Design of a Mission to Mars Using Electric Propulsion for Small Spacecraft

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Abstract

This paper explains the importance of electric propulsion in the aerospace industry and how this technology is evolving. Many big satellites have been incorporating some type of electric propulsion systems to reduce the weight of carrying a large quantities of bipropellant fuel. Electric propulsion systems are mainly used for GEO satellites to perform north and south station keeping maneuvers, but because this technology is advancing fast some satellites are including this type of systems to perform orbit raising and this increases the life of the satellite.

This paper studies the use of electric propulsion systems for small satellites and defines the advantages of using this type of propulsion for small satellites due to their weight restrictions, as well as a design of a mission to Mars using an electric propulsion system for small satellites. The mission focuses on a small satellite that is deployed by a big GEO satellite into GEO. Once the small satellite is deployed in GEO, it will utilize electric propulsion to get to a Mars capture orbit. Once the satellite achieves Mars orbit, it will deploy Mars surface science payloads.

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Nomenclature

EOR	Electric Orbit Raising
ESA	European Space Agency
GEO	Geostationary Earth Orbit
GTO	Geostationary Transfer Orbit
HET	Hall Effect Thruster
Isp	Specific Impulse
LEO	Low Earth Orbit
MCO	Mars Capture Orbit
MMS	Mars Mission Satellite
MTO	Mars Transfer Orbit
NASA	National Aeronautics and Space Administration
PPT	Pulse Plasma Thruster
PPU	Power Processing Unit
SSL	Space Systems Loral
TRL	Technology Readiness Level

Chapter 1

Introduction

Electric propulsion is not a new concept in the Aerospace industry, it has been around for decades and was first suggested by Robert H. Goddard in 1906 [7]. There are multiple applications for electric propulsion that a regular chemical propulsion system will be limited to do. In the past decade, electric propulsion systems have become more popular due to their wide range capability. Currently satellite companies are including electric propulsion systems into their satellites. Depending on the size and in what orbit the satellite is located, there are multiple types of electric propulsion systems that can be used.

Even though electric propulsion systems are not capable to take a rocket out of the atmosphere due to their low thrust/weight ratio, they are very impressive and efficient for in-space applications. One of the most commonly use applications for these types of thrusters is to maintain a satellite in orbit, which is called station keeping. Satellites are usually disturbed by solar drag, Radio Frequency drag, and other disturbances in space. Geostationary satellites are usually place in their orbit using common chemical thrusters, which deliver more thrust, but the fuel required to do that adds a lot of weight to the satellite making it more expensive to launch. Some satellites companies are starting to use electric propulsion systems not only for station keeping maneuver but also to perform orbit raising. Conducting orbit raising using electric propulsion takes longer, but it saves the satellite's customer valuable space and weight that can be used to add more payload.

With space exploration growing rapidly and the demand for low mass, more efficient, and low power consumption propulsion systems, the electric propulsion systems are starting to play a more important role specially for long distance exploration. NASA has been researching into electric propulsion systems for deep space missions. This type of propulsion is very convenient for deep space missions due to its higher Isp, which requires much less propellant compared to chemical thrusters. Even though electric propulsion systems lack of a fast acceleration because the thrust produced is very small, they can be activated for long periods of time to produced sufficient ΔV to reach other planets and moons.

Small satellites are very popular, affordable, and used by different government agencies and private companies to conduct research and science. Most CubeSat and some small satellites lack of a propulsion system due to the weight and size, and power restrictions. Small satellite developers, including NASA and other private companies are starting to implement Hall Effect thrusters. Hall Effect Thrusters and electric propulsion systems are normally used in large GEO satellites due to the higher price on these types of thrusters. There has been a lot of researching and implementation of electric propulsion systems in small satellites in the past decade and it will continue to increment due to the popularity and affordability of small satellites.

Previous Work on Electric Propulsion Systems

Due to the forgoing interest throughout the Aerospace industry, government and private agencies are researching and testing different types of electric propulsion for future missions. Due to the qualities and performance of electric propulsion this topic is addressed constantly for in-space application, including station keeping maneuvers for satellites, or deep-space traveling for future missions focusing in going to other planets or moons far away from Earth. Depending on the application some investigators and companies focus their research in electric propulsion systems for small spacecraft and other companies in electric propulsion that can be used for bigger spacecraft.

One of the main reasons for researching and utilizing electric propulsion in satellites and spacecraft is to reduce the cost of the mission and avoid carrying large quantities of chemical fuel that increase spacecraft mass. Another reason for using EP the higher efficiency that it can be achieve because of the higher Isp of EP systems when compare to common chemical propulsion systems. There are many types of EP systems available in the market depending of the spacecraft needs and size.

Pulse Plasma Thruster (PPT) is a type of electric propulsion that is used for small spacecraft or Nano-satellites, however, it does not have the thrust to weight ratio to operate with bigger spacecraft. Recently there has been an increase in the need and use of small spacecraft particularly for LEO missions, mostly because they are more affordable than a bigger spacecraft and they still provide the investigators with good data for their experiments. In his thesis paper Peter Vallis

Shaw [15] explains the importance and different applications that a PPT will have in a small spacecraft.

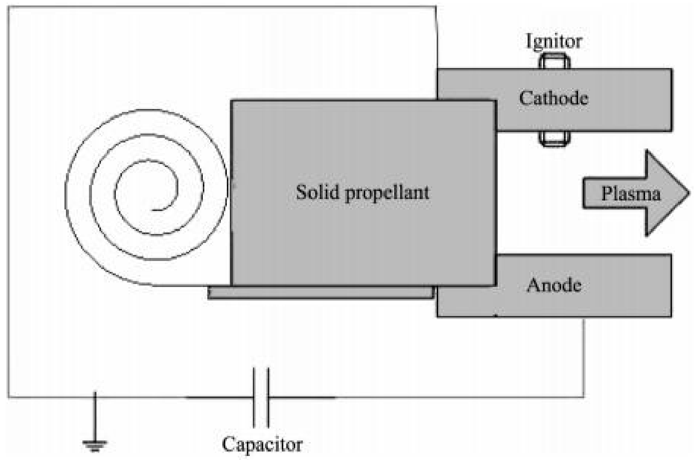


Figure 1 Pulse Plasma Thruster (PPT) Parts [14]

The above figure is a concept of the traditional layout and design of a PPT that uses a solid propellant (e.g Teflon). Shaw focuses on the need and the advantages of putting PPT in CubeSat. By installing PPT to CubeSats, principal investigators will be able to have active attitude control of the satellite instead of passive attitude control. The implementation of PPT to CubeSat would start a new paradigm for the CubeSat missions, which usually does not require of much input from the ground after they have deployed.

Continuing with interest in applying electric thruster technologies to small satellites, J. Mueller, R. Hofer, and J. Ziemer, discuss different type of propulsion that can be used in CubeSats. Muller, Hofer and Ziemer provide comparison between chemical and electric propulsion and emphasize the importance of using EP propulsion for CubeSats and also explain the improvements in PPTs that have allowed for a miniaturized PPT system. For example, the Dawgstar created by Aerojet as seen in figure 2.

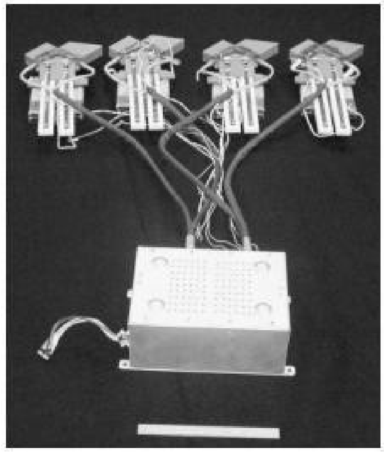


Figure 2: Dawgstar PPT system [14]

This system is a type of PPT that has four thruster modules and only one power processing unit (PPU) that delivers the power to the four modules. This is an innovative way of reducing weight by just having one PPU for four different thruster modules. One of the drawbacks of PPT is that they usually have low efficiency when compare with other types of electric propulsion systems. Even though these types of thrusters are great for small satellites, PPT do not have the thrust to weight ratio and characteristics required by the MMS mission analyzed in this paper.

On the paper by R. Shimmin, another interesting piece of technology called Resistojet thrusters is described. This is another type of electric propulsion that is a high interest for the small satellite community and is simplest forms of electric propulsion, which consists of heating up the propellant that in this case is a stored gas. This type of propulsion can provide the small satellite with a force in the mN ranges depending on the propellant used. The amount of power used to heat up the gas to make these thrusters work is in the 50W range. Resistojet thrusters are ideal for small satellites that do not require a complex thrusters' configuration but will still get high efficiency from these thrusters [27]. This technology also does not satisfy the requirements to be used in the MMS design.

JP Sheehan, Benjamin W. Longmier, Ingrid Reese, and Timothy Collard propose of a new low power plasma thrusters for Nanosatellites. In the design, they proposed an electrodeless plasma thruster that will be about 1mN, an Isp of 1000 sec, and 20% efficiency that they can use with CubeSats. Their main objective is to advance in the development of thrusters for CubeSats because these types of satellites are becoming more important and commonly used for different agencies and companies. They also mention that these thrusters will be capable of station keeping, formation flying, and even of deorbiting the satellite when there is no longer a need for the satellite [13].

Another interesting EP system is the Hall Effect Thrusters (HET). These types of thrusters have a higher thrust to weight ratio than the other smaller thrusters discussed above. HET are devices that consist on a cylindrical channel with an anode in the interior, also have a magnetic field across the cylindrical channel. The system also has a cathode to neutralize the ionized gas particles that are being accelerated through the cylindrical channel to produce the thrust. These types of thrusters are starting to get more commonly use in GEO satellites because this technology allows the satellite to be more efficient and do not require to carry large amounts of propellant for chemical thrusters. The satellites utilize these types of thrusters to perform different tasks, of which one of the most important ones is station-keeping: the maneuver that a satellite performs to stay in the same geographic location in space. Satellites have to perform these maneuvers because of external disturbances that might move the satellite and put it out of its required location. This type of maneuver is usually done using conventional chemical thrusters, but satellite companies are starting to implement more HET systems that aid in reducing the mass of the satellite by carrying less chemical propellant. Another important stage where the satellite needs a propulsion system is when the satellite needs to fire its thruster to reach at specific orbit in space. This stage is called orbit raising and is another task still dominated by conventional chemical thrusters. Satellite companies are starting to perform electric orbit rising (EOR). This process usually take about ten days to do using chemical thrusters, but doing EOR will take about six month to reach the same orbit. One of the biggest benefits of performing EOR is that it will extend the satellites effective life because it is more efficient.

M Dudeck, F Doveil, N Arcis and S Zurbach discuss the use of Hall Effect thrusters in GEO satellite communication and explain why electric propulsion is used instead of conventional bipropellant propulsion. One of the biggest reasons as explained before is using electric propulsion is to maintain the satellite in orbit due to external disturbances like solar winds, Sun radiation effects [18]. Figure 4 shows the different parts of a Hall Effect thruster

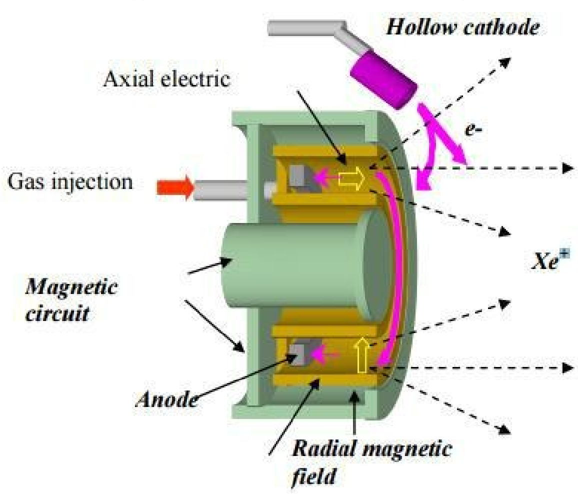


Figure 3: Hall Effect Thruster [18]

The most common type of Hall Effect thruster uses Xenon gas because it has been flight proven and has a high TRL level. Also, the configuration that is mostly used is the Annular Hall Effect thruster, which has a magnetic circuit in the middle similar to figure 4. European agencies are also considering electric propulsion and J. Gonzalez and G. Saccoccia explain the importance of electric propulsion and how European Space Agency (ESA) is developing EP for their current missions and future missions. In is clear to notice that other agencies like ESA are also advancing in the development of new technologies of electric propulsion to reduce cost of space missions. They also discuss the trend in telecommunication satellites in using electric propulsion to reduce the mass of the satellite and also by using electric propulsion they are able to increase the life of the satellites [11].

NASA is also looking to use electric propulsion more often in upcoming missions and improve the technology already in existence. NASA’s Solar Electric Propulsion (SEP) is a project that is focuses on using the power captured from the Sun using solar panels, and it used to power the electric propulsion systems onboard instead of using conventional chemical thrusters. Michael Barrett, who is the SEP project manager, explains how they are conducting tests to develop this technology in NASA’s Glenn Research Center, which is their leading center for solar electric propulsion [14]. He emphasizes that this is going to be the future of in-space propulsion because of all the improvements that this technology is bringing into space missions. He said that SEP will use a Hall Effect Thruster with advance magnetic shielding, which is capable of accelerating Xenon particles to 104,607 km/h allowing for orbit transfer maneuvers and space exploration. Barrett also explained that the plan is to make this system modular that it will be able to combine multiple Hall Effect thrusters to increase the force produce and get a necessary ΔV [21]. NASA is currently interested in developing new technologies for electric propulsion systems. In the NASA Technology Roadmap under the subtopic of in-space propulsion technologies, it is clear that they are seeking constantly for new proposals for this type of technology. NASA also provides with the current Technology Readiness Level (TRL) shown below for different technologies and what they are looking to get by specific times.

