beginning of mission), mid mission and end of mission. The weights of the propellant at each stage are the following.

Liquid Oxygen (lbs): 65, 43.9, 22.7

Liquid Methane (lbs): 34, 23.2, 12

The mass of the tank was assumed to be 27 lbs with a radius of 9.5 in. The distance from the Z axis (see Figure 15) the LO_2 tanks had was of 22 in and that of LCH_4 was of 27.35 in. The following plot shows the moments of inertia at three major points during the mission, where the difference in MIs' gets larger as the mission progresses since more LO_2 is used due to the engines mixture

ratio of 1.89, therefore a large change in its moment of inertia. Equations used for the following studies can be found in the appendix.

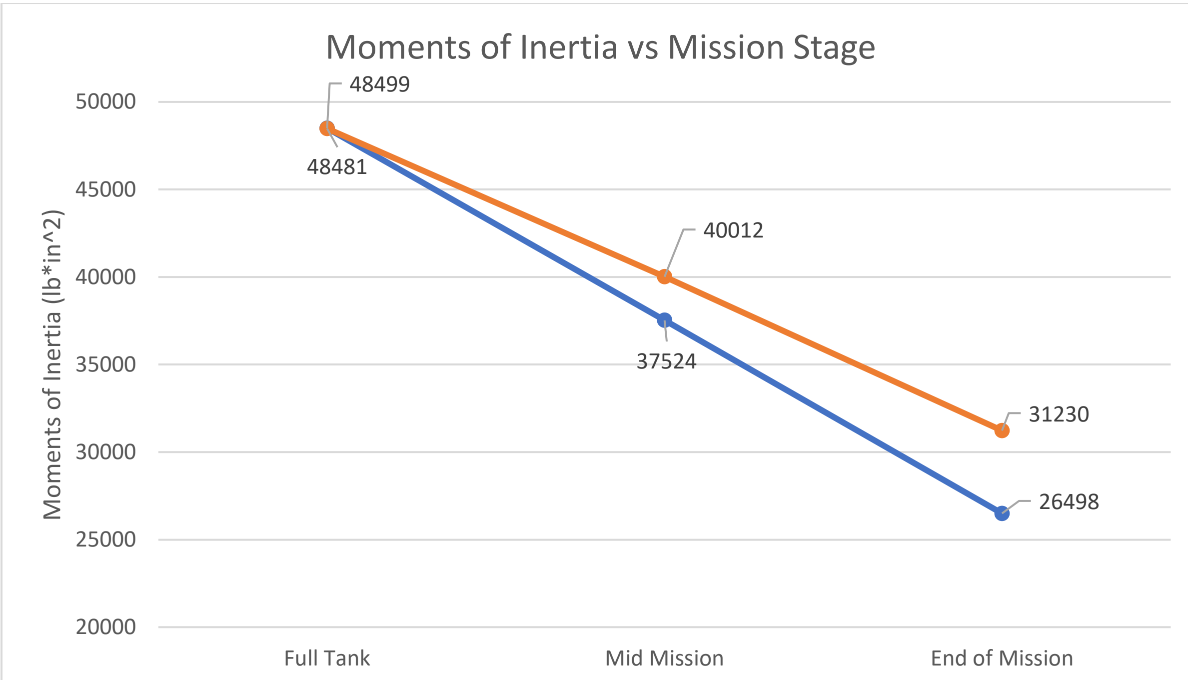

Figure 16: Moments of Inertia vs Mission Stage with Equilibrium at Beginning of Mission

As the figure shows, having an equilibrium at the beginning of the mission is not viable. This equilibrium at the first stage of the mission would increase, making the lander exponentially unstable as the mission progresses which could end in a potential crash if the landers gimbal system is not fast enough at correcting the landers direction for a stable flight. However, another iteration of this same study was done where the separation of the tanks from the Z axis was set so that there was at equilibrium at mid mission when the lander is at its most critical stage, hover and roll. The distance of the LO_2 mass from the Z axis was 22 in. and 26.42 for LCH_4. The rest of the assumptions like tank and propellant mass was kept the same. The following plot depicts the moments of inertia as the mission progresses where an equilibrium point can be seen at mid mission. The initial and final difference is of less than %6 when the MIs’ reach equilibrium at mid

mission compared to %15 difference at the end of the mission if the lander is at equilibrium during the initial stage of the mission with a full tank.

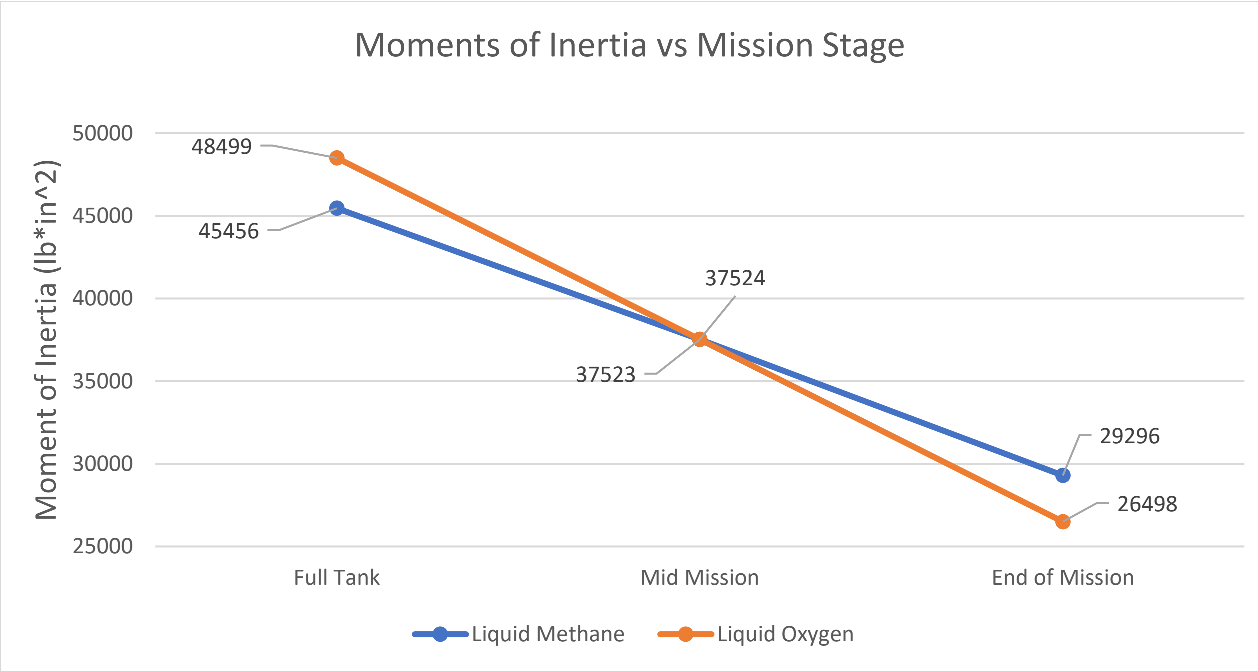

Figure 17:Moments of Inertia vs Mission Stage with Equilibrium at Mid Mission

To prevent having to build a lander chassis that is asymmetric a third iteration was proposed. This iteration had the same configuration as the 4-propellant tank, but instead it used 8 propellant tanks. Using 8 propellant tanks ensured that the lander would have no difference in moments of inertia in its X and Y axis, therefore it was thought to be a good solution. The iteration didn't go forward since having 8 tanks would add too much weight to the lander not only coming from the tanks, but also from the many valves and tubing that would be required for operation. This also elevated the cost for the tanks, even though smaller tanks were required having four of each would not be a cost-effective solution. This study proves that having the lander at equilibrium during flight is possible and a 4-propellant configuration for Janus is the best option even though more in depth study needs to be done to assess the landers moments once the structure and sub-system components are added to the calculations.

2.2 Flight Profile

For the project to begin conceptual designs of each component, first a flight profile had to be established to address the specific requirements each component would have to meet so that it could be integrated in to the lander. The mission that was established for Janus has a duration under 30 seconds during which all the capabilities of the components on the lander could be demonstrated in parallel and play a role during the mission.

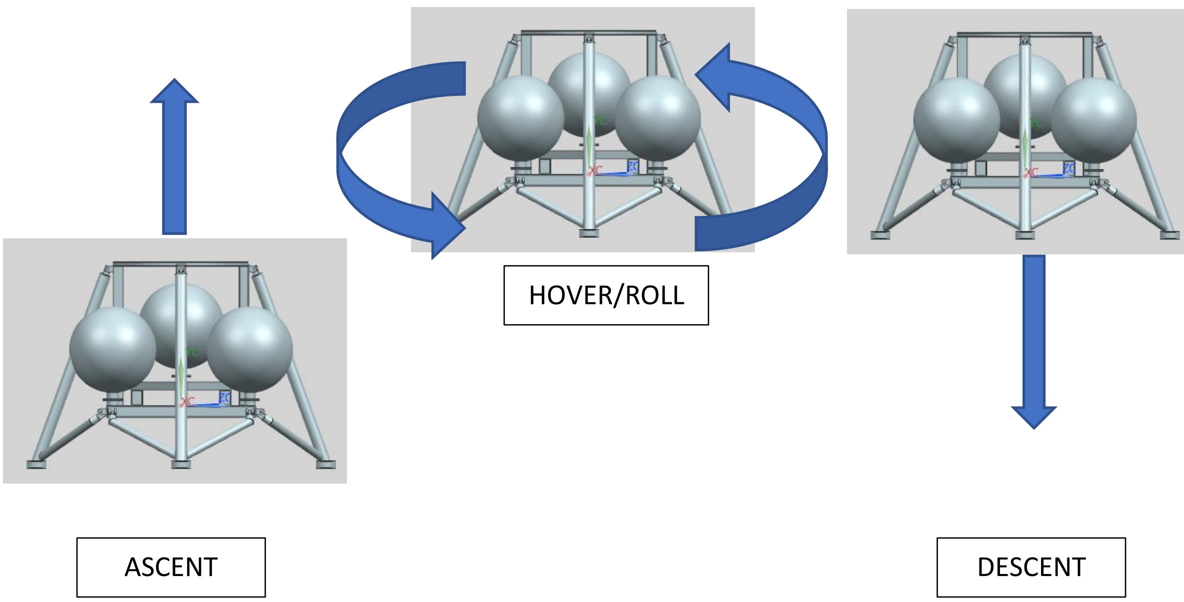

Figure 18: Flight Profile Stages

The vertical translation of the lander will be controlled using the engines throttling capabilities in combination with its gimbal system. The gimbal system will aid in the correction of the thrust vector of the engine to control the pitch and yaw rotational motion of the lander, therefore keeping the lander dynamically stable. Unlike other landers that move horizontally throughout the mission, Janus will only be required to translate vertically and rotate on its Z axis throughout the mission. Janus will be required to perform a vertical ascent of 20 ft where it will hover for 10 seconds and perform a roll maneuver. For the rotational motion during the mission,

RCEs' will be integrated to the lander. A predetermined number of RCEs' will be mounted to the structure of the lander in pairs to accelerate and decelerate the landers rotational motion during the mission. For this mission a total of 8 RCEs' with a thrust output of 5 lbf placed clock and counter clock wise will be used to complete the 360° rotation. Once the rotation is complete the lander will then do a controlled descent and end the mission upon landing.

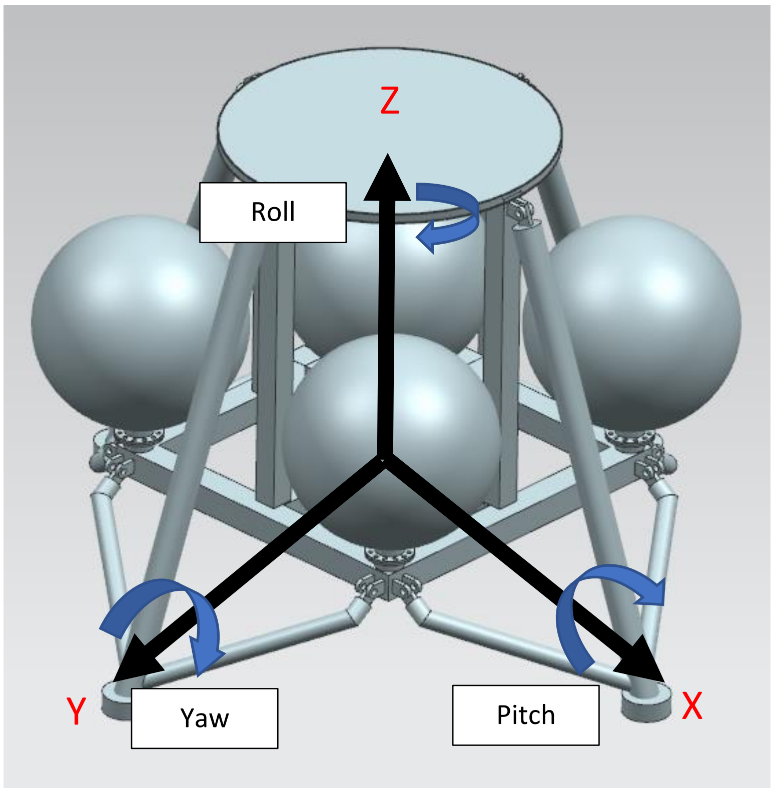

Figure 19: Janus Lander Coordinate System

2.2.1 Ascent Stage

During the ascent stage the CROME-X engine will be ignited and throttled above the official weight of the lander to begin accelerating. As the vehicle accelerates propellant weight goes down, therefore the engine must be continuously throttled down to keep a steady acceleration of the lander. During this stage, and the rest of the stages, the GNC system will be working to

maintain the vehicle in a vertical position by correcting the thrust vector through the CROME-X engine and the gimbal system.