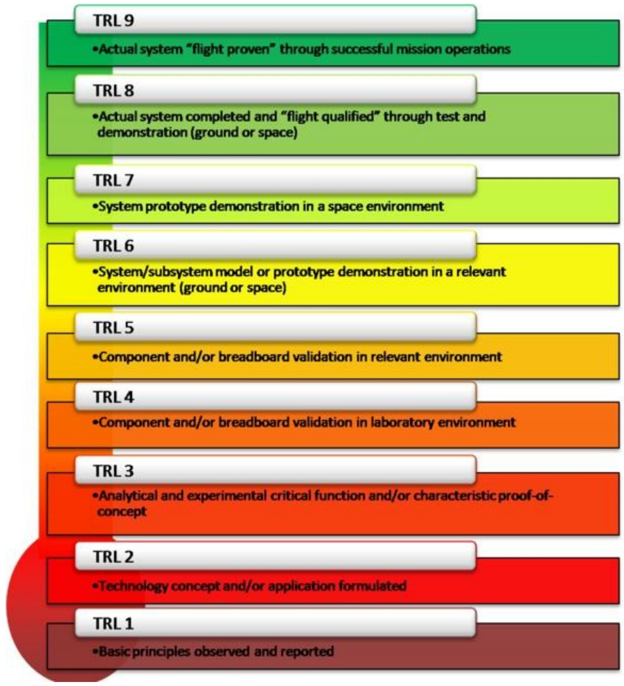


Figure 4: NASA TRL Chart [21]

2.2 Non-Chemical Propulsion 2.2.1 Electric Propulsion	**2.2.1.2 Hall Thrusters**

TECHNOLOGY

Technology Description: Hall thrusters are electrostatic thrusters that use a cross-field discharge described by the Hall effect to generate and accelerate the plasma.

Technology Challenge: Challenges include scaling to high-power and higher specific impulse (I_{sp}), and achieving sufficient lifetime. Integrated power processing unit drives cost, schedule, and risk for flight system implementation.

Technology State of the Art: 10 to 15 kW technology development unit. 50 to 100 kW laboratory unit. 100 kW proof of concept nested channel laboratory unit.		**Technology Performance Goal:** Near-term objective is to mature 10 to 15 kW system to flight and continue to explore higher power levels and long life for exploration missions to Mars and beyond.	
Parameter, Value:	**TRL**	**Parameter, Value:**	**TRL**
15 kW with 60% efficiency and 3,000 seconds I_{sp}; 50 kW with ~60% efficiency and 2,400 seconds I_{sp}; 60 kW with 65% efficiency and 2,500 seconds I_{sp}	3	Power: 10 to 100 kW; I_{sp}: to 3,000 seconds; Life: to 50,000 hours; Power Processing Unit input voltage: > 200 V	4

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

Figure 5: Hall Thruster Technology [21]

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Extending Reach Beyond LEO: DRM 5 Asteroid Redirect – Robotic Spacecraft	Enabling	2015	2018	2015	1 year
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	3 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	3 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	3 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	3 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	3 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	3 years
New Frontiers: Saturn Probe	Enhancing	--	2024	2016	1 year
New Frontiers: Trojan Tour and Rendezvous	Enhancing	--	2024	2016	1 year
Planetary Flagship: Mars Sample Return	Enhancing	--	2026*	2023	2 years
Planetary Flagship: Europa	Enhancing	--	2022*	2019	1 year
New Frontiers: Io Observer	Enhancing	--	2029	2021	1 year

Figure 6: Future Missions with Electric Propulsion Technology [21]

From NASA's technology roadmap, while it is evident that the future of in-space propulsion is moving toward electric propulsion, figures 6 and figure 7 show that the TRL of this technology has not matured to it full extend and there is still much be done with this technology. Also, the missions shown in figure 7 are just a few of the missions that NASA is planning on utilizing EP [21].

According to Richard Hofer, NASA's Jet Propulsion Laboratory (JPL) has been another important contributor to in space propulsion. JPL does a lot of testing and developing of conventional chemical thrusters but now they are also creating EP thruster to be used in NASA's missions. The engineers in JPL are aiming to increase the efficiency of the thruster while lowering the cost. All this effort is focused on increasing space transportation and space exploration [24].

Edgard Y. Choueiri, who teaches astronautic and applied physics at Princeton University and is the director of the electric propulsion and plasma dynamics laboratory, talks about NASA's ambition to create an efficient electric plasma engine to be used for future space missions including outer solar systems [6]. He talks about a mission developed by JPL called Dawn, which uses electric propulsion as its main propulsion. They chose to use electric propulsion because this type of propulsion has shown that work better for long distance missions which would require more propellant for conventional chemical thrusters.

Choueiri explains how the electric propulsion engine performs differently than a regular chemical rocket and also talks about the Hall Effect Thruster and how this type of Thruster is superior to the Ion Thruster that has a lower thrust to weight ratio. He also explains how the Hall current works, which is the main principle of this thruster: depending on the power that is put into the thruster the gas can accelerate to very high speeds in some cases up to 50,000 m/s

Wendy Lewis, Director of communications at Space Systems Loral (SSL) announced that SSL will start to collaborate with Busek inc. to advance in the development for more solar electric propulsion systems. SSL has been providing satellites with electric propulsion since 2004 and has built 18 satellites that use electric propulsion. Lewis also said that the collaboration with Busek will allow both companies to advance faster into the demand for electric propulsion, which will help them achieve their common goal of implementing this technology into their future projects. SSL uses Stationary Plasma Thrusters (SPT), which is another name for Hall Effect Thrusters that are used to maintain the satellite in their respective orbit by correcting the attitude of the satellite from disturbances in space. SSL is planning to use their SPTs in their satellites to perform EOR maneuvers that place the satellite in its designated location around the GEO belt [31].

One of the issues encountered when using HET is the plume produced by these thrusters. Y. Raitses, L. A. Dorf, A. A. Litvak, and N. J. Fisch conducted an experiment on the reduction of the plume produced by the Hall Effect thruster [32]. In their paper, they discuss about some of the advantages of using electric propulsion thrusters. Their research focuses in study the plume produced by these types of thrusters and how that is affecting the wall of the thruster and other parts in the thruster itself. Plume produced is also studied because the residual plume can get impinge into another spacecraft part and start eroding the impinged part including the solar arrays and therefore reducing its efficiency.

After experimenting with different propellant James Szabo, Mike Robin, Surjeet Paintal, Bruce Pote, Vlad Hruby, and Chas Freeman discussed the use of Iodine in EP systems instead of using Xenon as a propellant. Currently Xenon has been the most commonly used propellant on HET. The authors of this paper have seen some potential in using Iodine based on its characteristics and they believe that using Iodine will benefit the technology because it requires a lower storage pressure and also is high density. In their experiment, they discovered that Iodine's plume doesn't diverge as much as Xenon and also has an anode efficiency of 65%. Iodine can be stored in its solid state and this does not require as much pressure as a Xenon tank. It does, however, require heat of about 80-100 degrees Celsius to convert it gas.

They tested a Hall Effect thruster using Iodine and they noticed that the deposition of Iodine in the spacecraft was very small to none. The only material that they believe that might get affected by the Iodine plume is iron due to chemical reaction. It was concluded that Iodine will be a very good alternate propellant and in some cases, can be superior to Xenon [12]. Even though Iodine would be a good propellant candidate to analyze for this paper. The MMS will use Xenon as its propellant due to its flight proven reliability.

NASA has flown a few missions that used EP as their primary propulsion system. Deep Space 1 (DS1) was a mission that used Ion thrusters as its main propulsion system. The purpose of this mission was to fly by asteroid Braille first and then fly by comet Borrelly. These maneuvers were performed mostly using the Ion thruster onboard. This satellite also utilized a small quantity of chemical propellant for attitude control [10].

Another mission developed by NASA that used EP was DAWN. DAWN is a space probe that is still active and its mission was to fly by Vesta and Ceres. This probe is also using three Xenon Ion thrusters similar to the thruster used on DS1 [15].

Project Approach

After conducting research on electric propulsion systems available it was determine that the Hall Effect Thrusters present noticeable advantages of implementing this technology to perform interplanetary missions. This is because the HET has proven to be a reliable electric propulsion system due to its thrust to weight ratio, high TRL level, long life cycle, and high Isp. There is still a lot that can be learned from gathering computational and experimental data to keep expanding the understanding of this technology.

The objective of the project is to conduct a feasibility study of a small satellite to utilize vacant cavity on a SSL's GEO satellite and then use HET propulsion system to perform an interplanetary mission. The idea is to incorporate a small satellite in the vacant cavity and then separate it from the GEO satellite once the satellite is already on a geostationary orbit. The small satellite shall utilize a Hall Effect thruster to reach Mars orbit. Once the satellite is on Mars orbit, it will deliver a science payload on Mars surface. The HET will be used based on the characteristics explained above and it will also help to reduce the propellant mass required compared to a chemical thruster therefore increasing payload capability.

Chapter 2

Mission Specification and Requirements

The mission will consist of a small satellite called Mars Mission Satellite (MMS) that will be attached to a bigger GEO satellite and will separate/undock from it once the bigger satellite reaches GEO. After the MMS separates from the big satellite, it will perform a series of self-evaluation checks to test functionality of all its subsystems. Once the MMS proves to be fully functional to start its mission, the Satellite will be commanded to start it journey to Mars using Hall Effect Thrusters as its propulsion system.

The purpose of this mission is to send a small satellite from GEO to Mars orbit by using electric propulsion, thereby reducing the propellant amount required compare to conventional bipropellant propulsion systems. This will allow for the MMS to carry a bigger payload. Once the MMS reaches Mars orbit, it will deploy Mars surface science stations. After the two payloads are deployed, the MMS will continue to orbit the red planet to serve as a relay satellite to transmit the telemetry and commands back and forth between the payloads and the mission controllers.

One of the possible companies that will be able to provide MMS a ride to GEO is Space Systems Loral (SSL) and its Payload Orbital Delivery System (PODS). This mechanism will release MMS in GEO and then continue to its Mars mission. MMS design is based on the specifications of the PODS [29].

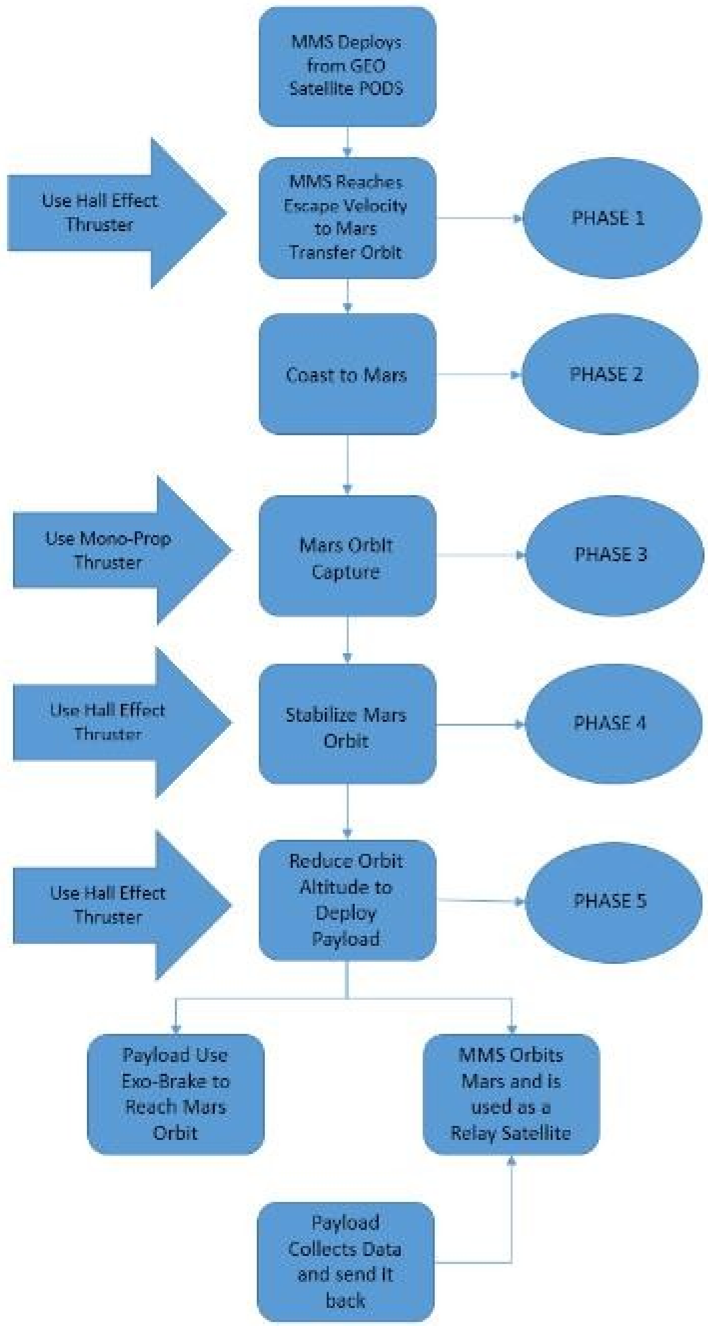


Figure 7: Mission Plan Diagram

Chapter 3

MMS Designs

The MMS will consist of multiple subsystems. This paper will go over some of the subsystems onboard of the MMS. The purpose of this paper is to design a mission that will take a small satellite from GEO to Mars Orbits using electric propulsion. As stated before, the design of the satellite will be based on the SSL PODS specifications.

The two major constraints that will define the MMS design are the volume of the satellite and its maximum allowed mass: based on the PODS's specification the maximum size that the MMS can be is 1m x 0.5m x 0.4m (standard size) and 1m x 1m x 0.6m (extended size). The extended size will be used for design purposes of the MMS. The maximum allowed mass for the extended size is 150kg [29]. These are the two major constraints that will defined the MMS design.