At T= 0 seconds the lander is at the launch pad where its initial height is 0 ft and the landers initial velocity is 0 ft/s. During T=0 the acceleration of the lander is of 1.9 ft/s^2 since the main engine is now at a full thrust of 1600 lbf as shown in Figure 19. From T=3.5-6.5 the lander achieves a height of 20 ft where its velocity went down gradually from 5.7 ft/s to 0 ft/s respectively. During this time the lander had a negative acceleration (-1.8 ft/s^2) done by the throttling the engine under the current weight of the lander to have a smooth approach to the desired height of 20 ft. where the ascent stage ends. At the end of the ascent stage the engine has been throttled to the equivalent of the weight of the lander, therefore is final velocity and acceleration are equal to 0. The following table describes the flight profile at the critical phases during the ascent stage of the mission.

Table 1: Janus Ascent Flight Profile

Flight Time (seconds)	Height (ft)	Velocity (ft/s)	Acceleration (ft/s^2)
0	0	0	1.9
3	8.6	5.71	1.9
3.25	10	5.71	0
3.5	11.4	5.71	-1.9
6.5	20	0	0

2.2.2 Hover/Roll

Once the lander is at its hover stage at a height of 20 ft., the engine will be throttled at exactly the landers weight, this will cause the lander to have no acceleration and hover using the gimbal system and engine to keep the lander dynamically stable in an upright position. During this stage the engine will continue to throttle down as the weight of the propellant keeps decreasing over time to keep a constant height of 20 ft during the roll maneuver. As soon as the lander is

hovering the roll maneuver will begin. For this maneuver, first the RCEs' in the clockwise direction will ignite causing the lander to revolver about its Z axis.

At T=6.5 the ascent stage has ended and a height of 20 ft has been achieved, therefore at this exact time the hovering and roll maneuver begins with the ignition of the pencil thrusters oriented clockwise, which gives the lander an angular acceleration of 22.5 deg/s^2. The pencil thrusters will be shut down after 2 seconds up to T=8.5 during which the lander will have a gradual increase in its angular velocity up to a max of 45 deg/sec and a rotation of 45°. At T=11.5 the lander will have completed 180° of rotation at which the angular velocity has been kept constant at 45 deg/s. At T=14.5 the counter clockwise pencil thrusters will ignite to bring down gradually the angular velocity from 45 deg/s to 0 deg/s. The lander will have a deceleration of -22.5 deg/sec^2 from T=14.5 to 16.5 seconds at which the lander will stop rotating gradually until a full 360° turn has been done at which the final angular velocity and the angular acceleration will now be 0. Throughout this stage the thrust of the engine has been kept equivalent to the weight of the lander to keep a 20 ft height. The following table describes the flight profile at the critical phases during the hover/roll stage of the mission.

Table 2: Janus Hover/Roll Flight Profile

Flight Time (seconds)	Degrees Rotated (°)	Angular Velocity (deg/s)	Angular Acceleration (deg/s^2)
6.5	0	0	22.5
8.5	45	45	0
11.5	180	45	0
14.5	315	45	-22.5
16.5	360	0	0

2.2.3 Descent

Once the lander has completed its roll maneuver and has no angular velocity the descent stage will begin. During this stage the engine will be throttled to a thrust under the weight of the lander to gain acceleration down to the landing site, where once again, it will have to be constantly throttled to keep a constant acceleration since the weight of the lander decreases due to propellant consumption by the engine. Once the lander is close to land the engine will be throttle up again for the lander to decelerate and slow down the descent of the vehicle. The thrust will then be brought down gradually until the lander has landed smoothly and safely on the landing site. The landing gear integrated on the lander will take any loads cause by the landing without having any damage to the main structure. At this point the main engine will be shut down and the mission will have concluded.

At T=16.5 the roll maneuver has concluded, and the descent stage begins where the engine is throttle down for the lander to begin its descent with negative acceleration of -1.9 ft/s^2. This acceleration will be kept constant down to a height of 10.9 ft when the engine is once again throttle up for the lander to decelerate and not crash against the launch pad. At T=19.84 the lander will now be at a velocity of -5.71 ft/s with an acceleration of 1.9 ft/sec^2. This means the lander will continue to fall, but since the engine is throttled above the weight of the lander its velocity will decrease gradually down to -0.25 ft/s. Once the lander approaches this velocity at T=22.7 the lander will be at a height of 1 ft. above the launch pad where it will land at this constant velocity preventing any damage to the structure or landing gear. The following table describes the flight profile at the critical phases during the descent stage of the mission.

Table 3: Janus Descent Flight Profile

Flight Time (seconds)	Height (ft)	Velocity (ft/s)	Acceleration (ft/s^2)
16.5	20	0	-1.9
19.51	10.9	-5.71	-1.9
19.67	10	-5.71	0
19.84	9.1	-5.71	1.9
22.70	1	-0.25	0
26.77	0	-0	0

2.3 Flight Profile Graphs

The next set of graphs at the end of this section depicts the flight profile vs time of the lander where height, rotation, velocity, angular velocity, acceleration, angular acceleration, thrust level of the CROME-X engine, thrust level of the RCEs', and the weight of the lander throughout the mission can be seen. In every graph below the stages of the mission the lander can be appreciated, where from 0 to approximately 6.5 secs is the ascent stage. The hover stage lasts approximately 10 seconds up until 16.5 seconds on the graphs, therefore the rest is the descent stage of the mission. Two of the most essential graphs can be seen in figures 19 and 20, where the thrust is relative to the weight of the lander. As explained before, since the lander decreases in weight, the engine must be throttled to either keep a constant acceleration or hover throughout the mission. The following sections describe each plot.

2.3.1 Figure 16, Displacement

The following plot depicts the height and rotation displacement of Janus throughout the mission. Through the plot it can be seen how the lander gains height exponentially up to 20 ft. at which point the rotation of the lander begins while keeping a steady 20 ft in height. Once the lander has achieved a full rotation the lander now lowers its height down to 1 ft where the lander now makes a soft descent to the touchdown at the landing site.

2.3.2 Figure 17, Velocity

This plot depicts the velocity acquired by the lander as it progresses through the mission as well as its angular velocity as it completes a rotation. Through the plot it can be appreciated how the lander acquires a max velocity of 5.7 ft/sec as well as a max angular velocity of 45 deg/sec. The peaks shown on the plot depict the gain in velocity as the lander approaches 20 ft of height for the ascent. The final velocity can be seen at the descent stage where it has a -0.25 ft/sec velocity at which it makes touchdown at the landing site.

2.3.3 Figure 18, Acceleration

The acceleration the lander acquires as the mission progresses is depicted in this plot. The landers angular acceleration through the roll maneuver can be seen here as well. Through the ascent stage the lander starts off with an acceleration of 1.9 ft/sec^2 at which the main engine is ignited at full thrust to begin the ascent. This constant acceleration last approximately 3 seconds where now a negative acceleration takes place at -1.9 ft/sec^2. During this motion the engine has been throttled to a thrust below the landers actual weight to keep a constant deceleration until the lander reaches 20 ft in height as shown in the plots, there is no acceleration meaning the lander is now at its hover stage. The descent stage depicts the same acceleration as the ascent, except this time there is a negative constant acceleration first, meaning the lander is now descending. As the lander approaches the landing site, the engine is once again throttled to a thrust above the landers weight

to gain a positive acceleration of 1.9 ft/sec^2 causing the lander to decelerate with respect to the landing site until touchdown.

2.3.4 Figure 19, Thrust

As explained before the engine during the ascent stage is throttled above the weight of the lander to gain an acceleration in which case the lander would begin its ascent. The plot depicts a lower thrust as the mission progresses in order to keep this constant acceleration. At the first peak observed in this plot, the engine thrust has been lowered substantially for the lander to begin decelerating as it approaches 20 ft. of height. Once the lander has achieved this height it is now throttled up to the equivalent of the weight of the lander to keep it height of 20 ft. above the landing pad. Throughout the hover stage the thrust is continuously throttled down as the weight of the lander decreases due to the consumption of the propellant by the engine. Once the hover stage has been completed the thrust is brought back down below the landers weight, therefore the lander accelerates back down to the landing site. Once the lander has reached an acceleration of -1.9 ft/sec^2 the thrust of the engine is throttled up to gain a positive acceleration therefore slowing down the descent of the lander and avoiding a crash. This thrust is then brought down as the lander approaches the landing site to have a smooth landing until engine shutdown.

2.3.5 Figure 20, RCS Thrust

Figure 20 shows the thrust output the RCS give combined. The lander will be outfitted with 4 RCEs' pointing clock and counter clockwise which will control the roll maneuver. In this plot at T=6.5 seconds the first set of RCEs' are ignited therefore having a max combined thrust of 16 lbf. This thrust makes the lander rotate at a constant acceleration and at a specified time the counter

clock wise RCEs' are ignited which is why the plot shows a negative thrust. This negative thrust decelerates the lander at a constant rate until it has completed the full rotation.

The MATLAB script used the following equations to develop the plots for the flight profile.

$$M = M_{d+}M_w \tag{1}$$

Where M_d is dry weight and M_w is wet weight, which changes.

$$\dot{M}_w = \propto_v f_v + \propto_r f_r \tag{2}$$

Where $\propto_{v,r}$ are coefficients that relate mass flow rate to thrust

$$(M_d + M_w)\ddot{y} = f_v - (M_d + M_w)g \tag{3}$$

$$\ddot{\theta}(M_d + M_w)\frac{\bar{r}^2}{2} = f_r d \tag{4}$$

Crank-Nicholson method was used integrate numerically equations (2), (3) and (4). Data was obtained using the thrust profile in Table 5.

The MATLAB script used the following equation to find the moment of inertia in the Z-axis throughout the mission as propellant weight decreases. The RCS equations then uses this moment of inertia to find the angular velocity, angular acceleration and time required for the rotation.

$$I = (M_d + M_w)\frac{\bar{r}^2}{2} = (1080 + 1510)\frac{2.5^2}{2} = 8093.75\ lbs \cdot ft^2 \tag{5}$$

Where $\bar{r}$ is the effective radius of the lander to the RCS engines (2.5 feet), therefore the following equation is given. The first result is that of the lander at T= 0 secs which assumes a full tank.

2.3.6 Figure 21, Weight

At the lander progresses through the mission it will be losing weight due to the engine burning propellant, therefore it will have a significant change in weight from beginning to end. This loss of weight is linear since the engine has a fixed MR throughout the mission and propellant is being burned at a constant rate. The initial weight of the lander is at 1510 lbs and the final weight is 1337 therefor 173 lbs of fuel combined (LO_2-LCH_4) will be burned throughout the mission. Throughout the mission the main engine will begin burning fuel at an approximate 7.5 lbs/s combined (LO_2 and LCH_4) and will end up at an approximate of 6.4 lbs/s according to the theoretical data table proportioned by CROME-X team. This table can be found in the appendix.

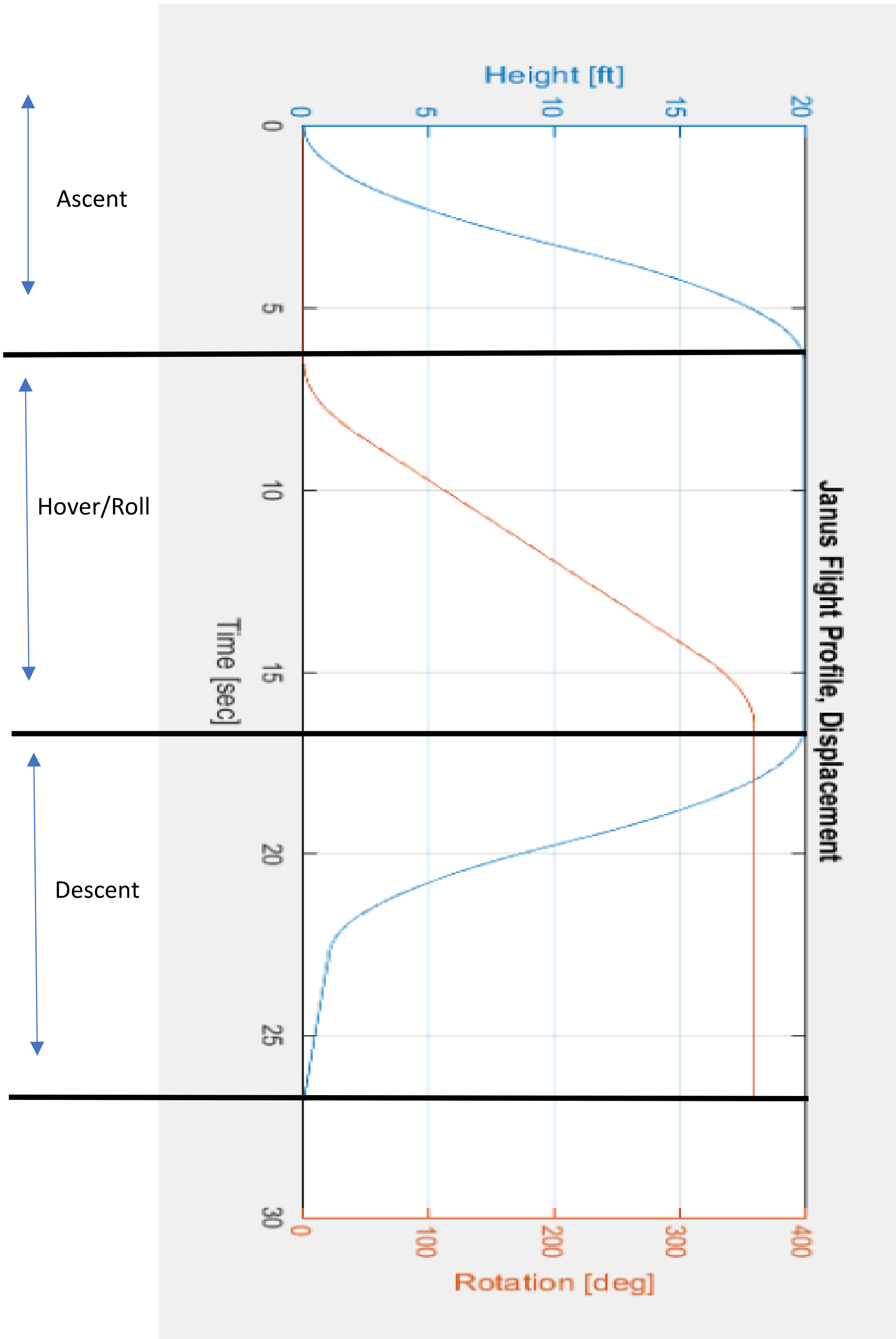

Figure 20: Janus Displacement vs Time

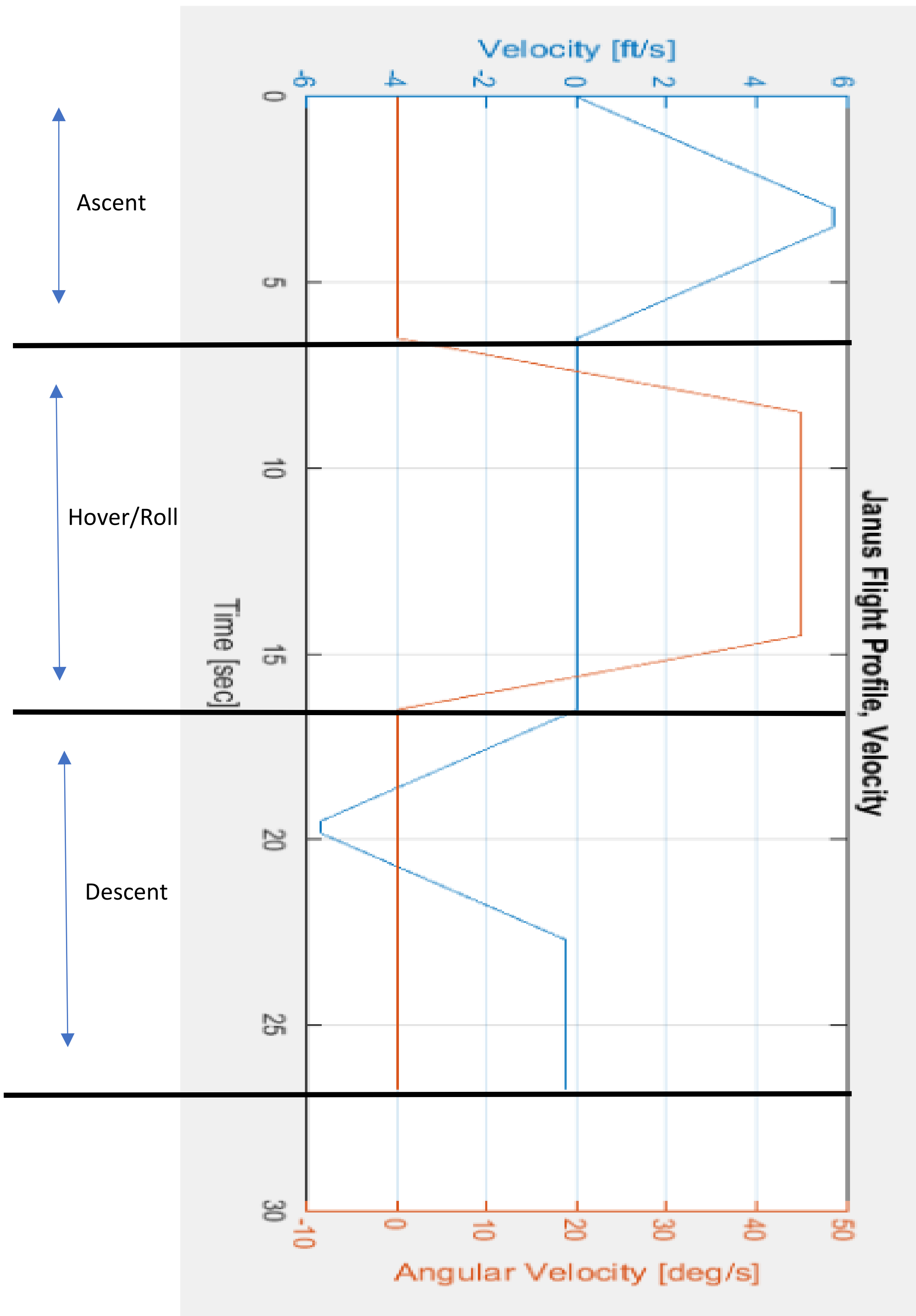

Figure 21: Janus Velocity vs Time

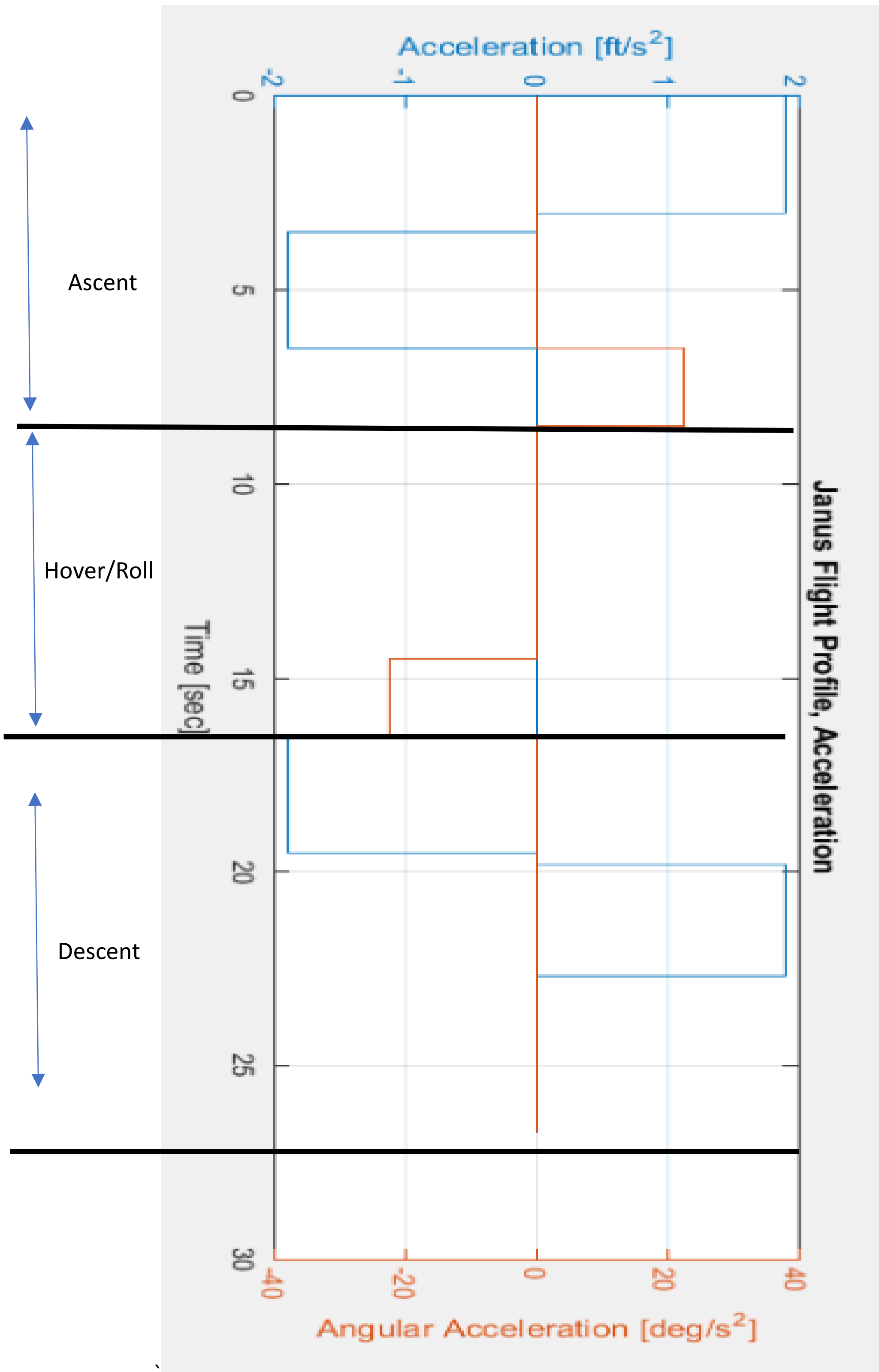

Figure 22: Janus Acceleration vs Time

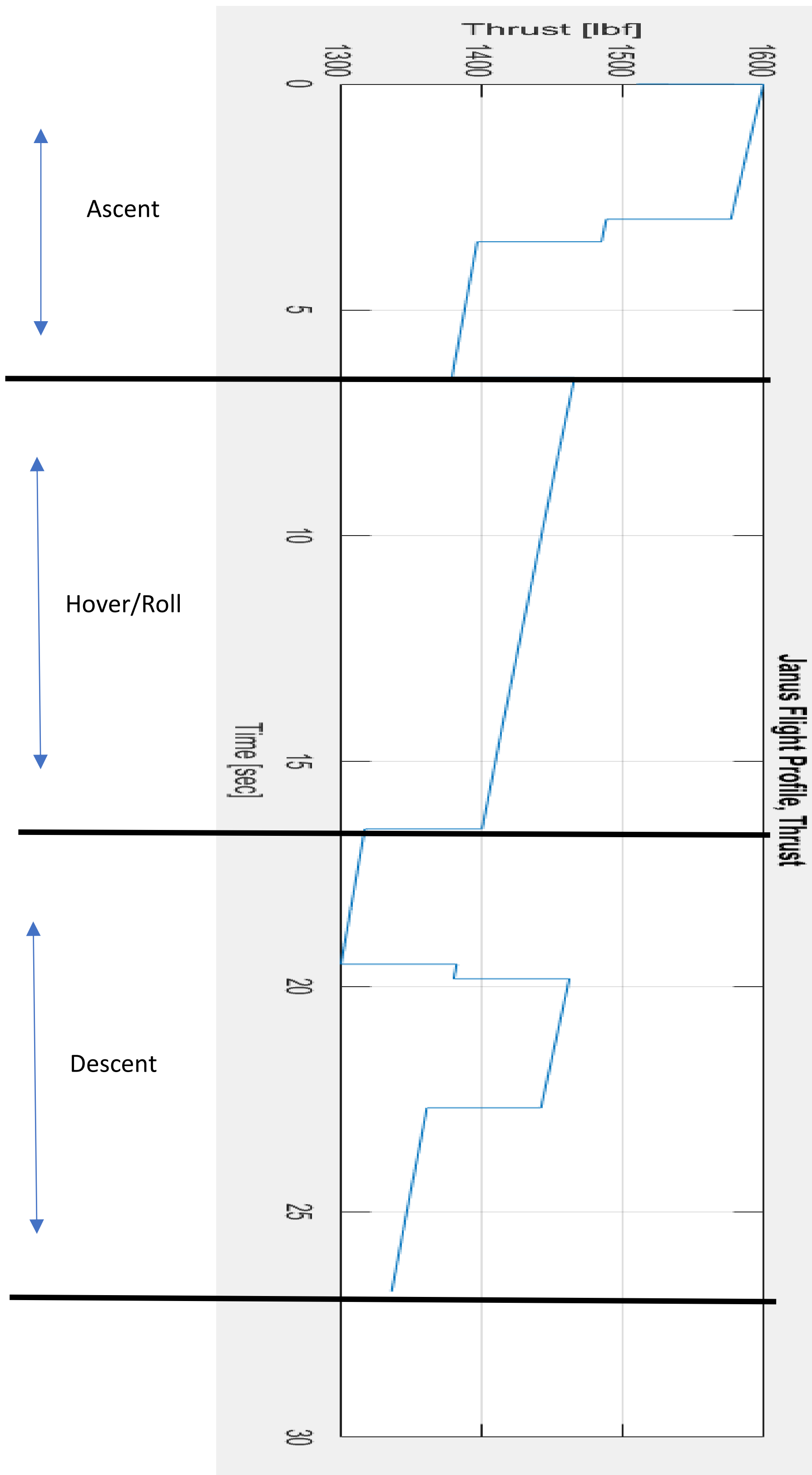

Figure 23: Janus Thrust vs Time

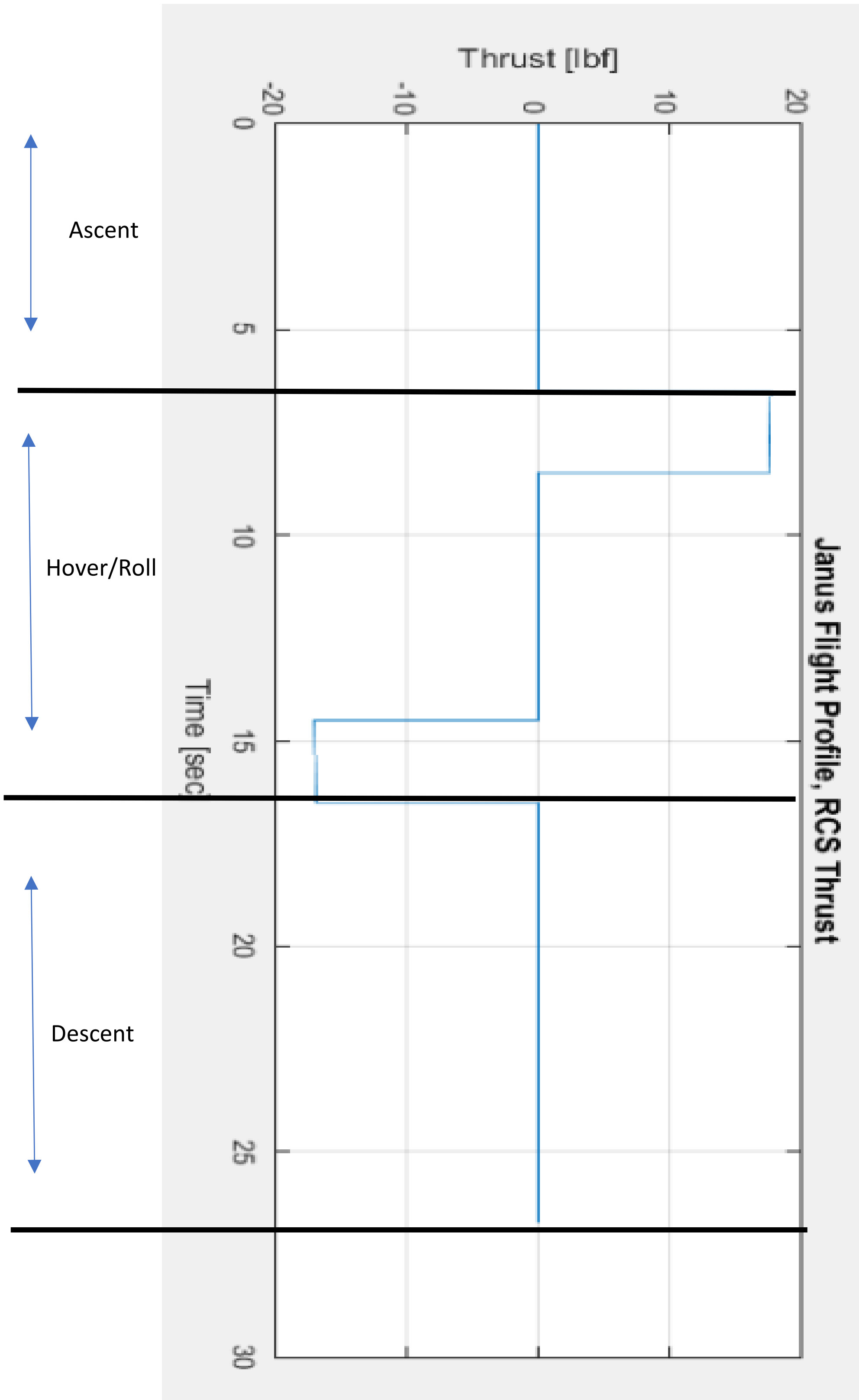

Figure 24: Janus RCS Thrust vs Time

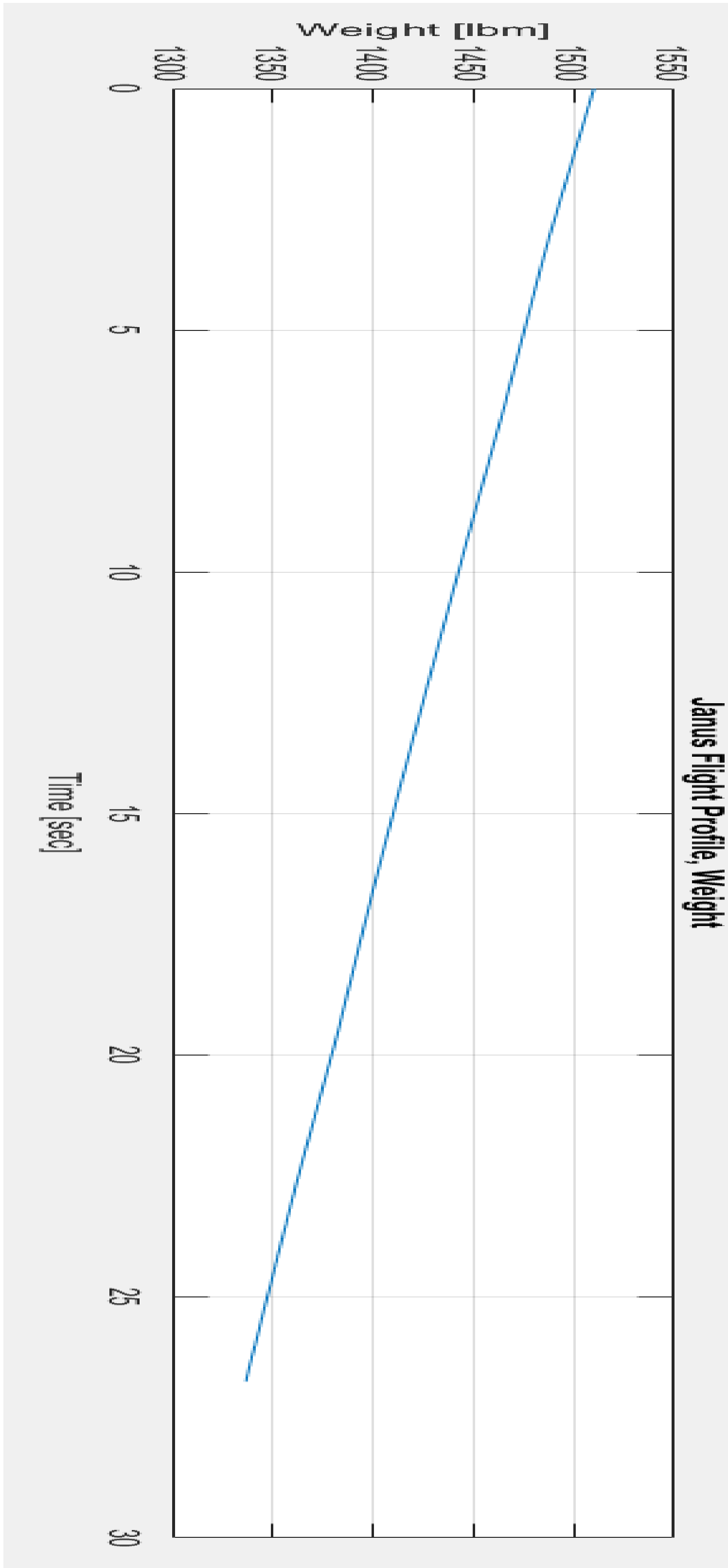

Figure 25: Janus Weight vs Time

2.4 Weight Budget

For Janus to have a successful mission all its components need to comply with the weight requirements established. This weight budget will set a limit on the weight each component has as a limit once each sub-system is complete. This weight limit on each component will ensure a smooth integration to the lander and will keep the lander under a specified weight of 1500 lbs. This weight budget will prevent the lander from being "overweight" in which case the CROME-X, would no longer be viable for the lander therefore requiring a more powerful engine.

Table 4: Janus Weight Budget

System	Weight (lbs)	QTY	Total (lbs)
Propellant LOX and CH4	430	1	430
Helium	40	1	40
Propellant Tanks (4)	40	4	160
Helium tank	20	1	20
Payload	300	1	300
Engine	50	1	50
RCS (8)	5	8	40
Gimbal	50	1	50
Piping and valves	150	1	150
Power	25	1	25
Electronics	25	1	25
Landing Gear	50	1	50
Structure	120	1	110
Miscellaneous	50	1	50
Total			1500

Some of the weight budget estimates were considered based on actual weight of the components being devised in house such as the CROME-X engine which currently weighs in at about 95 lbs. and will serve only for J-1 testing since the engine contains features which are not

required for J-2 or J-3, therefore the weight of the engine can be decreased significantly. Other estimates such as the payload, piping and miscellaneous are not fixed and might change over time if and only if required.

The following is a description of the components in the weight budget.

Propellant LOX and CH4: These fixed weights are an iteration calculated based of the required propellant of the engine for the duration of 2 missions in combination with the total weight of the lander.

Helium: The pressurant weight is calculated based on the volume required to fill the propellant tanks while keeping a constant pressure of about 320 psi once the propellant has been depleted. A separate excel sheet is used to calculate this weight.

Propellant Tanks: The weight of the propellant tanks is placed here. All 4 propellant tanks are identical excluding the Helium tank.

Helium Tank: Helium is stored here; this tank can also be called a pressurant tank.

Payload: the part of a vehicle's load, from which revenue is derived; passengers and cargo.

Engine: Main source of thrust of the lander, which will be throttled to control vertical translation of the vehicle.

Propellant LOX and CH4: These fixed weights are an iteration calculated based of the required propellant of the engine for the duration of 2 missions in combination with the total weight of the lander.

Helium: The pressurant weight is calculated based on the volume required to fill the propellant tanks while keeping a constant pressure of about 320 psi once the propellant has been depleted. A separate excel sheet is used to calculate this weight.

Propellant Tanks: The weight of the propellant tanks is placed here. All 4 propellant tanks are identical excluding the Helium tank.

Helium Tank: Helium is stored here; this tank can also be called a pressurant tank.

Payload: the part of a vehicle's load, from which revenue is derived; passengers and cargo.

Engine: Main source of thrust of the lander, which will be throttled to control vertical translation of the vehicle.

To obtain 1500 lbs as the max weight Janus could have the following equation was used.

$$ma = \propto T - W \tag{6}$$

Where m= mass, a= acceleration, $\propto$=assumed efficiency of CROME-X, T=Theoretical CROME-X max thrust and W= Total weight of Janus. Mass is then changed to weight/gravity and the remaining equation will be the following, where g= gravity.

$$\frac{W}{g} a = \propto T - W \tag{7}$$

After rearranging the equation to obtain W the final equation then becomes the following, where the obtained result is the max total weight of the lander.

$$W = \frac{\propto T}{1+\frac{a}{g}} = \frac{.8(2000)}{1+\frac{2}{32.2}} = 1506\ lbs \tag{8}$$

The mas weight allowed for the lander is of 1506 lbs. and was rounded to 1500 lbs.

Chapter 3: Project Janus

3.1 Project Overview