The focus of the mission is to design small satellite that will take science payload to Mars using electric propulsion and the selection of the electric propulsion system and its components. The design of the electric power system of the MMS is an important component because this will be the subsystem that will feed the required power to the propulsion system. The electric power system needs to be able to feed power to the propulsion system while proving power to other subsystems to maintain others' satellite functions.

The MMS shall have a small Xenon propellant tank that can hold the required propellant to complete its mission. Mercury was also considered as a propellant, but it is toxic, expensive, contamination of the spacecraft, and environmental issues during ground testing [8]. The quantity of propellant required will be determined by the type of propellant and the efficiency of the propulsion system chosen for the MMS mission.

The MMS will also have a small chemical thruster that will be used to perform ΔV maneuvers for Mars Capture Orbit. An additional tank will be required to store the propellant. The mono-propellant thruster will be used to make instantaneous changes in velocity to adjust orbit and speed because of it higher thrust when compared to the HET onboard the MMS.

Subsystems Design

The Propulsion subsystem, which is the main focus on the design of this satellite will consist of a HET that will be used mainly to take the satellite from GEO to Mars Transfer Orbit (MTO) and attitude control. The electric propulsion system will consist of two HET for redundancy. The two HETs will be store inside the satellite to fit the PODS size requirements. Once the MMS is deployed the HETs will get released and locked on the back side of the satellite. The thrust of each of the Aerojet BPT-2000 HETs is 123mN.

The MMS will also consist of a small chemical thruster that will be mostly used for Mars Capture Orbit (MCO). The small chemical thruster will be used to attain the ΔV needed for the satellite to go from a MTO to a MCO faster than what it would take the HET onboard the satellite.

In order to supply the power needed to the satellite, the MMS power subsystem will include two solar arrays. The solar arrays assembly will be attached to the MMS via a rotating arm that will allow the solar arrays to track the sun. This special feature will keep the satellite power positive during the journey to mars allowing the solar arrays to turn towards the sun position. Each solar array consists of three panels, each of the panels will have a surface area of $0.40m^2$ and produce 0.54 kw. The six panels will produce 3.247 kw based on the Solaerotech's 3rd generation triple-junction solar cell used on this design. The power produced by the two solar arrays will be more than the minimum of 1.2kw required to power the HET. that will deploy and then they will lock in place.

The satellite will carry two payloads, each of the size of a 3-U CubeSat to insert into the Martian atmosphere when it reaches Mars. The payloads will consist of two small Mars surface science stations that will collect data from Mars surface. The payload will be stored inside the MMS and then jettison from the satellite from an opening in the front side of the satellite. The mechanism releasing the payloads in mars will be spring-loaded system. The payload shall have an Exo-Brake, currently studied and tested at NASA Ames Research Center, which will assist to control re-entry and reducing impact on the surface.

The MMS will also have a small Ka Phase Array Antenna (PAA) developed by Boing to relay information from the Mars surface science payloads to the relay satellites and then to Earth [3]. This antenna in combination with the Small Deep Space Transponder also used in DS1 will be the communication subsystem of MMS [28].

Table 1: Comparison of MMS Total Mass and DS1 Total Mass

MMS Mass Properties				**DS1 Mass Properties**	
Part	**Quantity**	**Mass/Unit (kg)**	**Total (kg)**		**Mass(kg)**
Structure Frame (Aluminium 6061)	1	12.959	12.959	Dry Mass	373
Electric Power System	1	12.9	12.9	Xenon Propellant	81.5
Hall Effect Thruster	2	5.2	10.4	Chemical Propellant	31.1
Chemical Thuster	2	2	4	Total	485.6
Xenon Propellant	1	50	50		
Chemical Propellant	1	40	40		
Payload	2	4	8		
Antenna	1	1.8	1.8		
Transponder	1	3.2	3.2		
Miscellaneous (Cables, Tubing, etc)			4		
Total			147.259		

CAD Subsystems Design

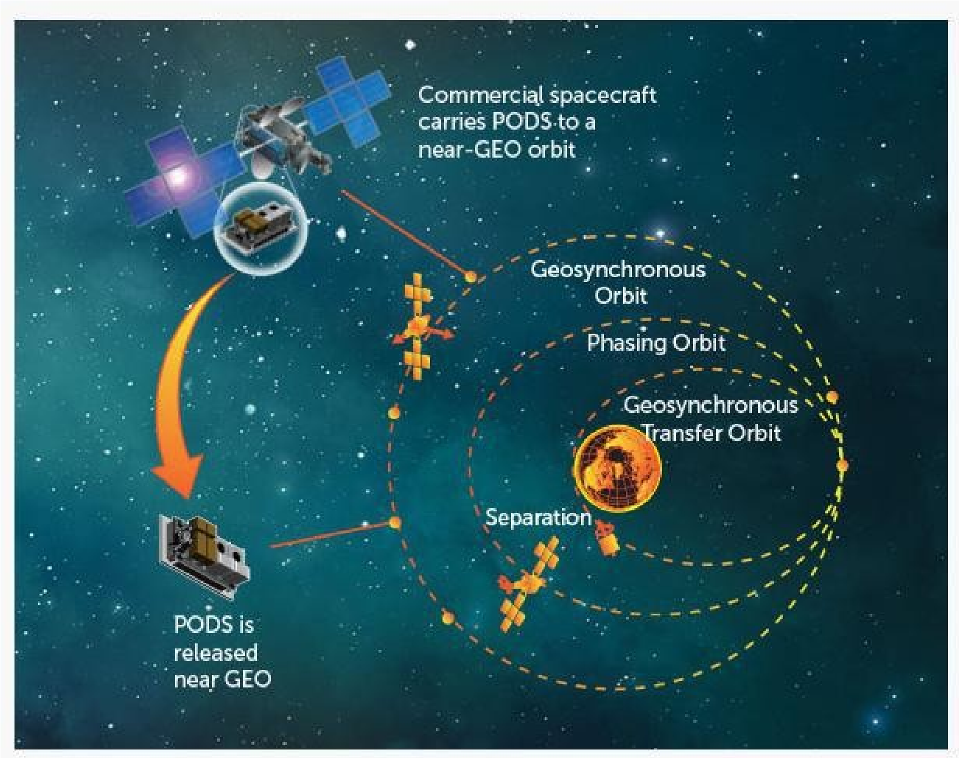


Figure 8: SSL PODS Mission Concept [29]

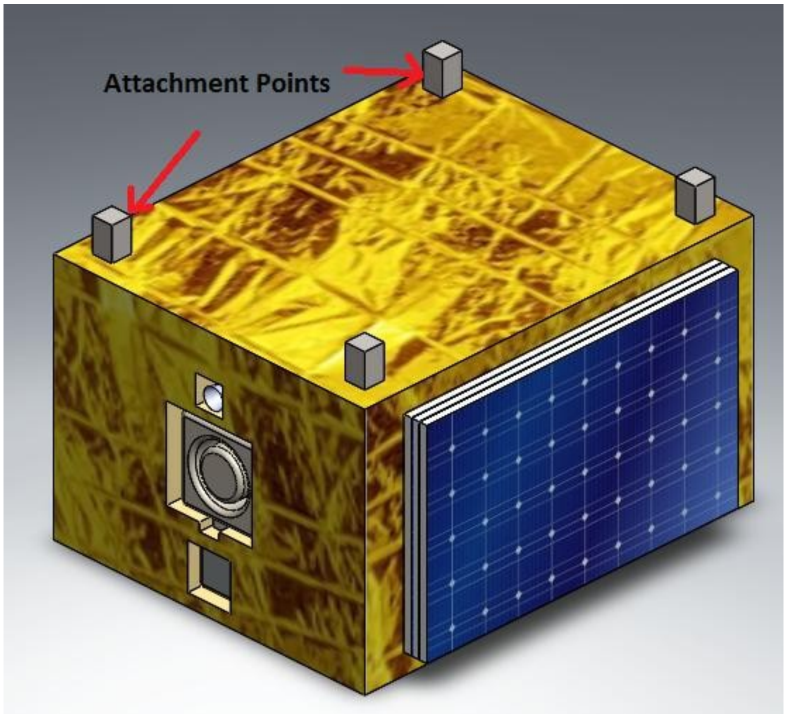


Figure 9: Stored Configuration of MMS [4]

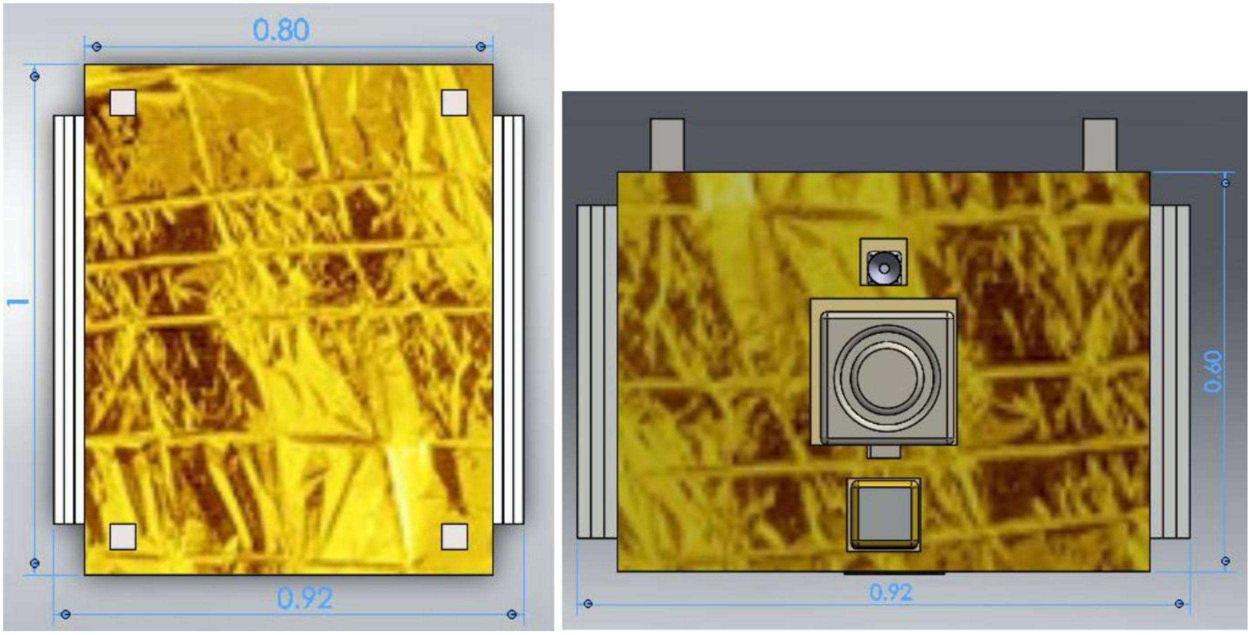


Figure 10: Dimensions in Meters Showing Design Within Requirement [4]

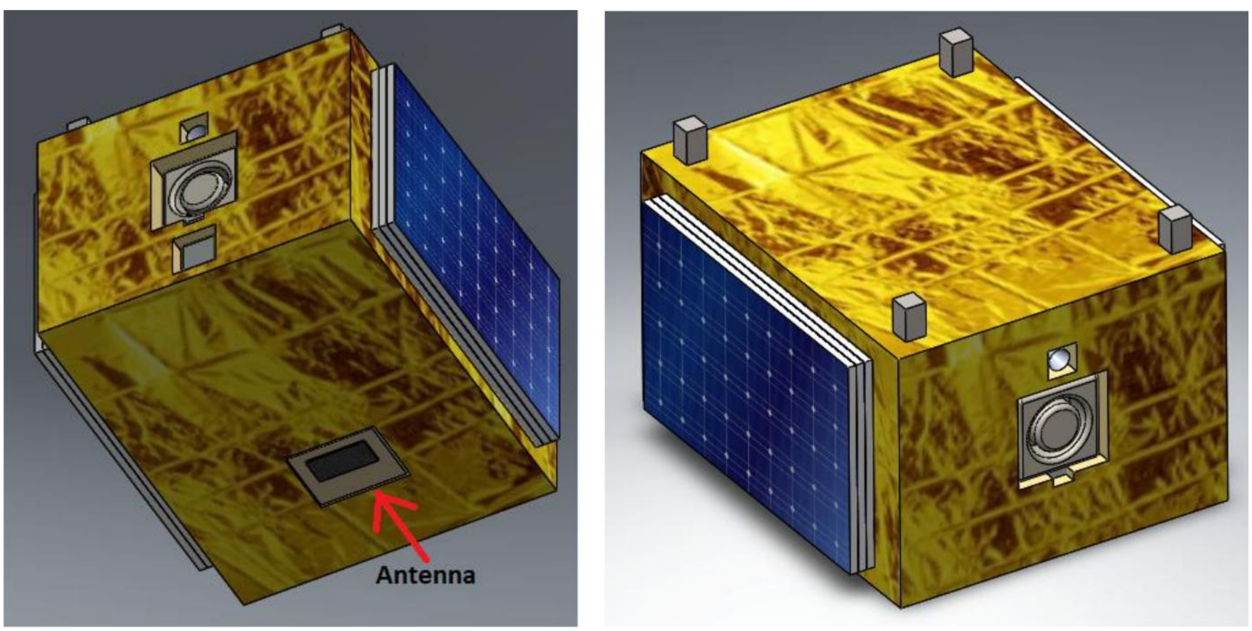


Figure 11: (Left) Bottom View of MMS with Stowed Antenna. (Right) Back View of MMS [4]