Since it's opening, the cSETR's main goal has been to promote the research and education in propulsion and energy engineering. In the past years the center, just like NASA and other companies, has dedicated research toward the study of LO_2-LCH_4 propellant based propulsion systems. One of cSETR's main objectives is Janus, where many of the sub-systems under development will be integrated to make the lander.

Janus has been under development since August 2015 when the cSETR was awarded NASA's Minority University Research and Education Project (MUREP) Institutional Research Opportunity which makes up the MIRO grant. Janus will be an autonomous lander that will have the capability to perform a vertical take-off and landing (VTOL). The sub-systems that will be integrated to the lander consist of: shear-coaxial torch igniter for the CROME-X (2000 lbf) engine and a set of pencil thrusters (5 lbf) that will act as the RCE to roll the lander. The lander will also incorporate a gimbal system which will control the landers pitch and yaw through the mission.

The project will be separated in to three different phases, J-1, J-2 and J-3. The different phases will simplify the implementation of the sub-systems to the lander since each prototype will incorporate newer versions of each sub-system until it is completely autonomous at J-3. The project was separated in to these phases to facilitate short term goals where research can be done in shorter periods of time with successful results. This was a result of the high hire and leave rate that the cSETR experiences. This happens since new graduate students at the cSETR are in the research projects for a short amount of time before they graduate and having them focus on short term goals has a better result and gives students motivation.

For Janus to have an organized design process first a flight profile was established. This flight profile will serve as the base guide for the requirements each component needs to accomplish for it to be used during J-2 and J-3 stages. The flight profile selected was

3.3 Project Phases (J-1, J-2, J-3)

For Janus to be completed it was separated in to three different stages which are: J-1, J-2 and J-3. J-1 will be in a static tested bed configuration where combustion components will be tested at ground level. J-2 will have a flight oriented lander configuration where all combustion components and guidance and navigation control will be tested while the lander is tethered to a crane to avoid any kind of damage to the lander and its components. J-3 will be the final phase of the project which will test Janus's ability to take off, fly and land autonomously.

The separation of Janus in to three stages was done for many reasons like the ability to have short term goals that could be achieved by students at the cSETR throughout their Master's or PHD program. Janus is a complex lander that will not require certain components to be developed early in the project timeline, therefore this separation aims for the cSETR to focus on projects that are crucial for the first stage and as projects are completed more flight oriented projects will be given to incoming students.

3.2.1 J-1

J-1 will be the first project phase. This phase does not require Janus to look like a flight vehicle and more like a static testbed on which the performance of the propulsion systems will be assessed and critical information such as the amount of thrust the engines are capable of will be obtained from. This phase will also serve as a training ground to develop two different test stands, a horizontal and a vertical one as well.

The J-1 phase will include the CROME-X, CROME and Reaction Control Engines including all the instrumentation and feed lines required to test each individual component. The tanks and tank stands will not be flight hardware for this phase and will be the opposite. These components will be larger, heavier, and robust versions of the same to reduce risk of mishandling or performance damage. The tanks and engine will not have a flight configuration assembly, where the tanks are close to the engine, and on the opposite, will have separate tank stands for the LO_2 and LCH_4 which will be placed at a safe distance apart from the engine on opposite ends to prevent an explosion in case of leak.

J-1 will lack some of the subsystems that J-2 and J-3 will require for flight such as the gimbal system, which changes the thrust vector of the engine to correct the lander's path. The landing gear will not be designed at this stage either. Since this phase requires a static thrust stand there is no point in designing landing gear so early in the project. The Guidance and Navigation Controls (GNC) will also be developed at J-2 phase of the project for the same reason as the landing gear.

During testing for the CROME-X and CROME engines the following is a list of the critical values that are to be obtained:

-Engine efficiency (thrust output)

-Specific Impulse (ISP)

-Throttle response time

-Dampening instabilities

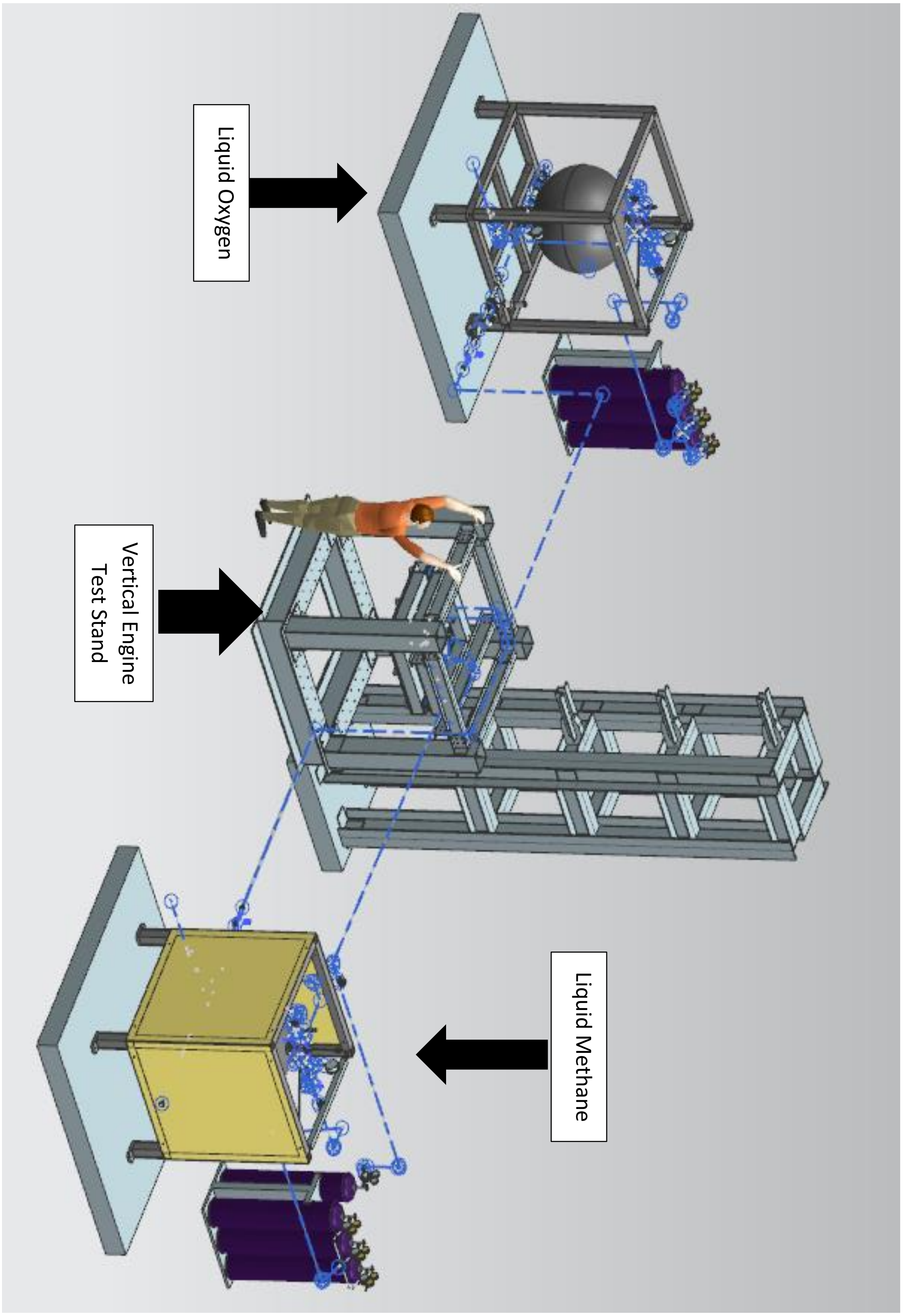

Figure 26: Example J-1 Set-up

To obtain the thrust values of each engine a load cell module is being devised in house. This module is a thrust measuring system that can be detached from the static testbed, so the engines can be tested with or without it.

The next step would be to test all components at once where a flight profile simulation can be done to see how each component behaves under these conditions. At this point a flight sequence should be done to comply with the stated flight profile in which each component will be fired at a specific time and thrust to accomplish a successful flight during J-3.

Due to the similar components both Janus and Daedalus have, many of the testing elements such as the static testbed and propellant tanks will be shared and have a similar testing sequence, which will prevent manufacturing of different test stands and other components for both teams, therefore reducing costs required for testing.

3.2.2 J-2

The J-2 phase of the project is intended to introduce and integrate flight components that will be used on Janus. This integration will now require a lander design in which all systems will be placed in a flight oriented manner. This phase will resemble what NASA did during their testing of Morpheus, in which the lander is attached to a crane with its landing gear attached to the ground. This type of testing allows the GNC and main engine to be tested without the risk of crashing and damaging the lander with a potential total loss.

During this phase many of the flight systems will be designed and integrated to the lander. An example of these is the propellant tanks, the tanks used for J-1 were heavy and not flight oriented since they were designed to be larger and heavier to have more propellant available for testing without the need to refuel so often. The structure of the lander will also be much lighter

than the once used in the static testbed, possibly so will the feed lines and all the other components use for the operation of the engines and RCE's such as valves. The most noticeable implementation however, will be the gimbal system. The power source has not been discussed in much depth, but unlike J-1 were a similar version of a car battery will be used, J-2 will require a much lighter power source which can be obtained through a SOFC.

Figure 27: Tethered and Ground Restrained Morpheus

Figure 28: Tethered Morpheus

J-2 will have two sub-phases where the integration of the gimbal defines these. First the lander will be tethered and restrained to the ground using chains. An example representation of this can be seen in figure 16, where project Morpheus tested in the same fashion. During this phase the engine and RCE's will be tested. At the same time, the gimbal system will undergo a series of tests to evaluate its response time for a thrust vector change. Once the gimbal system has been debugged, the second sub-phase will come in to effect. This phase will be when the chains

constraining the lander to the ground are removed and the lander is now able to hoover and rotate without the risk of crashing since it will still be tethered to a crane. An example of this phase can be seen in figure 17. During this phase the GNC will be tested in a short-range motion where the ability of the lander to take off, hoover, rotate and land will be tested. For the landing testing the main engine will not be required to be ignited since the crane can perform a controlled “drop” of the lander where the teams can evaluate the performance of the landing gear. During flight, if the GNC or any other component fail the main engine can be shut down and the lander will hang from the crane to avoid a crash. Testing at this phase will give the teams opportunity to correct any problems encountered without the risk of damage to the lander’s components.

The following is a list of things the team will have accomplished by the end of J-2:

-Flight-ready power source

-Telemetry systems for lander

-GNC development and integration

-Design and integration of landing gear

-Gimbal system design and integration

-First controlled flight of the lander

3.2.3 J-3

The last phase of Janus is J-3. During this phase Janus will perform free flight and will no longer be tethered to a crane or restrained in any other way. Janus will be tested for its capability to take off, hoover, rotate, and land autonomously.

The lander will have the same flight configuration as J-2 where all its components are flight-ready and only the required propellant for the mission's flight profile will be loaded. J-3 will progressively be tested at different heights until the planned flight profile established at the beginning of the project is completed. Upon completion of a successful flight mission, JANUS will demonstrate the reliability and performance capabilities of an integrated liquid oxygen/liquid methane propulsion system and autonomous control.

3.3 Sub-System Components and Technologies

The lander will have different technologies that have all been designed at the cSETR by students over the course of the years. These different technologies would all serve a different purpose and an effort to make all of them utilize the same propellant has been done. Using the same propellant source will facilitate the design process and will make J-1 serve as a test bed as well. Since all the projects will use the same propellant the tanks manufactured for J-1 will help the projects test their components individually before J-2 starts without the need of all the other sub-system components unlike J-3 where all system will need to be working for it to have a successful flight.

Even though the different components are all fed from the same propellant tanks, they all have different design requirements and operating conditions. One of these operating conditions is the rate and pressure at which propellant is being fed to each. Since they all require different rates, the vehicle must be able to accommodate to it, therefore a pressure fed system was selected for this lander. It will give the lander the ability to have a constant pressured feed system where components can be outfitted with regulators that will deliver every component its specified needs.

The lander requires a reliable source of power to keep all components working and this presents a challenge too. Having traditional batteries on J-3 would bring weight up which is critical for mission success and should be kept at its minimum when possible, therefore an alternate solution has become the use of a methane solid oxide fuel cell (SOFC). This kind of power cell is power generator that converts power using a chemical reaction. Unlike conventional batteries the cell provides power using the propellant carried on the lander. Just like all the other sub-system components the cell would use LO_2-LCH_4 obtained from the lander to create a chemical reaction that would create enough power for the landers components. No research up to this day has been done to evaluate the use of this kind of power source, but it is in the scope of the project once J-1 has been completed and flight hardware for J-2 and J-3 is the next objective.

Janus will not only incorporate components that have been done traditionally through manufacturing or stock purchase, but will also look in to the possibility to incorporate 3D printing wherever it serves a better purpose that traditional manufacturing. Printing in this fashion gives the opportunity to build complex shapes that would be too costly, take a long time to manufacture or even be impossible to manufacture the traditional way. Incorporating this kind of rapid prototyping would require extended research but will surely provide students with knowledge in the feasibility of incorporating this kind of materials on J-2 or J-3.

The different sub-system components will be changing over time as Janus steps in to its new phases over the years, where every new phase will bring different sub-systems and updated versions of previous ones as well. The sub systems will be described, and detail of their role in the lander will be explained as well.

3.3.1 Torch Igniter

Once of the first components designed at the cSETR for Janus and Daedalus was the shear coaxial torch igniter for CROME and CROME-X engines. Since Janus's main propellant is LO_2-LCH_4, the torch igniter was designed to use this combination as its propellant source. This was done to avoid having different propellants on board. The igniter's purpose is to ignite using the same propellant as the lander's main engine using a modified car spark plug. The igniter will be placed at a specified position inside the CROME-X engine and once fired its flame will ignite the propellant coming inside the engine's combustion chamber therefore having a successful engine ignition. The igniter is composed of few parts which makes it easy to replace in case of damage. It has a shear coaxial injector to feed propellant in to its combustion chamber. The oxygen is injected through the back of the igniter in to the combustion chamber unlike the methane which is introduced through a manifold. Once in the manifold there is four tangential holes that go in to the combustion chamber thus creating a swirling motion and mixing the propellant and oxidizer for a stable combustion. The igniter has a modified car spark plug that was adapted to fit on the igniter and deliver the initial spark for combustion to take place. A pressure transducer (PT) port has been welded to the combustion chamber area of the igniter. This PT will show the combustion chambers pressure which can be used to verify ignition success. The igniter has a solenoid valve for each propellant line to control the igniters propellant feed

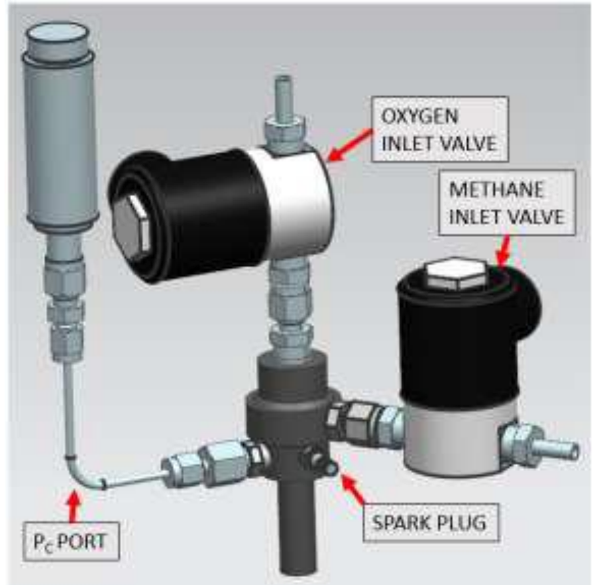

Figure 29: Shear Coaxial Torch Igniter Assembly CAD

timing. The oxidizer valve opens before the propellant line to achieve a successful ignition therefore requiring one valve for each.

Once the igniter was manufactured it was hot fire tested several times since spring 2016 for almost a year. The igniter was originally designed to be used in any variation of propellant state

Figure 30:Shear Coaxial Torch Igniter

(e.g. liquid methane-liquid oxygen or liquid methane-gaseous oxygen) and test using all four possible combinations of propellant where hot fire tested. Gas-gas combination had a 100% ignition rate and proved to be the most reliable way to ignite the engine unlike other propellant states that had a less optimal ignition success rate. Since the igniter will ignite at gas-gas propellant state a method to convert liquid propellant in the lander tanks to a gaseous state has to be devised before propellant is fed to the igniter.

3.3.2 Reaction Control Engine (RCE)

The reaction control engine (RCE) is another main component for Janus and Daedalus designed at the cSETR. The engine is also known as a pencil thruster due to its appearance that

resembles a metallic pencil. This type of engine has a low thrust capability compared to the main engine being used and are sometimes used to correct rocket or lander direction in combination or without a gimbal system. In the case of Janus, the RCE's will only be used to perform a roll maneuver during its hoovering stage of the mission.

The initial design of the pencil thruster was given to the cSETR by NASA. NASA had previously used this design, which was a converted version of a torch igniter they had designed for an Aerojet engine. The pencil thruster was then modified at the cSETR where one of its noticeable changes was its combustion chamber length and nozzle shape. The modifications where done for the thruster to out-put 5 lbf of thrust and for them to be able to use LO_2-LCH_4 propellant. Like the torch igniter, the thruster kept the PT at the combustion chamber to monitor the pressure and verify a successful ignition.