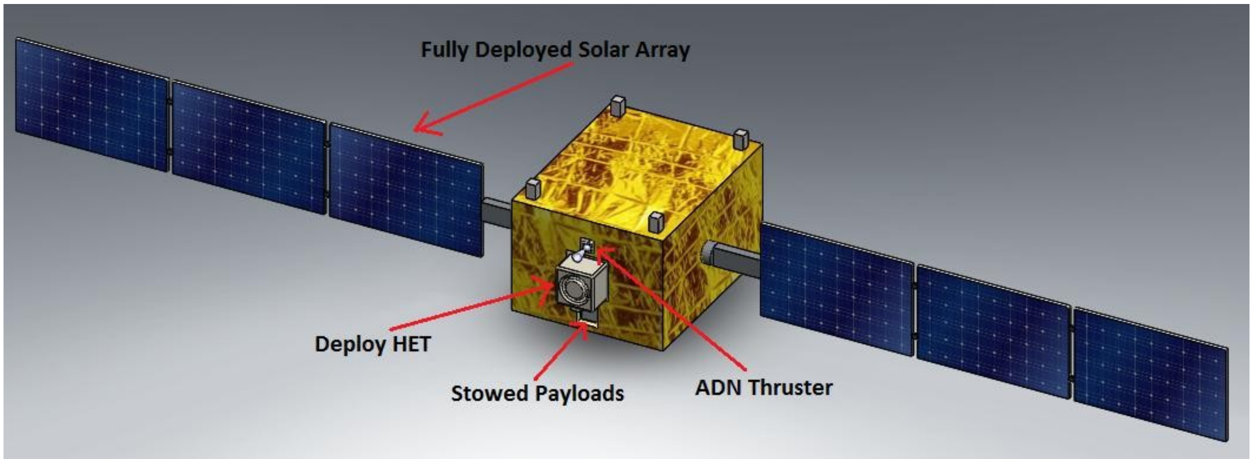


Figure 12: Isometric view of Deployed Concept Model [4]

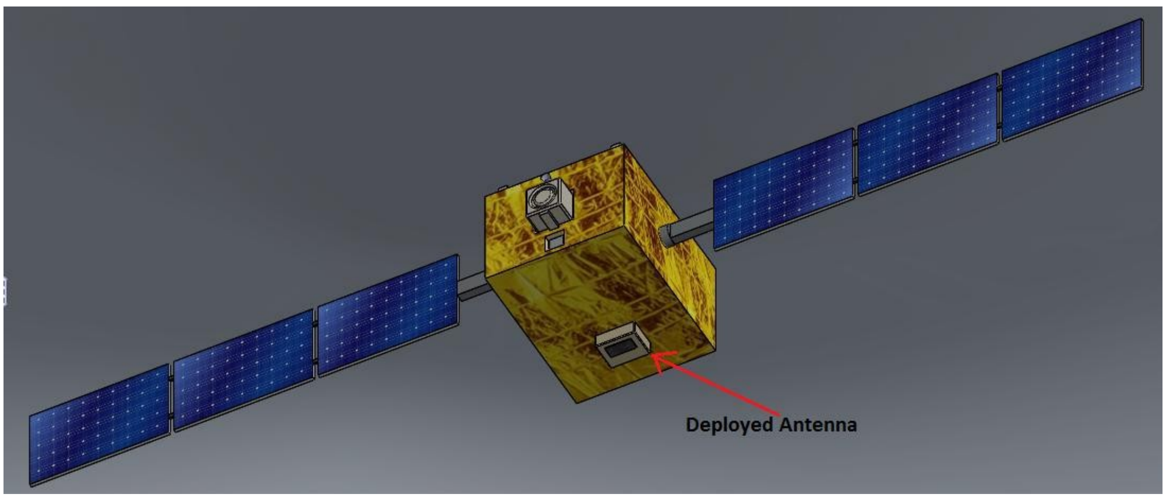


Figure 13: Deployed Antenna

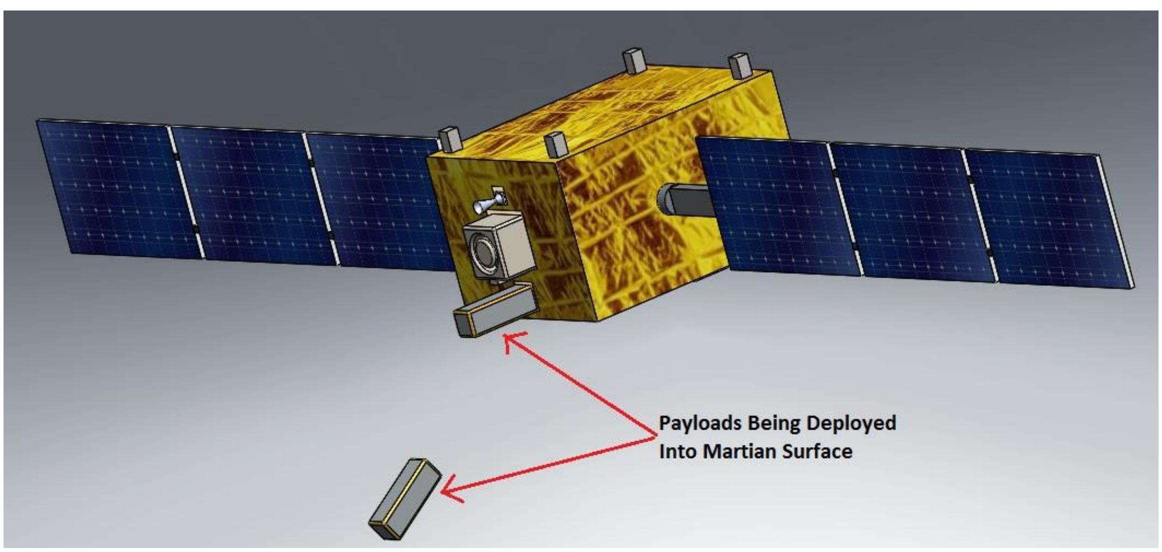


Figure 14: MMS Deploying Payloads on Mars [4]

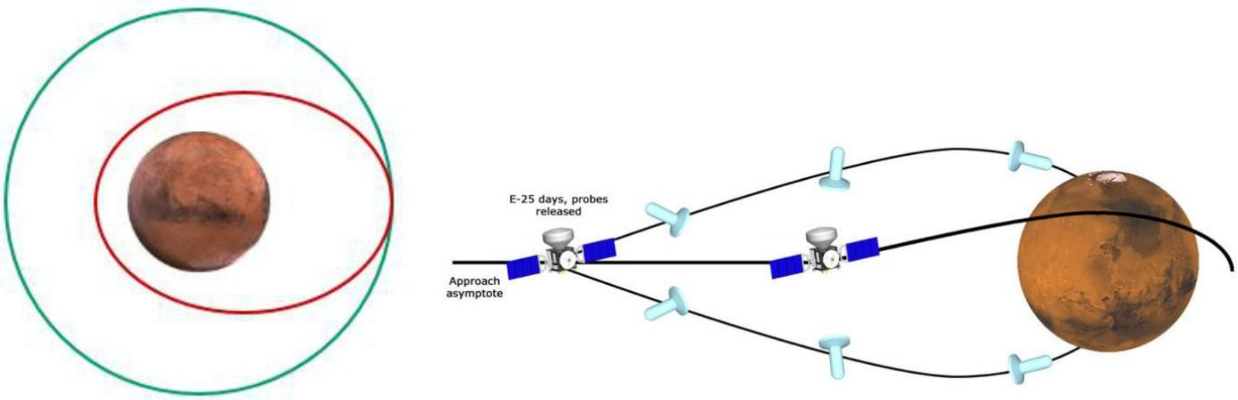


Figure 15: Representation of Payload Entering Mars Using Exo-Brake Technology [19]

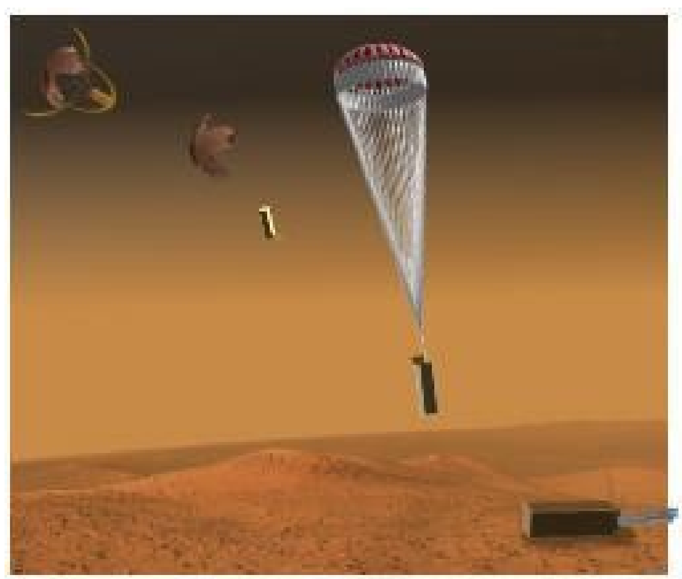


Figure 16:Representation of Mars Surface Science Payload Entering Mars [19]

Chapter 4

GMAT Simulation

This simulation uses primarily Hall Effect Thrusters to propel the satellite into a Mars transfer orbit and uses Mono-Propellant ADN green propellant thrusters to perform a Mars capture orbit. After performing a Mars capture orbit, the MMS will use the HET to reduce the altitude of the orbit to deliver the Mars surface science payloads into the Martian surface.

The initial mass parameters used for the simulation were the following:

Table 2: Initial Spacecraft Mass

Initial Spacecraft Mass	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	40.00
Xenon Propellant Mass	50.00
Total Spacecraft Mass	150.00

On Phase 1 of the mission, the MMS will use its HET to increase velocity to escape earth and go into a Mars Transfer Orbit. After performing this maneuver, the mass parameters for the spacecraft are the following:

Table 3: Spacecraft Mass after Phase 1 Maneuver

Spacecraft Mass After Phase 1 Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	40.00
Xenon Propellant Mass	12.23
Total Spacecraft Mass	112.23

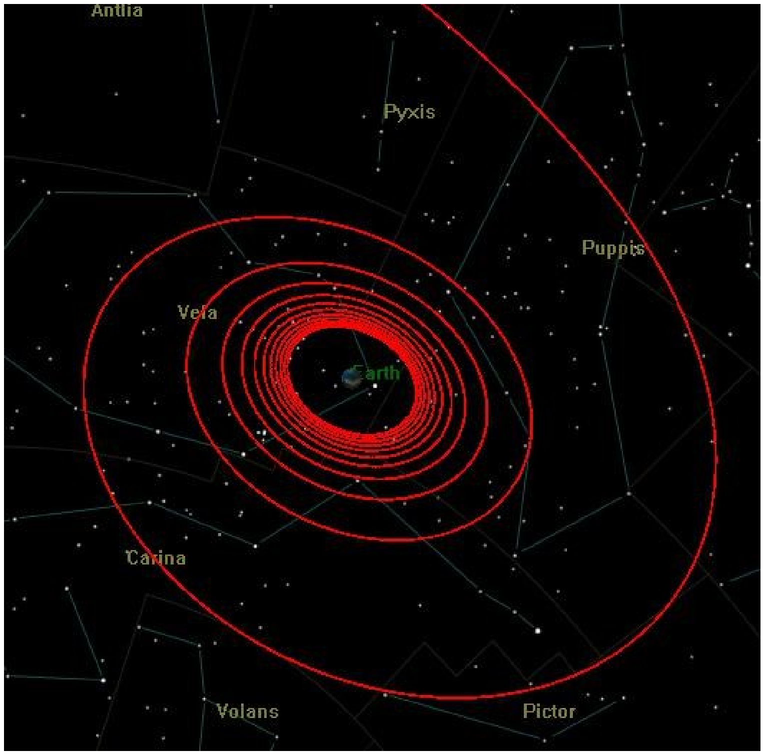


Figure 17:Orbit Transfer from GEO to MTO Using HET

The figure above is a representation of the MMS orbit when executing Phase 1 of the mission. The orbit increases from GEO, which is the starting orbit of the mission after being deployed from another GEO satellite. The orbit increases altitude and velocity when firing the HET in a circular orbit until reaching escape velocity. This represents that EP does not use Hohmann transfer maneuver to increase orbit like chemical propulsion would do.

After firing the HET for a little bit longer than a month in Phase 1 to reach the velocity required to Mars Transfer Orbit, the satellite will coast for approximately two weeks and then it will perform another HET firing in the opposite direction of the velocity vector for almost one week. This second HET firing during the MTO will help the MMS to reduce velocity will be defined as Phase 2 of the mission. After performing Phase 2 of the mission, the MMS mass parameters are the following:

Table 4: Spacecraft Mass After Phase 2 Maneuver

Spacecraft Mass After Phase 2 Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	40.00
Xenon Propellant Mass	8.55
Total Spacecraft Mass	108.55

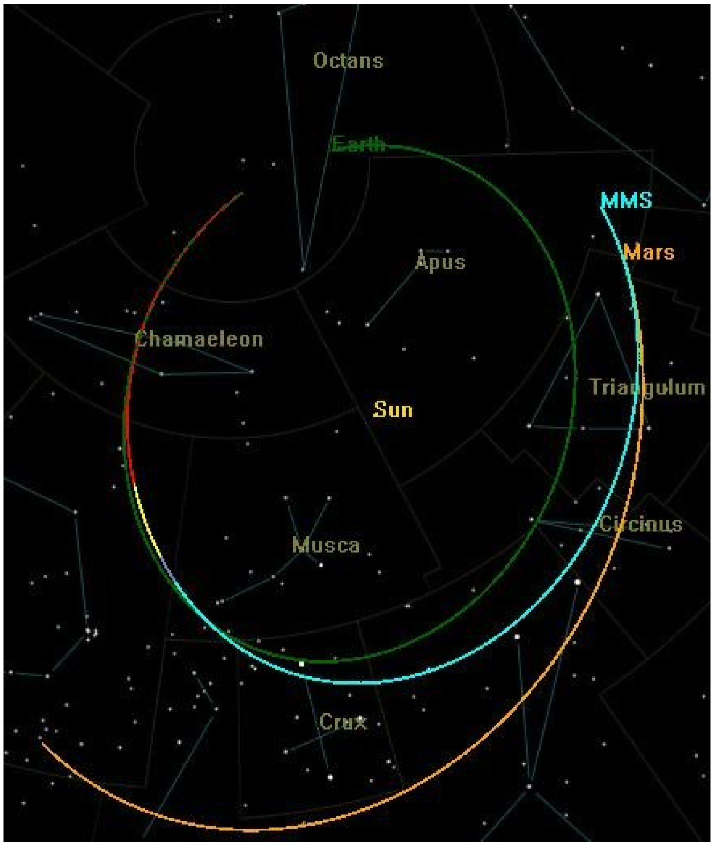


Figure 18: Orbit of MMS from GEO to MTO

The above figure is a representation of the complete orbit of the MMS from the GEO insertion orbit to MTO with respect to the sun as fixed point. Assuming a nominal MTO the MMS will approach mars and then execute several maneuvers to get into a Mars Capture Orbit. This is explained in detail in Chapter 5.