The RCE uses a modified car spark plug that has an extended electrode that fits inside the combustion chamber and once current is passed through it the electrode closes the circuit with the inside wall of the combustion chamber causing a spark. Unlike the torch igniter, the RCE has three manifolds where the fuel film cooling, oxidizer, and propellant pass through. The propellant passes through the manifolds and is then mixed inside the combustion chamber and the initial spark ignites the mix.

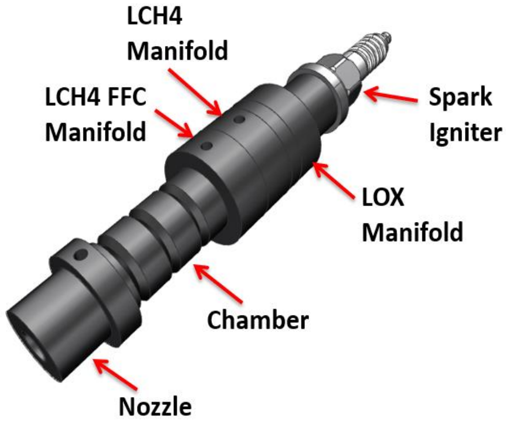

Figure 31: RCE Component Description

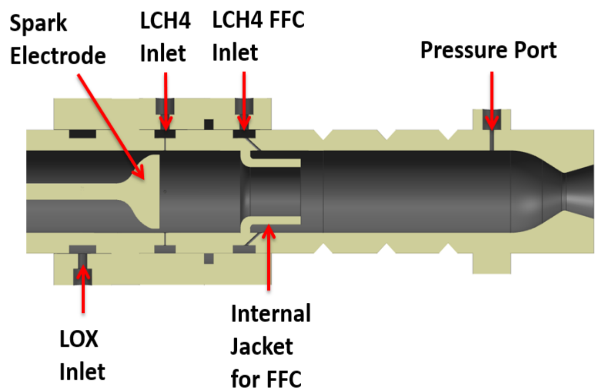

Figure 32: REC Cross Section and Component Description

The RCE will play a crucial role for mission success in Janus. Part of the mission is for the lander to complete a full rotation; therefore, it is imperative that the RCE's are working flawlessly to achieve the roll at the specified time and thrust. The thrusters will be set in pairs at a given distance on the outer part of the lander facing in opposite directions (clockwise and counter clock wise). The thrusters will then fire creating a coupled moment, rotating the lander on its Z axis (ground to top of lander is considered Z axis) afterwards, the thrusters in the opposite direction will fire to decelerate the landers rotation until a full 360° turn is achieved. The equation of angular motion:

$$t = \sqrt{\frac{2\theta}{\alpha}} \tag{9}$$

Where t=time, θ=final position, α=angular acceleration. The equation is used at constant acceleration to determine the time required to make a 180° turn. The result will then be doubled to obtain the total time for a 360°

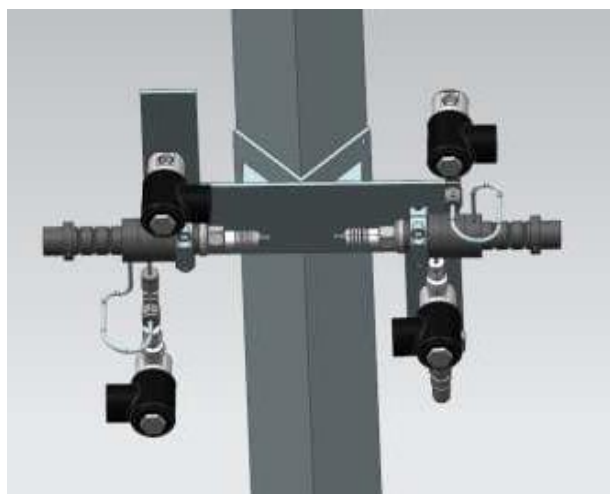

Figure 33: RCE's Mounting Example

rotation. The system assumes it will take the same time to make the lander rotate a half turn with the RCE's ignited clockwise and then decelerated in the same time using the counter-clockwise RCE's. Air resistance and difference in thrust output from RCE to RCE was neglected for this first stage calculation and increasing the number of RCE's placed on the lander will decrease the time the full rotation will take since alpha is a function of torque which increases with the quantity of RCE's installed. The full sub-system assembly will be constituted of everything the thruster requires to operate including valves, therefore making the integration to the lander modular.

Before the RCE was implemented on J-1 it was first submitted to a variety of tests including a water test to understand flowrate and pressure drop across the system. This could be then interpreted the adequate pressure required in tanks for the thruster to perform adequately could be found. This type of system would work on pressure fed systems, therefore it would integrate with no problems with Janus using a regulator to set the optimum pressure for the RCE's. The RCE was then hot fire tested using a torsional thrust stand set up. The set up (Figure 33) consists of a moment arm that has a counterweight equal to the weight of the system. They are pivoted used a nearly frictionless point to disregard energy lost through friction which would affect the thrust measurement. As thrust is applied the moment arm moves and a laser reads the displacement. The Picture below depicts the hot fire test set-up at the torsional thrust stand.

The RCE was tested up until summer 2017 when it failed for reasons non-related to the thruster and had to be salvaged. A remodeled version of the RCE is under development and is expected to be delivered by 2018.

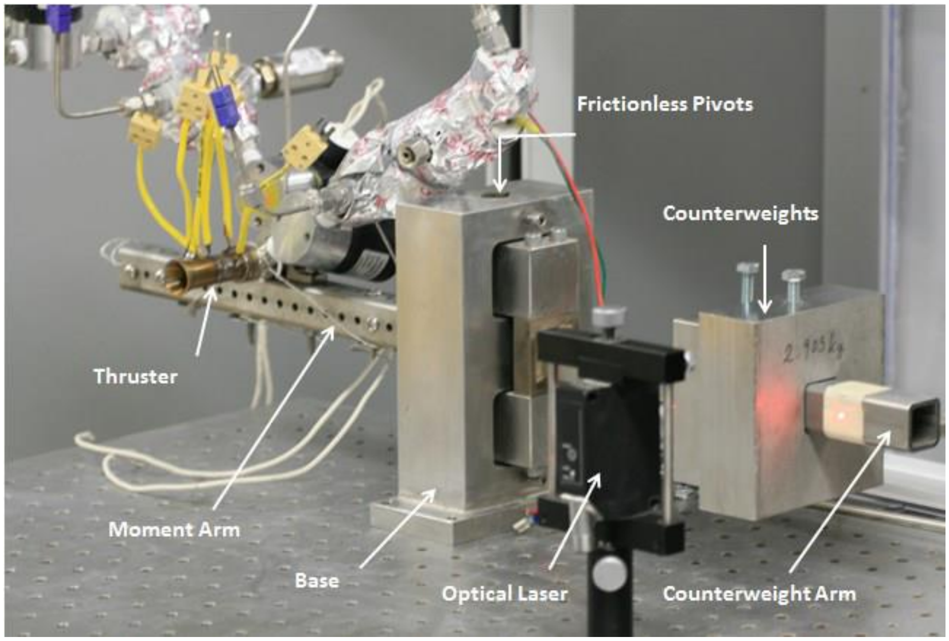

Figure 34: Torsional Thrust Stand Set-up

3.3.3 Guidance Navigation and Control (GNC)

One of the most important components for the success of Janus is the Guidance Navigation and Control (GNC). The GNC will be the landers control guide that will keep the lander stable during flight. The GNC will oversee the lander telling every component on board what to do and when to keep the lander safe throughout the mission. Unfortunately, due to lack in personnel the project has been re-scheduled and will be retaken once the project moves on to J-2, J-1 does not require GNC nor will it be possible for it to be tested therefore this is not an immediate problem.

2.3.4 Propellant Tanks and Feed System

During all three phases of the project (J-1, J-2, J-3) the delivery system is considered a sub-system assembly on its own. For J-1 the tanks will be 36 in. in diameter. The propellant tanks will house LO_2-LCH_4 which will feed not only the main engine, but also the shear coaxial torch igniter and the RCE's as well. Since Janus will be a pressure fed system, there will also be a pressurant

tank. The pressurant tank will house an inert gas, helium. These tanks will be filled to a specific pressure and through a regulator will feed the gas to the propellant tanks to keep a constant pressure in the engine's propellant feed system.

The tanks were initially being devised in-house, but due to the complexity and dangers manufacturing a tank implied, it was decided the tanks would be purchased through an external source. The tanks have been quoted through Buckeye and revisions are being made to approve and purchase the tanks for J-1. Just like many of the components for Janus, the tanks will also undergo changes as the project progresses. The tanks being developed by Buckeye for J-1 are too heavy to be considered for flight in J-2 or J-3, therefore they will only be used during J-1 to feed the different components being tested. The helium used to pressurize the tanks will come from regular K type bottles during J-1. Once the project progresses towards J-2 a flight pressurant tank will be devised along with flight propellant tanks.

The tanks being developed by Buckeye will have the following specifications:

- 36 in. inner diameter
- 3/16 in. wall thickness
- Material: Stainless Steel 304
- Max working pressure: 400 psi
- Tank Volume: 14.1 ft^3
- Price: $49,000 (2 tanks)
- Delivery time: 10-12 weeks after approval and purchase

The approximate weight of the tanks will be at 220 lbs each. The tanks selected were made of stainless steel since no flight hardware is necessary for J-1 and the durability and strength of SS

304 does not pose as big of a threat of explosion like an aluminum or carbon fiber wrapped tank would. These tanks will be kept at approximately -310° F at which the propellants will be kept at its cryogenic state. Before the tanks are filled up with propellant, they will first undergo a pre-chill process where liquid nitrogen is pumped through the tank and lines. Liquid nitrogen is used for this process since it is cheaper to vent liquid nitrogen than liquid methane or liquid oxygen. Another reason liquid nitrogen is used is because it does not present an environmental hazard and is not flammable, unlike methane or oxygen. Tank drawings provided by the manufacturer can be found in the appendix.

3.3.4.1 Propellant Tank Stands

For J-1 the propellant feed system will be divided. The tanks will be placed in different tank stands that have been specifically designed for J-1 phase of the project. Unlike J-2 or J-3 the tanks are not required to be placed on a flight configuration therefore they will be placed apart from each other. The separation of the LO_2 tank from the LCH_4 thank during this phase will prevent any probable cause for an accidental explosion to occur.

During J-1 the tank stands will support 3 g's of load in case the tank stand suffers any kind of drop or topple during the movement or installation from the facility to the test site. The tank will be moved in and out of the test facility as required, therefore the tank stand is being designed to hold the tank static with a factor of safety to yield of 3.0. Since the tank stand will not be used for J-2 the weight is not as critical, therefore its weight requirement is for the tank to be under 4500 lbs. The main dimensions of the propellant tank stand are 54 x 54 x 70 in (width x length x height). These dimensions may vary as the propellant tank stand iterations continue. The tank stand will be constructed of a material capable of withstanding any accidental damage caused by hammers or instrumentation during testing. This material will also be machinable for component brackets

or supports to be adapted to the structure during testing phase such as gauges or valves. It is also important for the tank stand interface to the tank to be able to withstand any thermal gradient between cryogenic liquid (approx. -300 F) in the tank and ambient temperature (approx. 120 F). The stand will have a minimum of 1 foot radially around the tank to leave enough space for piping installation and easy access for tools. The tank will be exposed to weather therefore it is required for thank to have corrosion resistance to LOx and weather elements as well. The full list of requirements and specs for the tank stand is in the appendix.

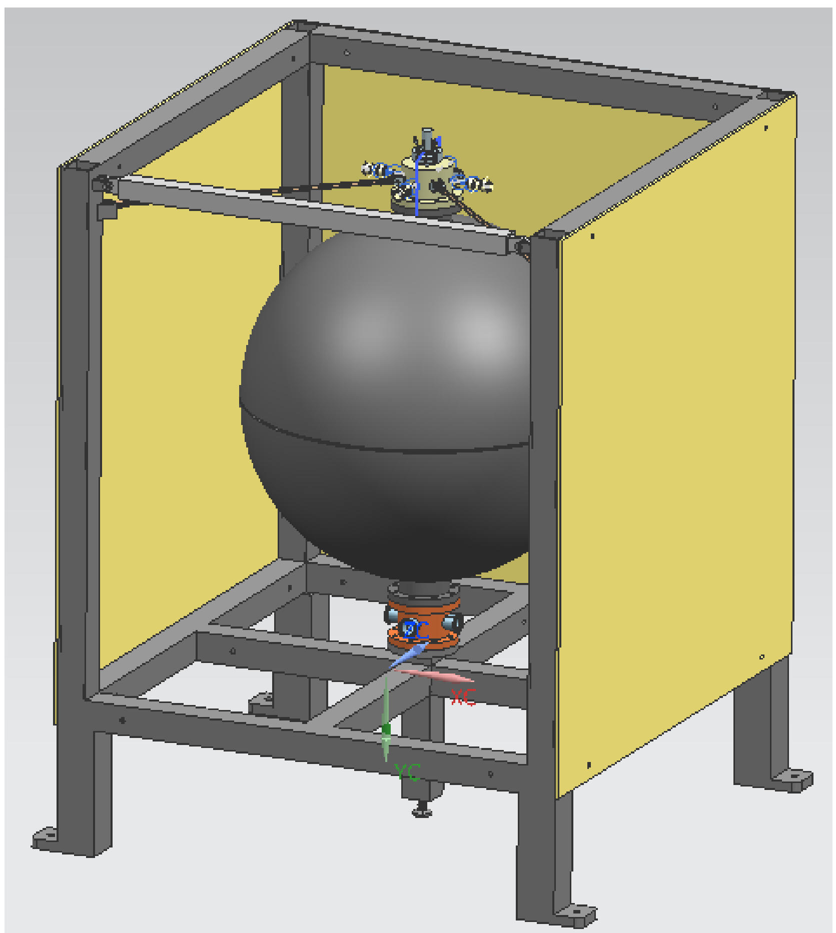

Figure 35: First Iteration Tank Stand

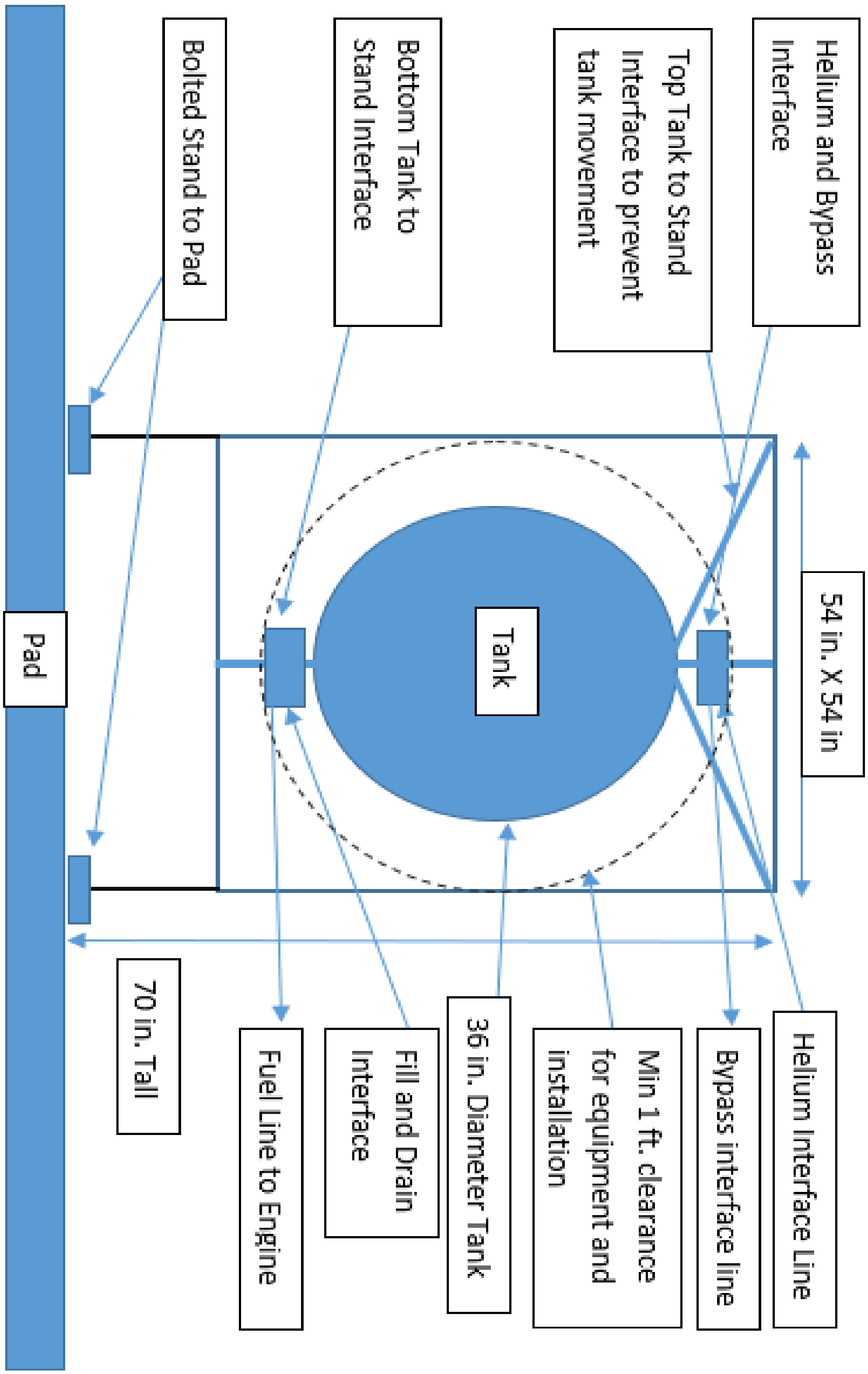

Figure 36: Tank Stand Interface Definition

Chapter 4: Summary and Conclusion

The cSETR has focused on the development of LO_2-LCH_4 propulsion technologies. Janus is part of this research where the main goal is to prove the reliability of LO_2-LCH_4 as a propellant and demonstrate the capabilities students have the develop such task. Janus will be a robotic lander that will be used as a testbed for different propulsion technologies that will be devised in house such as a 2000 lbf engine a GNC system and Reaction Control Engines. The final goal of Janus is to perform a fully autonomous flight on which it ascends to a height of 20 ft, performs a roll maneuver using RCS, and descends and lands back on the ground. The development of the lander will be done through three different stages as the projects are completed. These stages are known as J-1, J-2 and J-3. During these stages the lander will begin integrating the different technologies that are being developed at the cSETR until a fully autonomous lander is achieved.

A flight profile has been developed where the different phases of the mission are explained. Each phase has a different velocity, thrust and acceleration among others, at which the lander will be during the mission. The weight budget of the lander has also been developed. This weight budget will dictate the maximum allowable weight each component or sub-system can have in order to keep the lander under a specified weight once the integration of all components to the lander is done. A study was done to find the best possible arrangement for the tanks during the J-2 and J-3 phase was done. This study showed that a 4-propellant tank lander with different tank offsets from its central axis was the best option. At the same time, progress has been done individually on the design and testing of most of the subsystems.

The documentation of this work will serve as a tool for the new team members to carry the project forward by explaining the reasoning behind the decisions made and the process taken.

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Appendix

A.1 CROME-X Engine Data

The following is theoretical data of the engine that will be used for Janus provided by the CROME-X team. The main purpose of the data is to be able to see the total overall flowrate the system will have at a specified thrust.

Table 5: CROME-X Engine Data

Thrust [lbf]	Combustion Isp [s]	Total Combustion Flowrate [lbf/s]	LOX Flowrate [lbf/s]	Combustion LCH4 Flowrate [lbf/s]	Cooling LCH4 Flowrate [lbf/s]	Total LCH4 Flowrate [lbf/s]	Total Overall Flowrate [lbf/s]	Engine Isp [s]
2	3	0.82	0.60	0.22	0.09	0.32	0.91	3
98	84	1.17	0.85	0.32	0.14	0.45	1.31	75
193	127	1.52	1.11	0.41	0.18	0.59	1.70	114
288	154	1.88	1.37	0.51	0.22	0.72	2.09	138
383	172	2.23	1.62	0.60	0.26	0.86	2.48	154
478	186	2.58	1.88	0.70	0.30	1.00	2.88	166
574	196	2.93	2.14	0.79	0.34	1.13	3.27	176
669	204	3.28	2.39	0.89	0.38	1.27	3.66	183
764	210	3.63	2.65	0.98	0.42	1.40	4.05	189
859	216	3.98	2.90	1.08	0.46	1.54	4.44	193
954	220	4.33	3.16	1.17	0.50	1.67	4.83	197
1049	224	4.68	3.41	1.26	0.54	1.81	5.22	201
1144	228	5.03	3.67	1.36	0.58	1.94	5.61	204
1239	230	5.38	3.93	1.45	0.62	2.08	6.00	207
1335	233	5.73	4.18	1.55	0.66	2.21	6.39	209
1430	235	6.08	4.43	1.64	0.70	2.35	6.78	211
1525	237	6.43	4.69	1.74	0.74	2.48	7.17	213
1620	239	6.78	4.94	1.83	0.78	2.62	7.56	214
1715	241	7.12	5.20	1.93	0.83	2.75	7.95	216
1810	242	7.47	5.45	2.02	0.87	2.89	8.34	217
1905	244	7.82	5.71	2.11	0.91	3.02	8.73	218
2000	245	8.17	5.96	2.21	0.95	3.15	9.12	219
2095	246	8.52	6.22	2.30	0.99	3.29	9.50	220
2190	247	8.87	6.47	2.40	1.03	3.42	9.89	221
2285	248	9.21	6.72	2.49	1.07	3.56	10.28	222

A.2 Moments of Inertia of Tanks Equations

The governing equations to determine the moment of inertia of the tanks and mass of the propellant with respect to the Z axis of the lander are the following.

Moment of Inertia of a hollow sphere used for tank: $I = \frac{2}{3} m_T r^2$

Moment of Inertia of a solid sphere used for propellant: $I = \frac{2}{5} m_F r^2$

The summation of these two equations then gives us the total moment of inertia, where m_T is the mass of the tank, m_F is the mass of the propellant and r is the radius.

$$I_T = \left(\frac{2}{3} m_T r^2\right) + \left(\frac{2}{5} m_F r^2\right)$$

Both the tank and the fluid were assumed to have the same radius.

The parallel axis theorem is then used to find the Ix and Iy respectively by repeating the process for each propellant. The tank weight was assumed to be the same for both cases.

$$Ix = 2(I_T + ((m_T + m_F) * R^2))$$

Where R is equal to the distance from I's parallel axis to Z.

A.3 Propellant Tank Stand Requirements

Test Stand Operation Description

The stand will be used to assemble and house the tank assembly (from here on referred to as *Tank)* and protect the tank during transport to and from the firing area. The stand will be used to mount the propellant feedlines for testing. It will also be used to hold tank and propellant during testing. The tank will also be supplied by an external source of Helium to purge the tank and lines when required.

Interface Definition

See Figure 35

Design Loads

- Tank stand with tank must withstand 3 g's of load in case of drop or topple of stand during installation, transportation, and operation.
- Tank stand to tank interfaces must hold tank static during transportation and operation. Factor of safety to yield of 3.00.
- Stand requires impact resistance (ex. accidental hammer hit, dropped instrumentation).
- Stand must withstand 4000 lbs. at tank to stand interface to support tank, instrumentation, and piping weight. Factor of safety to yield of 3.00.
- Tank stand interface must withstand the thermal gradient between cryogenic liquid (-300 F) in tank and ambient temperature of air (120 F).
- Full stand and tank assembly (empty tank) must be less than 4500 lbs.
- Dynamic load first major mode frequency must be over 100 Hz.
- Stand must be able to resist dynamic loading at amplitude equal to or greater than the thrust level of the engine.

Design Requirements

- Two separate assemblies:

  -Tank Stand

  -Tank

- Side installation of tank (2 Hitch points on tank for transportation and installation provided by tank manufacturer) with removable side beam on tank stand to facilitate tank handling and installation.
- Tank to stand interface on top and bottom of tank.

  -Bottom: Connection from stand to tank which will also serves as inlet and outlet of propellant.

  - 4 X 1.5 in. sanitary fittings

  -Top: Connection from stand to tank which will prevent tank movement and provide inlet/outlet of pressurant and level sensor interface.

  - 4 X 12.5 mm fittings

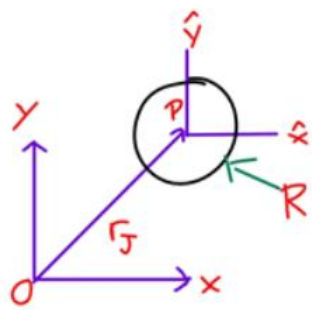
  - 1 x 2 in sanitary fitting

  Specify size and quantity
- Stand must have lift points for a forklift to install on pad.
- Tank stand must accommodate to bolt pattern on pad.
- Enough space in tank stand for tank insulation, instrumentation, interfaces, and installation (minimum of 1 foot radially from tank).
- Tank stand must provide accommodation for piping and instrumentation support.
- Tank stand must provide accommodation for electrical harnessing.
- Tank stand must have corrosion resistance to LOx and weather.
- Material used for stand must be machinable and weldable.
- Ballistic protection from fragments coming from engine failure (explosion) in contact with tank.

- Thermal barriers to withstand heat from engine combustion gases during normal operation.
- Thermal barriers to withstand heat from engine combustion gases in case of failure.
- Ballistic and thermal protection must be removable.

B.1 Mathematica Moments of Inertia Model

The following is a snapshot of the Mathematica code used to calculate the moments of inertia in the X and Y direction of the lander. This was done to determine the possibility of having the tanks at different distances from the Z axis to reach an equilibrium during the hover phase of the mission.




```
;

mT = 27 "lb";
mLOX = {65, 43.9, 22.7} "lb";
mLCH4 = {34, 23.2, 12} "lb";
R = 9.5 "in";
rJLOX = 22 "in";
rJLCH4 = 26.428 "in";

IpLOX = ((((2/3)) * mT * R^2) + (((2/5)) * mLOX * R^2));
IoLOX = 2 * (IpLOX + ((mT + mLOX) * rJLOX^2)) // N;
IpLCH4 = ((((2/3)) * mT * R^2) + (((2/5)) * mLCH4 * R^2));
IoLCH4 = 2 * (IpLCH4 + ((mT + mLCH4) * rJLCH4^2)) // N;
Itot = (2 * IoLOX) + (2 * IoLCH4);

IoLOX
IoLCH4
Itot
```

Figure 37: Mathematica Moments of Inertia Code

B.2 Propellant Tank Drawing

The following is the drawing proportioned by Buckeye of the propellant tank. Modifications are needed therefore the tank hasn't been approved yet.