Chapter 5

Mars Capture Orbit Sequence of Events

Once the MMS reaches Mars, the MMS needs to perform a ΔV maneuver to transfer from MTO to MCO. To perform this maneuver the MMS will use its mono-propellant thruster for an instantaneous ΔV adjustment. Once the ΔV maneuver has been performed the MMS will continue to fire its HET to reduce the orbit altitude and descend to Mars to deliver two surface science stations. The payloads will be the size of a 3-U. After the payload has been delivered, the MMS will continue to orbit Mars and serve as a relay satellite to transmit information between the surface stations and earth. It will continue to utilize its HET for attitude control during this phase of the mission.

Since Mars is an average of 0.5 AU farther away to the Sun than the Earth, the solar flux power arriving to Mars is lower than Earth. This reduces the electric power produced by the solar arrays on MMS to approximate 40% of the beginning of life electric power produce on Earth orbit. The electric power on Mars orbit would be 1.95kw, which is still enough electric power to operate the HET.

Depending of the eccentricity of the orbit that the MMS wants to achieve, the MMS could perform different types of MCO maneuvers to get into Mars orbit, which they will be called scenarios in this paper. The differences between each of the maneuvers will be the change in eccentricity of the final orbit and periapsis altitude, which could be beneficial for the deployment of the two payloads into Mars surface.

5.1. Scenario 1: Low Eccentricity Orbit

5.1.1. Phase 3.A

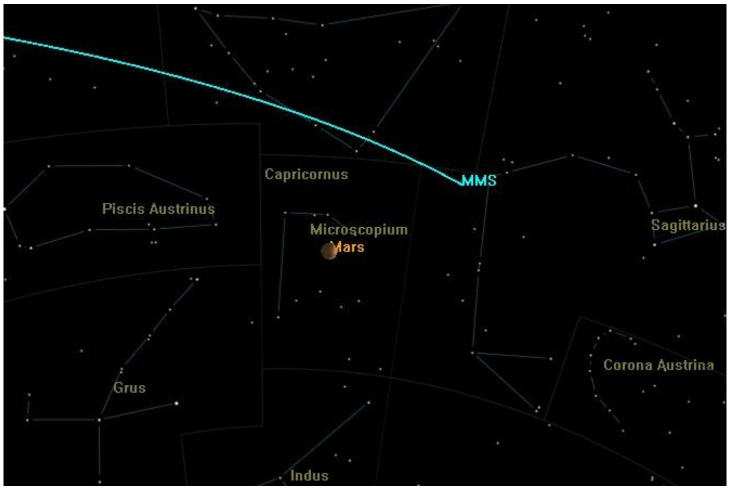


Figure 19: MMS ΔV Execution at Periapsis Phase 3.A

The above figure represents the moment at which the first ADN based ΔV maneuver of -0.5 km/s that will initiate a decesent toward Mars. This would be Phase 3.A of the mission. This change in ΔV will result in a low orbit eccentricity. The mass of the MMS and its propellant after Phase 3.A maneuver will be the following.

Table 5: Spacecraft Mass After Phase 3.A Maneuver

Spacecraft Mass After Phase 3.A Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	0.73
Xenon Propellant Mass	8.55
Total Spacecraft Mass	69.27

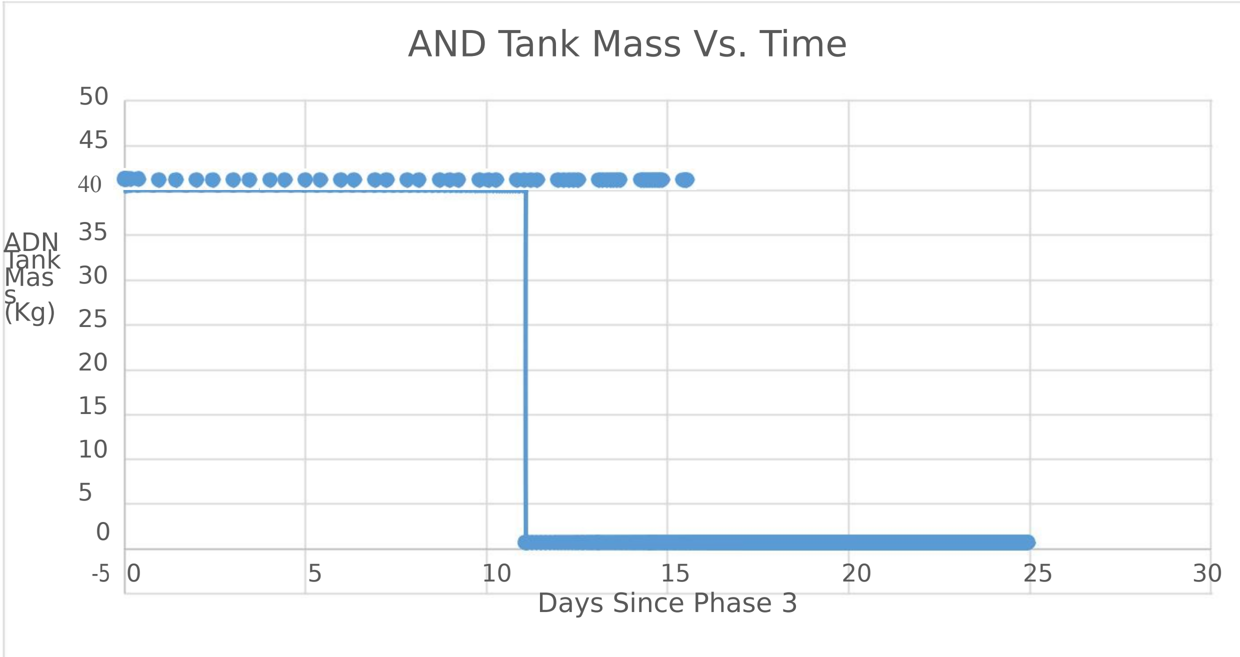


Figure 20: Days Vs. ADN Used in Scenario 1 Phase 3.A

5.1.2. Phase 4.A

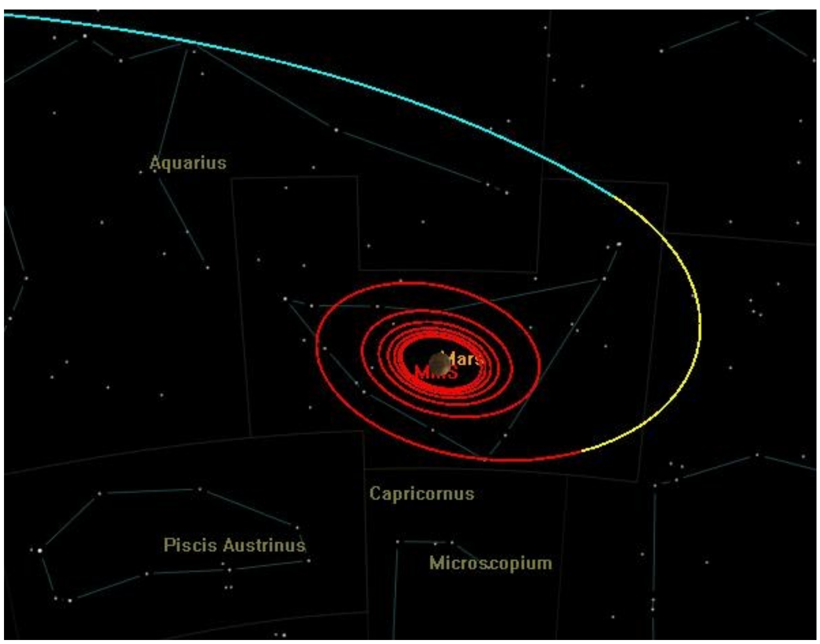


Figure 21: Profile of the MCO and HET Firing in Scenario 1, Phase 4.A

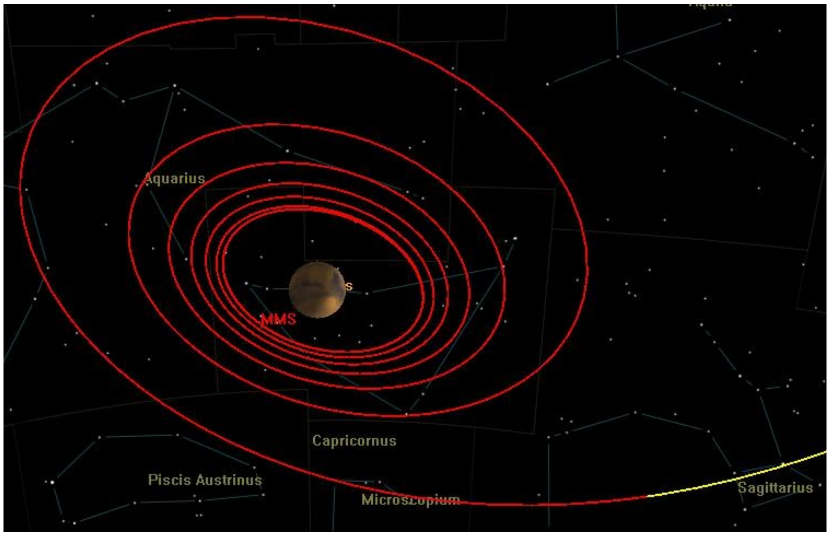


Figure 22: Mars Approach Close-up of Phase 4.A in Scenario 1

The above figures represents the MMS HET based maneuver to reduce its altitude with respect to Mars. The red color represents the HET active firings. This is will be Phase 4.A of the mission. After this Phase of the mission, the MMS will stay in Mars orbit and it will continue to use its HET to adjust the orbit altitude. Once the MMS has achieved the desired orbit altitude, the next Phase of the mission will be to deploy the two Mars surface science payloads, which it will be Phase 5 of the mission. The mass of the MMS and its propellant after Phase 4.A maneuver will be the following.

Table 6: Spacecraft Mass After Phase 4.A Maneuver

Spacecraft Mass After Phase 4.A Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	0.73
Xenon Propellant Mass	4.50
Total Spacecraft Mass	65.22

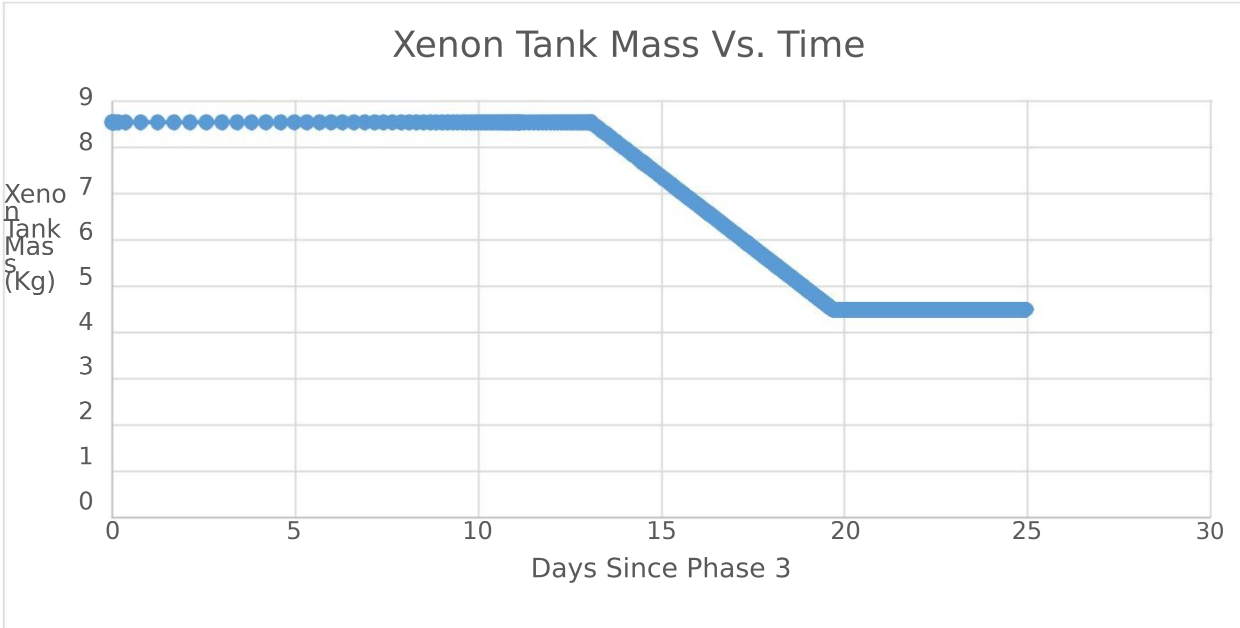


Figure 23: Time Vs. Xenon Propellant used in Scenario 1, Phase 4.A

The following two graphs show the mission profile in Scenario 1:

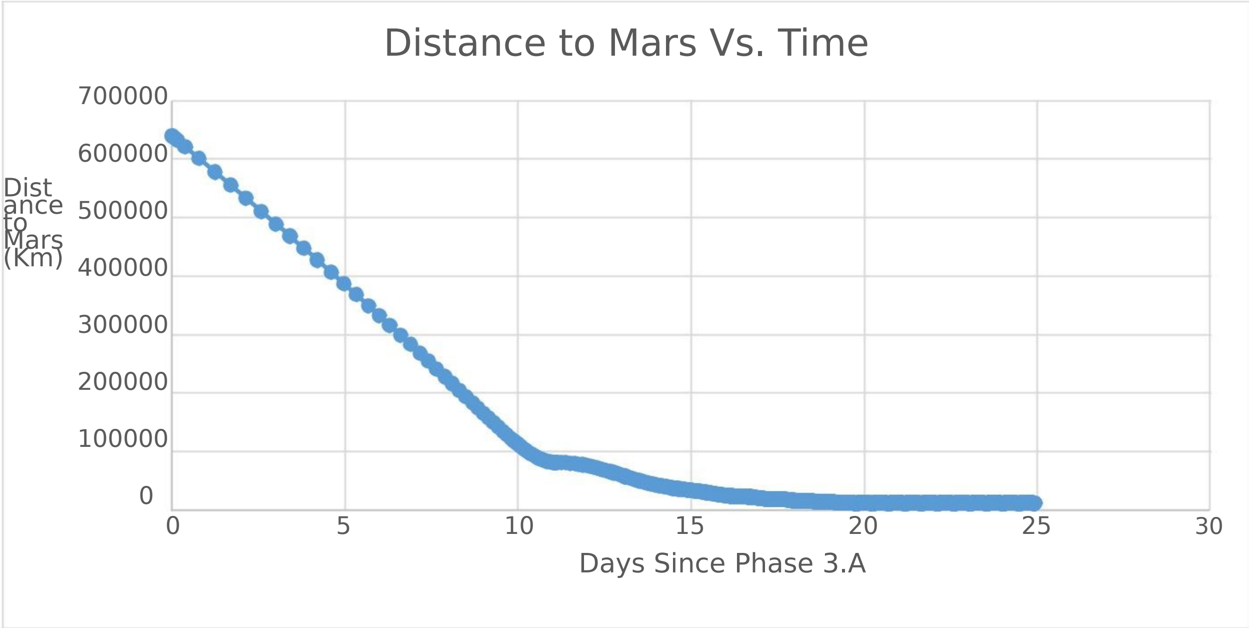


Figure 24:Time (Days) Vs. Distance to Mars Scenario 1

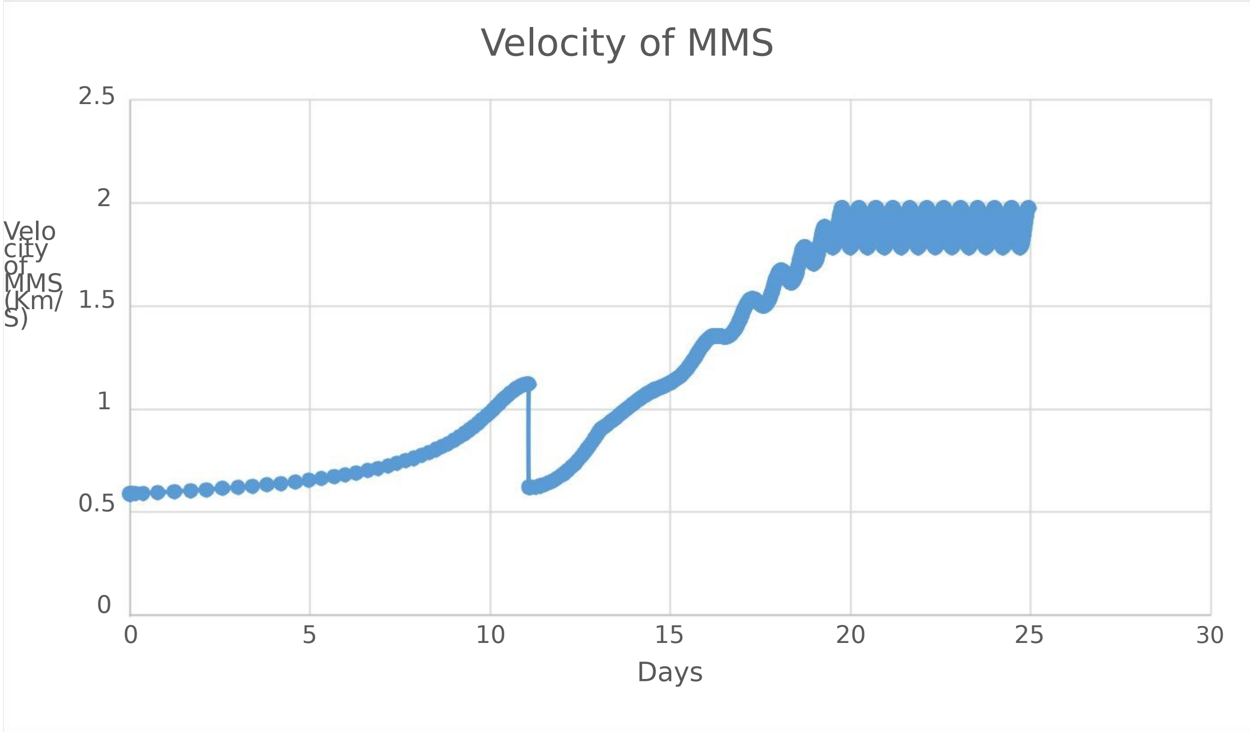


Figure 25: Time Vs. Velocity of MSS in Scenario 1

5.2. Scenario 2: High Eccentricity Orbit

5.2.1. Phase 3.B

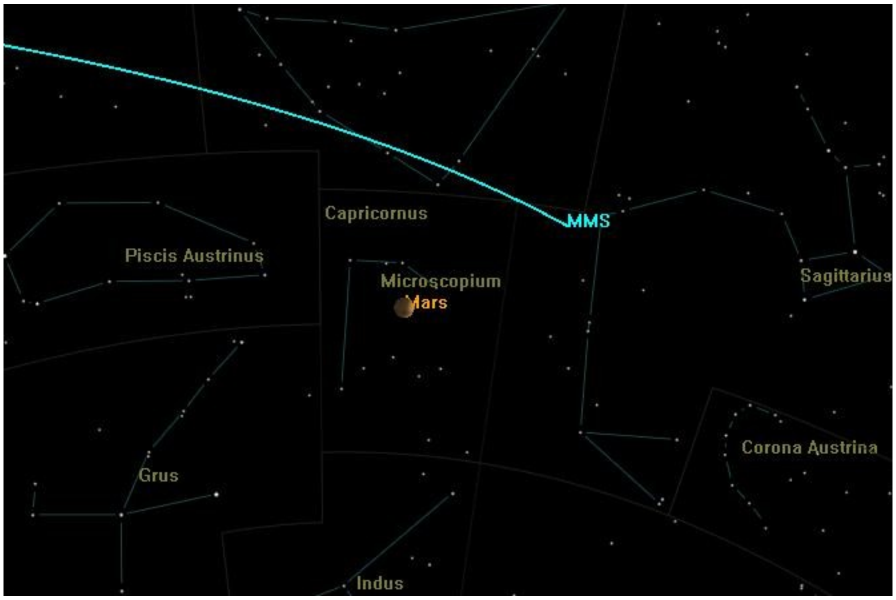


Figure 26:MMS at Mars Periapsis when Performs the ΔV Maneuver

The above figure represents the moment at which the first ADN based ΔV maneuver of -0.3 km/s that will initiate a decesent toward Mars. This will be Phase 3.B of the mission. This change in ΔV will result in a high orbit eccentricity, which will allow the MMS to deploy the payloads closer to Mars surface than scenario 1. The mass of the MMS and its propellant after Phase 3.B maneuver will be the following.

Table 7: Spacecraft Mass After Phase 3.B Maneuver

Spacecraft Mass After Phase 3.B Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	14.36
Xenon Propellant Mass	8.55
Total Spacecraft Mass	82.91

5.2.2. Phase 4.B

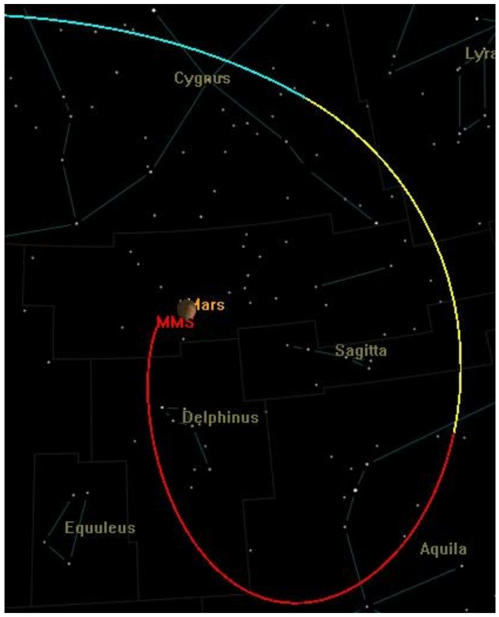


Figure 27: Firing of the HET to Reduce Altitude, Phase 4.B of Scenario 2

The above figures represents the MMS HET based maneuver to reduce its altitude with respect to Mars. The red color represents the HET active firings. This is will be Phase 4.B of the mission. After conducting this HET firing the MMS will be in a high eccentricity Mars orbit. The mass of the MMS and its propellant after Phase 4.B maneuver will be the following.

Table 8: Spacecraft Mass After Phase 4.B Maneuver

Spacecraft Mass After Phase 4.B Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	14.36
Xenon Propellant Mass	5.71
Total Spacecraft Mass	80.07

5.2.3. Phase 5.B

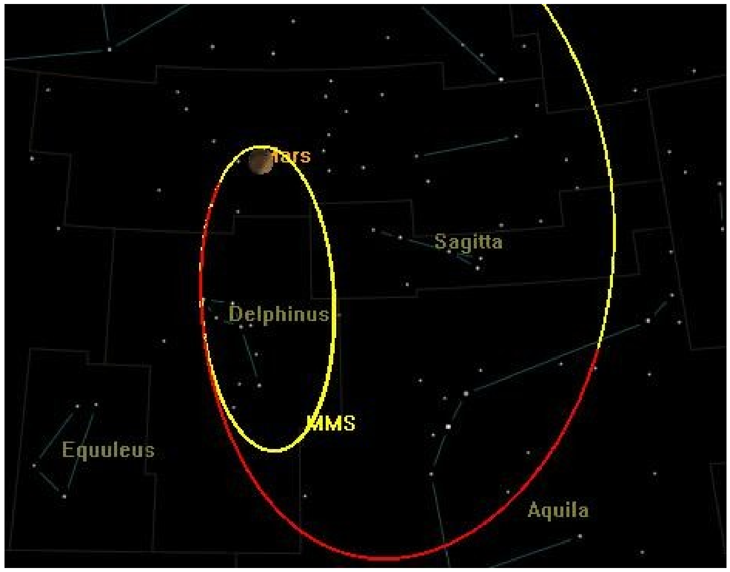


Figure 28: MMS Orbiting Mars with a High Eccentricity Orbit

The figure above represents the MMS on the high eccentricity Mars orbit achieved after Phase 4.B maneuver.

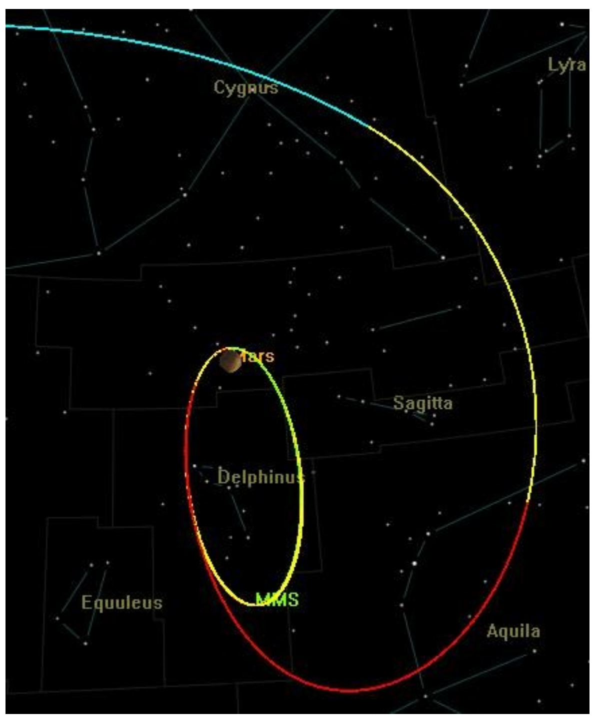
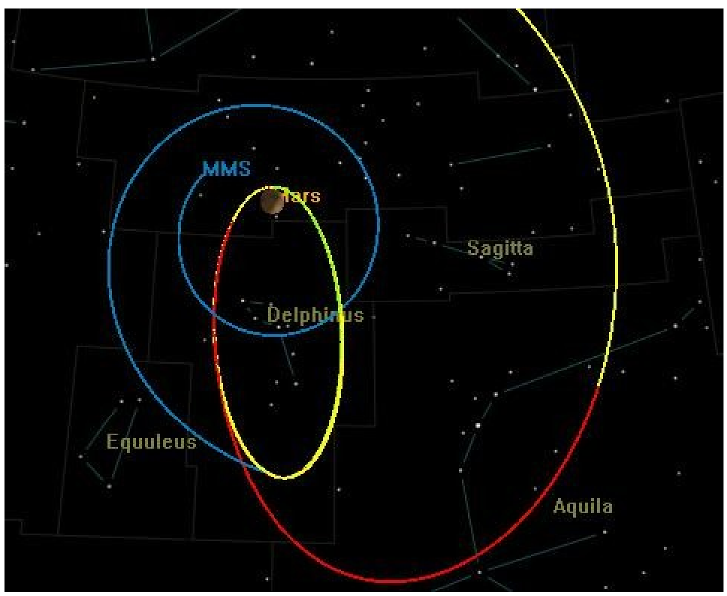


Figure 29: MSS at Mars Apoapsis to Reduce Eccentricity of Orbit, Phase 5.B ADN based Maneuver

After completing a couple of orbits, the MMS will approach it periapsis position and then deploy the payloads into Martian surface. After deploying the payloads, the MMS will execute Phase 5.B, which is another ADN based ΔV maneuver of +0.5km/s at apoapsis using to reduce the eccentricity of the orbit making it easier for the HET to take over and control the attitude of the MMS at a lower eccentricity orbit. The mass of the MMS and its propellant after Phase 5.B ADN based maneuver will be the following.