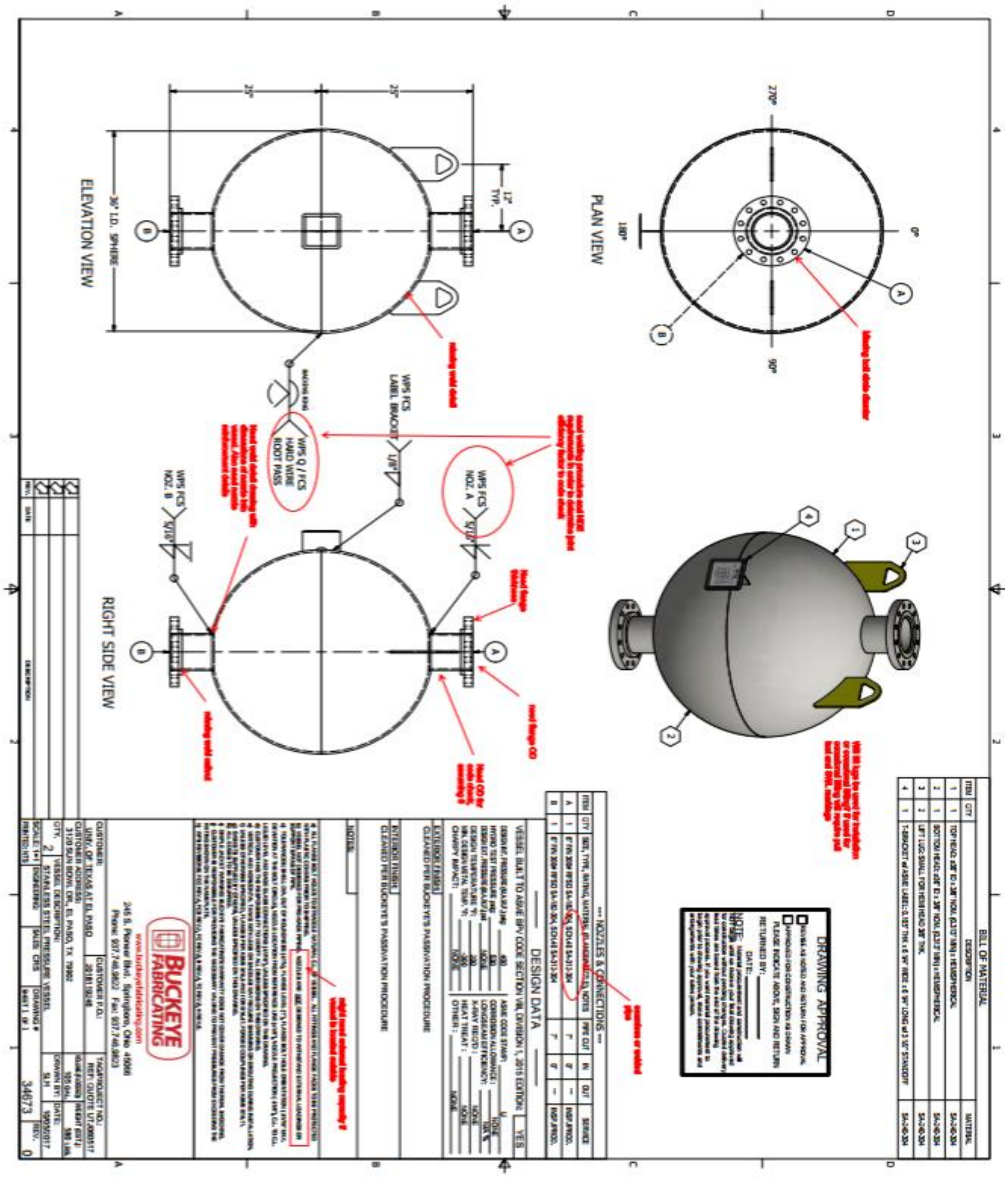


Figure 38: Buckeye Propellant Tank Drawing

B.3 MATLAB Script

The following script was used to find the flight profile of JANUS. The plots generated in section 2.3 were generated using this script.

```
clear

% Janus flight profile

% units are in ft and seconds
%
%  L = [ft]
%  V = [ft/s]
%  A = [ft/s^2]
%

g = 32.2;

dt = .005;                % timestep
H = 20;                   % height to fly to

W0 = 1510;                % initial wet weight of the vehicle
rl = -9.12/2000/g;        % mdot = r thrust  for linear
rr = rl;                  % mdot = r thrust  for rotary

de=5;                     % diameter to RCS engines
re=de/2;
ze = re^2/de;

Izz = W0/g*re^2/2;

V = 20/3.5;                 % steady state velocity
a = V/3;
Omega = 360/8;
alpha = Omega/2;
thold = dt; %2.5; % hold time
vtd = 0.25;  % touchdown velocity
dtd = 1.0;  % touchdown distance

% phase 1 -  constant acceleration
tf = V/a;
t1 = 0:dt:tf;
np = length(t1);
```

```
s1 = [ 0.5*a*t1.^2;
       a*t1;
       a*ones(1,np);
       zeros(3, np)];
fprintf('End of phase 1 = %5.2f [sec]\n',tf)

% phase 2 -  constant velocity
vs = s1(2,np);
ds = s1(1,np);
ts = t1(np);

tf = (H-2*ds)/V;

t2 = ts:dt:(ts+tf);
np = length(t2);

s2 = [ vs*(t2-ts)+ds;
       vs*ones(1,np);
       0*ones(1,np);
       zeros(3, np)];
fprintf('End of phase 2 = %5.2f [sec]\n',ts)

% phase 3 -  constant deceleration
vs = s2(2,np);
ds = s2(1,np);
ts = t2(np);

tf = vs/a;
t3 = ts:dt:(ts+tf);
np = length(t3);

s3= [ds+vs*(t3-ts)-0.5*a*(t3-ts).^2;
     vs-a*(t3-ts);
    -a*ones(1,np);
       zeros(3, np)];
fprintf('End of phase 3 = %5.2f [sec]\n',ts)

% phase 3b hold
ts = t3(np);
ds = s3(1,np);
t3b = ts:dt:(ts+thold);
np = length(t3b);
s3b = [ds*ones(1,np) ; zeros(5,np)];
fprintf('End of phase 3b = %5.2f [sec]\n',ts)
```

```matlab
% phase 4 -  constant angular acceleration
vs = s3b(2,np);
ds = s3b(1,np);
ts = t3b(np);

tf = Omega/alpha;
t4 = ts:dt:(tf+ts);
np = length(t4);

s4 = [ ds*ones(1,np);
       zeros(2,np);
       0.5*alpha*(t4-ts).^2;
       alpha*(t4-ts);
       alpha*ones(1,np)];
fprintf('End of phase 4 = %5.2f [sec]\n',ts)

% phase 5 -  constant angular velocity
vs = s4(5,np);
dds = s4(1,np);
ds = s4(4,np);
ts = t4(np);

if 2*ds<360
    tf = (360-2*ds)/Omega;
else
    tf = ts+dt;
end
t5 = ts:dt:(tf+ts);
np = length(t5);

s5 = [ dds*ones(1,np);
       zeros(2,np);
       vs*(t5-ts)+ds;
       vs*ones(1,np);
       0*ones(1,np)];
fprintf('End of phase 5 = %5.2f [sec]\n',ts)

% phase 6 -  constant angular deceleration
vs = s5(5,np);
dds = s5(1,np);
ds = s5(4,np);
ts = t5(np);
```

```
tf = vs/alpha;
t6 = ts:dt:(tf+ts);
np = length(t6);

s6 = [ dds*ones(1,np);
       zeros(2,np);
       ds+vs*(t6-ts)-0.5*alpha*(t6-ts).^2;
     vs-alpha*(t6-ts);
    -alpha*ones(1,np)];
fprintf('End of phase 6 = %5.2f [sec]\n',ts)

% phase 6b hold
ts = t6(np);
ds = s6(1,np);
dsr = s6(4,np);
t6b = ts:dt:(ts+thold);
np = length(t6b);
s6b = [ds*ones(1,np) ; zeros(2,np); dsr*ones(1,np) ;
zeros(2,np)];
fprintf('End of phase 6b = %5.2f [sec]\n',t6b(np))

% phase 7 -  constant accleration down
dds = s6b(4,np);
ds = s6b(1,np);
vs = s6b(2,np);
ts = t6b(np);

tf = V/a;
t7 = ts:dt:(tf+ts);
np = length(t7);

s7 = [ ds+vs*(t7-ts)-0.5*a*(t7-ts).^2;
     vs-a*(t7-ts);
    -a*ones(1,np);
     dds*ones(1,np);
       zeros(2,np)];
fprintf('End of phase 7 = %5.2f [sec]\n',ts)

% phase 8 -  constant velocity decent
dds = s7(4,np);
ds = s7(1,np);
vs = s7(2,np);
ts = t7(np);
```

```
tf = (H-2*(H-ds)-dtd)/V;

t8 = ts:dt:(ts+tf);
np = length(t8);

s8 = [ -V*(t8-ts)+ds;
       -V*ones(1,np);
       0*ones(1,np);
       dds*ones(1,np);
       zeros(2, np)];
fprintf('End of phase 8 = %5.2f [sec]\n',ts)

% phase 9 -  constant landing deceleration to landing
decent speed
dds = s8(4,np);
ds = s8(1,np);
vs = s8(2,np);
ts = t8(np);

tf = (-vs-vtd)/a;

t9 = ts:dt:(ts+tf);
np = length(t9);

s9 = [ ds+vs*(t9-ts)+0.5*a*(t9-ts).^2;
     vs+a*(t9-ts);
     a*ones(1,np);
       dds*ones(1,np);
       zeros(2, np)];
fprintf('End of phase 9 = %5.2f [sec]\n', ts)

% landing at constant decent velocity
dds = s9(4,np);
ds = s9(1,np);
vs = s9(2,np);
ts = t9(np);

tf = ds/vtd;

t10 = ts:dt:(ts+tf);
np = length(t10);

s10 = [ -vtd*(t10-ts)+ds;
       -vtd*ones(1,np);
```

```
        0*ones(1,np);
        dds*ones(1,np);
        zeros(2, np)];
fprintf('End of phase 10 = %5.2f [sec]\n',ts)

% ------------------
s = [s1 s2 s3 s3b s4 s5 s6 s6b s7 s8 s9 s10];
t = [t1 t2 t3 t3b t4 t5 t6 t6b t7 t8 t9 t10];

% ---------------------------------------------
% thrust and mass calculations
np = length(t);
trs = t4(1);
trf = t7(1);
T = zeros(1,np);
Tr = zeros(1,np);
T(1) = (W0/g)*s(1,3)+W0;
W = zeros(1,np);
W(1) = W0;
for n = 2:np  % Crank-Nicholson
    ddt = t(n) - t(n-1);
    a = s(3,n);
    m = W(n-1)/g;
    T(n) = ((a+g)*(m+ddt*(rl/2)*T(n-1)))/(1-
ddt*(rl/2)*(a+g));

    alpharad = s(6,n)*(pi/180);
    Tr(n) =  (m+0.5*ddt*rr*T(n-1))/...
        (1/(ze*alpharad)-0.5*ddt*rr);

    W(n) = W(n-1) + g*0.5*ddt*rl*(T(n) + T(n-1)) ...
        + g*0.5*ddt*rr*(Tr(n) + Tr(n-1));
end

%-----------------------------------------------
figure(1)
clf
subplot(3,1,1)
[hAx,hLine1,hLine2] = plotyy(t, s(1,:), t, s(4,:));
grid on
title('Janus Flight Profile, Displacement')
xlabel('Time [sec]')
ylabel(hAx(1),'Height [ft]') % left y-axis
ylabel(hAx(2),'Rotation [deg]') % right y-axis
```

```
subplot(3,1,2)
[hAx,hLine1,hLine2] = plotyy(t, s(2,:), t, s(5,:));
grid on
title('Janus Flight Profile, Velocity')
xlabel('Time [sec]')
ylabel(hAx(1),'Velocity [ft/s]') % left y-axis
ylabel(hAx(2),'Angular Velocity [deg/s]') % right y-axis

subplot(3,1,3)
[hAx,hLine1,hLine2] = plotyy(t, s(3,:), t, s(6,:));
grid on
title('Janus Flight Profile, Acceleration')
xlabel('Time [sec]')
ylabel(hAx(1),'Acceleration [ft/s^2]') % left y-axis
ylabel(hAx(2),'Angular Acceleration [deg/s^2]') % right y-
axis

figure(2)
clf
subplot(3,1,1)
plot(t,T)
grid on
title('Janus Flight Profile, Thrust')
xlabel('Time [sec]')
ylabel('Thrust [lbf]')

subplot(3,1,2)
plot(t,Tr)
grid on
title('Janus Flight Profile, RCS Thrust')
xlabel('Time [sec]')
ylabel('Thrust [lbf]')

subplot(3,1,3)
plot(t,W)
grid on
title('Janus Flight Profile, Weight')
xlabel('Time [sec]')
ylabel('Weight [lbm]')

Wf = W(length(W));
fprintf('Total propellant consumed %7.2f [lbm]\n',W0-Wf)
```

Vita

Jahir Fernandez was born in El Paso, Texas where he studied all the way through college. After graduating from Father Yermo High School he pursued his Bachelor of Science in Mechanical Engineer. During his las two years as an undergraduate, Jahir worked full time as a design engineer where his passion for engineering grew. After graduating Jahir decided to continue his education and pursue a Master's in Mechanical Engineering while working as a research assistant for the center for Space Exploration and Technology Research (cSETR).

Jahir has accepted a job offer from General Motors Company as a TRACK Engineer where he will be in the interior design branch. He will begin working after receiving his Master's in Mechanical Engineering on December 2017.

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This thesis was typed by Jahir Fernandez.