Table 9: Spacecraft Mass After Phase 5.B ADN Maneuver

Spacecraft Mass After Phase 5.B ADN Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	1.85
Xenon Propellant Mass	5.71
Total Spacecraft Mass	67.56



After performing the ΔV maneuver to decrease eccentricity of the orbit, as part of Phase5.B the HET starts firing to continue to reduce the eccentricity of the orbit and achieving a desired eccentricity and then continue to control the attitude of the satellite. The MMS can continue to adjust its orbit altitude and eccentricity using the HET onboard. The mass of the MMS and its propellant after Phase 5.B HET based maneuver will be the following.

Table 10: Spacecraft Mass After Phase 5.B HET Maneuver

Spacecraft Mass After Phase 5.B HET Maneuver	
	kg
Dry Mass	60.00
ADN Green Propellant Mass	1.85
Xenon Propellant Mass	3.55
Total Spacecraft Mass	65.40

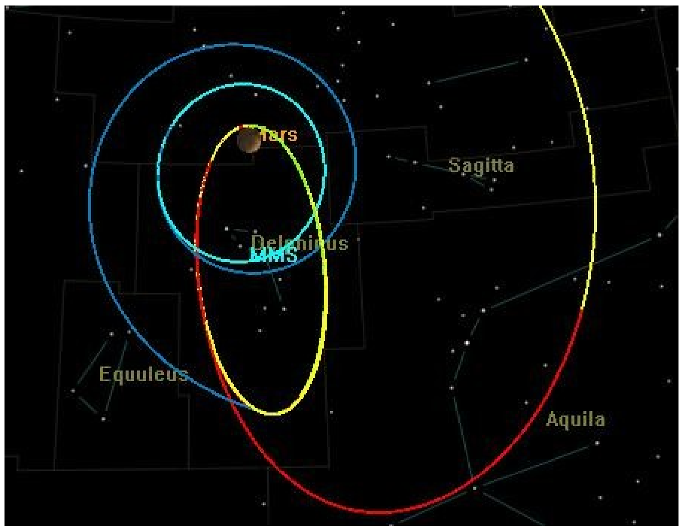


Figure 30: Final Orbit of the MMS at Lower eccentricity in Scenario 2

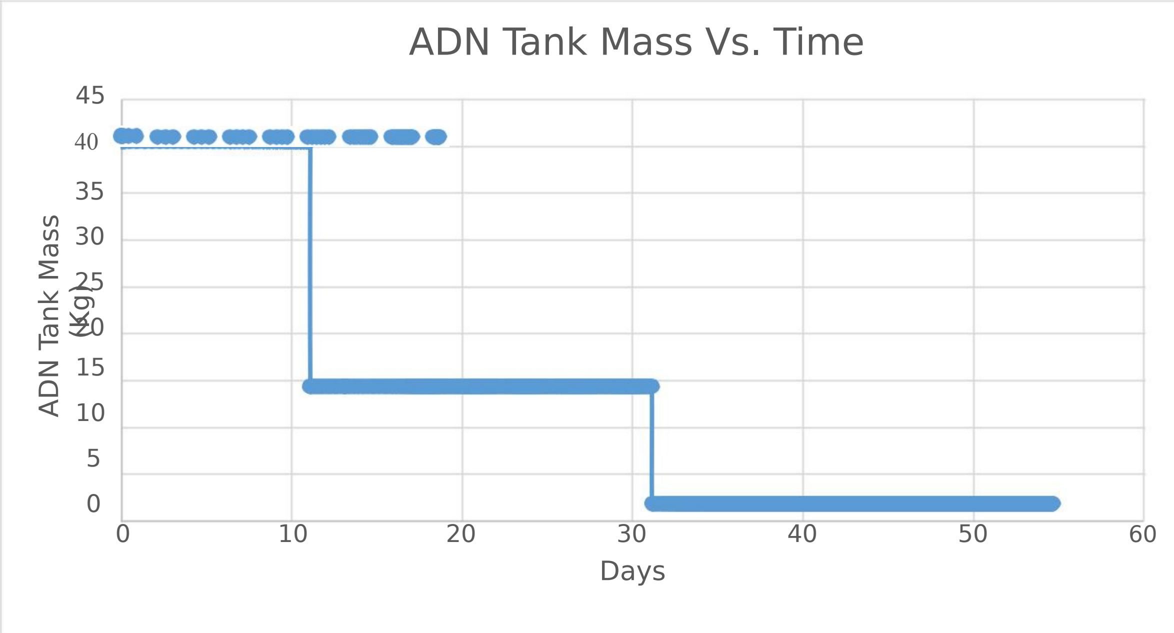


Figure 31: Time Vs. ADN Used in Scenario 2

The above graph is a representation of the ADN green propellant used during Scenario 2. The first drop on the graph is the maneuver executed in Phase 3.B and the second drop is the maneuver executed in phase 5.B

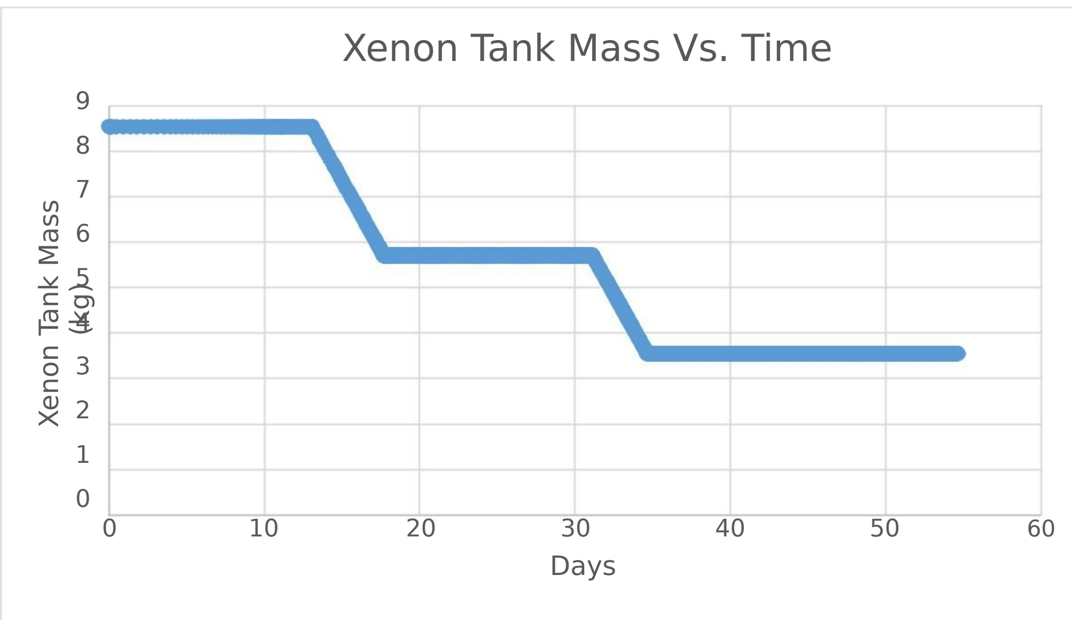


Figure 32: Time Vs. Xenon Propellant used in Scenario 2

The above graph is a representation of the Xenon propellant used during Scenario 2. The first drop on the graph is the maneuver executed in Phase 4.B and the second drop is the maneuver executed in phase 5.B

The following two graphs show the mission profile in Scenario 2:

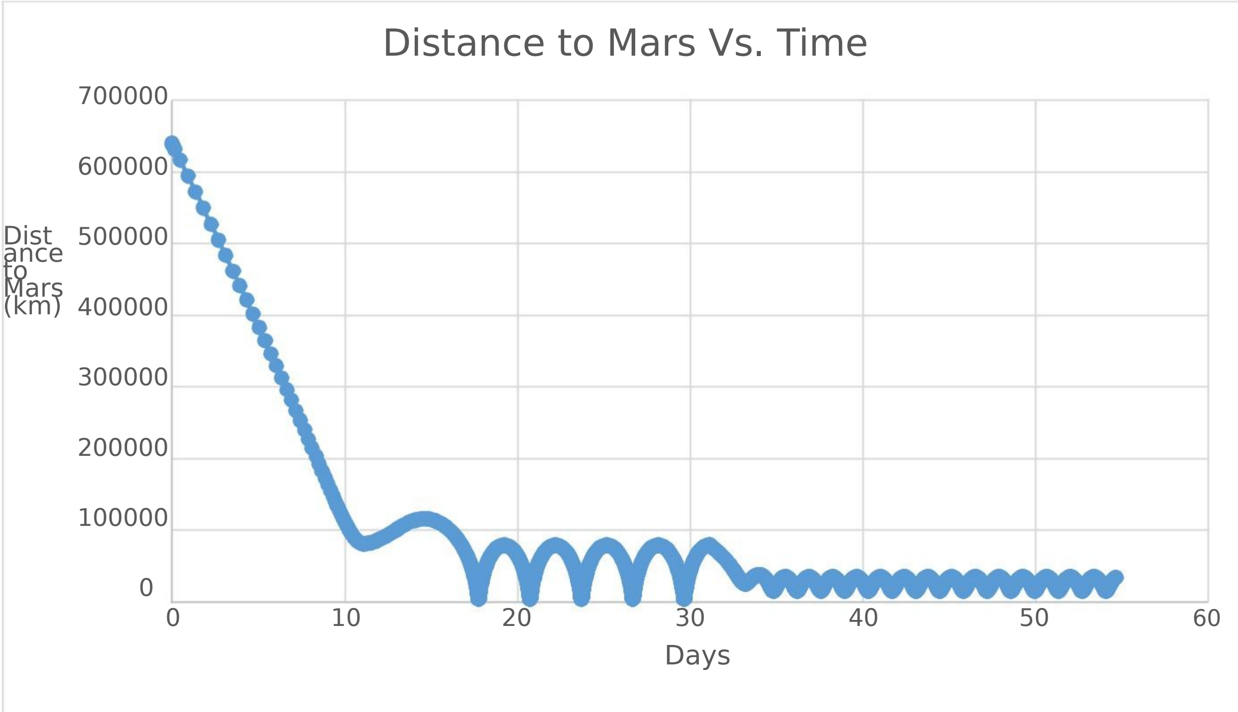


Figure 33:Time Vs. Distance to Mars Scenario 2

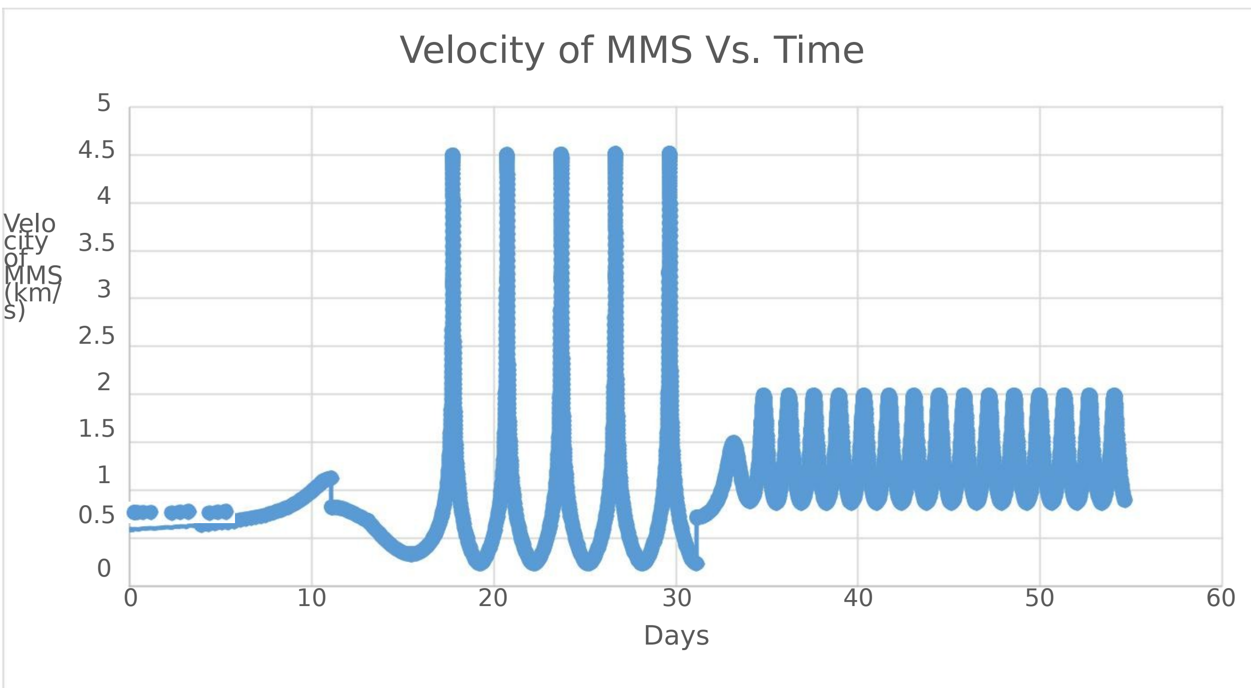


Figure 34: Time Vs. Velocity of MSS in Scenario 2

Conclusion

With the current technology and all the momentum that space exploration missions have, missions to Mars are starting to be a topic of conversation again. A new, low cost interplanetary capability is discussed with a Mars mission as a first example. This could be considered as an extension of the CubeSat concept in the sense that a spacecraft with well-defined interfaces that could be carried into a GEO orbit inside an empty compartment on a larger spacecraft (host) and then jettisoned once in orbit.

A Mars mission in which a 150-kg electric-propulsion based spacecraft is carried in the empty compartment on the host spacecraft is studied as well as the feasibility of taking such spacecraft from GEO to Mars orbit using a combination of electric and chemical propulsion, such as existing Hall Effect Thrusters and ADN green propellant, which could potentially save money spent on chemical propellants, and reduces propellant mass that could otherwise be used to carry larger payloads. The NASA trajectory code GMAT rev.2016a is used to define and solve the trajectory. Further refinement is considered that changes the mission profile with varying the use of the ADN mono-propellant to perform first mission optimization steps during the critical MCO maneuvers. When captured at Mars orbit, the MMS can deploy two Nano-Satellites surface landers (currently studied by NASA Ames) in support of Climatology and other missions. Also, after the MMS completes the payload deployment phase, the MMS can serve as an additional communication relay.

In summary, the proposed study can help extend idea of the ride-share mission concept by the utilizing unused volume and cargo on SSL GEO satellites. While Mars is a first and attractive choice, this can be further extended to other interplanetary bodies in the solar system.

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Appendix

Datasheets

Solar Cells Datasheet

ZTJ Space Solar Cell

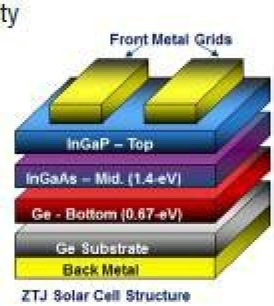
3rd Generation Triple-Junction Solar Cell for Space Applications

29.5%
Minimum Average Efficiency

Space Qualified & Characterized to the AIAA-S111-2005 & AIAA-S112-2005 Standards

FEATURES & CHARACTERISTICS

- 3rd generation triple-junction (ZTJ) InGaP/InGaAs/Ge Solar Cells with n-on-p polarity
- Solar cell mass of 84 mg/cm^2
- Fully space-qualified with proven large volume manufacturing and flight heritage
- Excellent radiation resistance with P/Po = 0.90 @ 1-MeV, 5E14 e/cm^2 fluence
- Compatible with corner-mounted silicon bypass diode for individual cell reverse bias protection
- Excellent mechanical strength for reduced attrition during assembly and laydown
- Weldable or solderable contacts
- Custom sizes available
- Available as a Coverglass Interconnected Cell (CIC) for integration onto solar panels



ZTJ Solar Cell Structure

Typical Performance Data

Electrical Parameters @ AM0 (135.3 mW/cm²)	
BOL Efficiency at Maximum Power Point	29.5%
Voc (V)	2.726
Jsc (mA/cm²)	17.4
Vmp (V)	2.41
Jmp (mA/cm²)	16.5

Temperature Coefficients

Fluence (e/cm²)	Voc (mV/°C)	Jsc[1] (µA/cm²-°C)	Jmp[2] (µA/cm²-°C)	Vmp (mv/°C)	Pmp (µW/cm² -°C)
0	-6.3	11.7	9.1	-6.7	-85.7
1.00 E+ 14	-6.6	11.4	9.1	-7.0	-92.3
1.00 E+ 15	-6.9	11.3	10.6	-7.3	-89.9
1.00 E+ 16	-7.4	11.5	13.4	-6.6	-57.2

(1) Jsc is the symbol for normalized Isc (2) Jmp is the symbol for normalized Imp

Radiation Performance at 1 MeV Electron Irradiation, EOL/BOL Ratios

Fluence (e/cm²)	Voc	Isc	Vmp	Imp	Pmp[1]
3.00 E+ 13	0.96	0.99	0.98	0.99	0.99
1.00 E+ 14	0.95	0.98	0.97	0.99	0.96
5.00 E+ 14	0.91	0.97	0.93	0.96	0.90
1.00 E+ 15	0.89	0.94	0.91	0.94	0.85
3.00 E+ 15	0.86	0.89	0.87	0.86	0.75
1.00 E + 16	0.82	0.82	0.83	0.74	0.62

(1) Per AIAA-S-111 Standards

Key Space Qualification Results

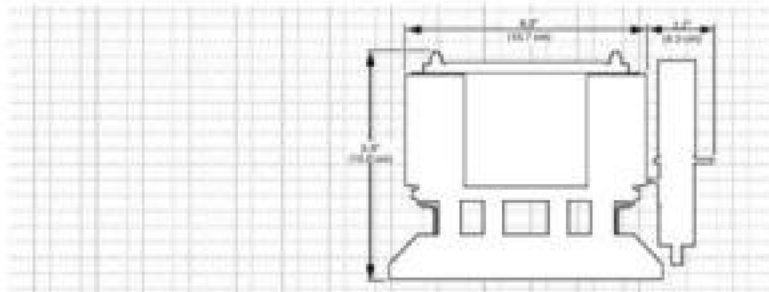
Test Performed	Industry Quality Standard	Typical Test Results
Metal Contact Thickness	4-8 µm	6 µm
Dark Current Degradation after reverse bias	ΔIspec<2%	<0.4%
Electrical Performance after 2,000 thermal cycles -180°C to +95°C	<2%	No Change
Contact Pull Strength	>300 grams	>600 grams
Electrical Performance Degradation after 40 day humidity exposure at 60°C and 95% relative humidity	<1.5%	No measurable difference

MADE IN USA ISO 9001 QUALITY ASSURANCE

http://solaerotech.com/wp-content/uploads/2016/10/ZTJ-Datasheet-Updated-2016.pdf

Hall Effect Thruster Datasheet

BPT-2000 Hall Effect Thruster



Design Characteristics

- Propellant .... Xenon
- Mass (Thruster & Cathode) .... <5.2 kg
- Envelope Dimensions .... 15 x 17 x 22 cm
- Nominal Input Power .... 2200 Watt
- Operational Power Range .... 1200 – 2700 Watt
- Nominal Voltage .... 350 Volt
- Operational Voltage Range .... 250 – 400 Volt

Performance at 2.2 kW

- Thrust .... 123 mN
- Specific Impulse* .... 1765 sec
- Efficiency* 48%
- Life (Continuous)** .... >6000 hr
- Total IMpulse .... $>2.6 \times 10^6$ N-sec
- Nominal Flowrate .... 7.1 mg/sec
- On/Off Cycles .... 6000 cycles

Status

- Flight Prototype Unit Fabricated and Tested

* Corrected for facility pressure effects
** Based on accel life tests and analysis

AEROJET

Approved for public release and export

http://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/Electric%20Propulsion%20Data%20Sheets.pdf

GMAT Simulation Parameters

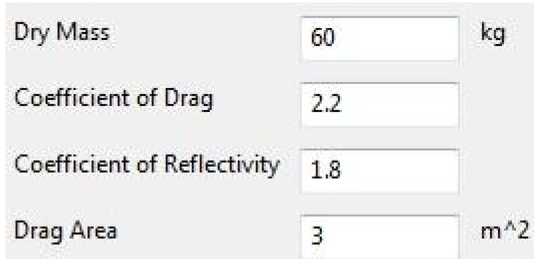


Figure 35: Ballistic and Mass Parameters

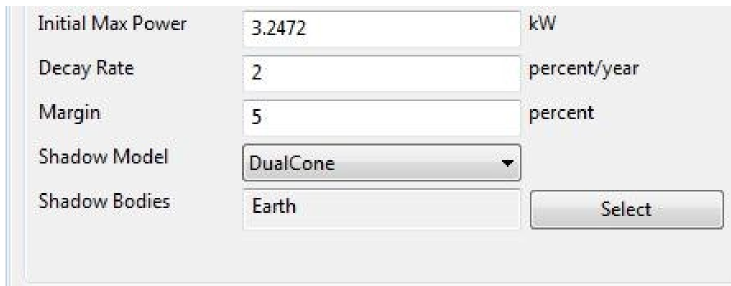
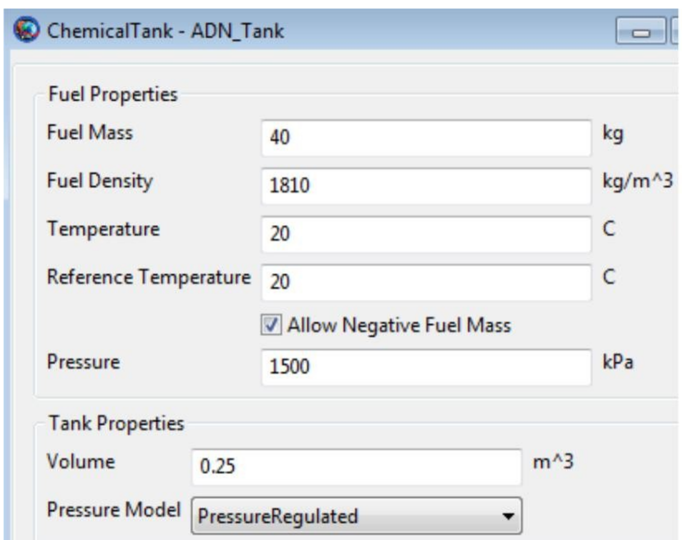


Figure 36: Electric Power System Parameters



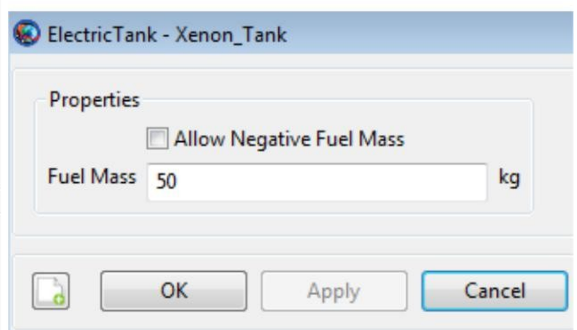
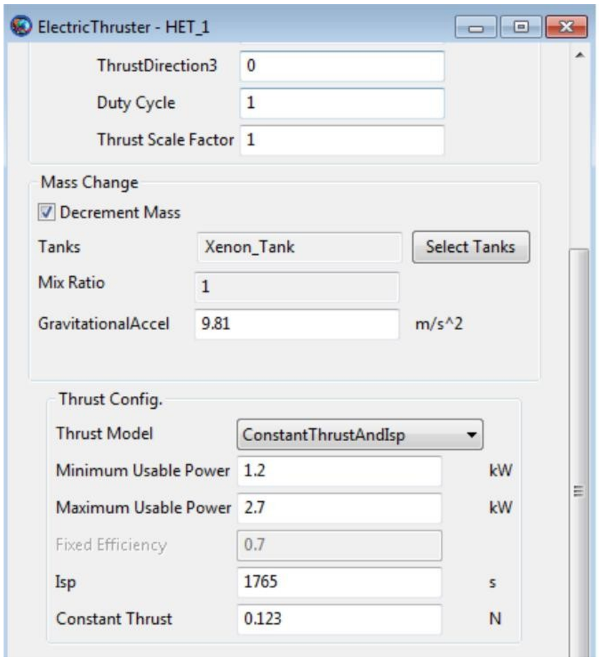


Figure 37: Mono-Propellant Tank Parameters

Figure 38: Xenon Tank

Parameters



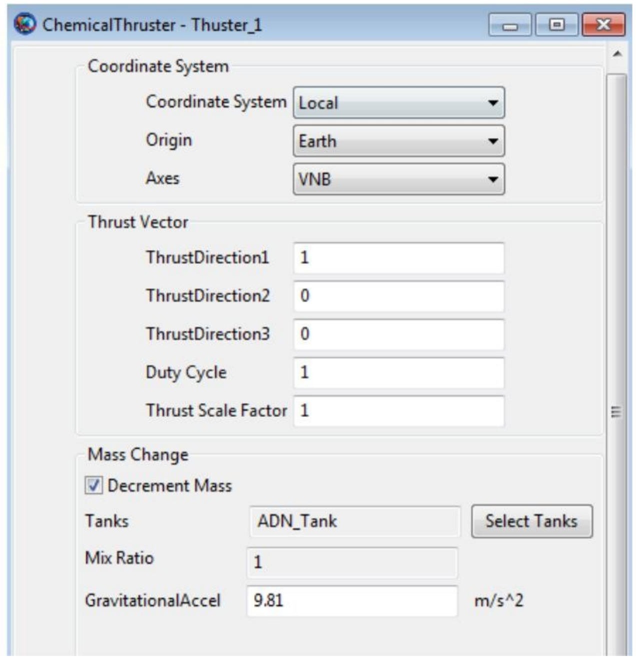


Figure 39: HET Thruster Parameters

Figure 40: Mono-Propellant Thruster Parameters

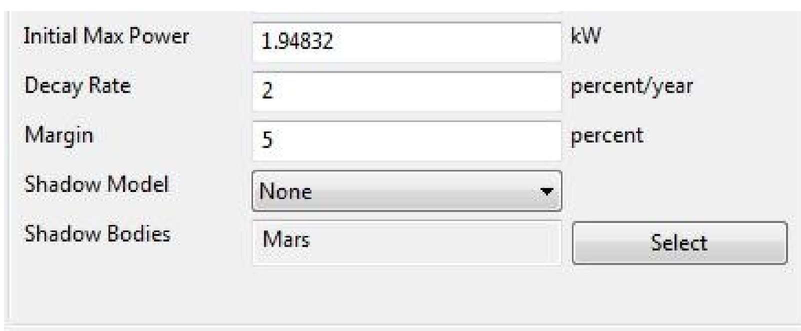


Figure 41: EPS at Mars Orbit