NASA/CR—2003-212350

Comparison of Mars Aircraft Propulsion Systems

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May 2003

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Introduction

For all aircraft, the characteristics and capabilities of the propulsion system are key elements in establishing the aircraft's feasibility and flight envelope. This is especially true for an aircraft that is to fly on Mars. The Martian environment, as well as the launch from earth and transit in deep space, produces additional obstacles to an aircraft's performance capabilities. Aircraft volume is critical since the aircraft must be stowed and fit into an aeroshell capsule for transit and entry into the Mars atmosphere. Therefore propulsion system volume is also critical and any components (such as a propeller) that are large would also need to be stowed.

Because of the thin atmosphere on Mars, where the surface atmospheric density is comparable to that at 30 km (110 kft) here on Earth, the ability to generate lift is fairly difficult. To compensate for this thin atmosphere, the aircraft would need to fly fast and limit its mass as much as possible. Therefore minimizing propulsion system mass is a critical factor in enhancing the capabilities of the aircraft. Also due to the high speed at which the aircraft will need to travel and the rocky surface of Mars, landing the aircraft is not a realistic option. This means then that the aircraft will be able to fly only as long as the propulsion system can operate. Since mission duration will depend on the efficiency of the propulsion system, the more efficient and lightweight the system is the longer the mission and the more capable the aircraft.

Because of this critical impact that the propulsion system has on the aircraft's capabilities as well as mission viability, a detailed examination of various propulsion systems was performed to determine what type of system is most applicable for a desired mission duration. Three main types of propulsion systems were evaluated, each with various sub types of systems. The systems examined are listed below.

- Electrical Propulsion Systems
    - Fuel Cell Power Propulsion System
    - Battery Powered Propulsion System
- Combustion Engine Systems
    - Piston Expander Combustion Engine Propulsion System
    - 4 Cycle Internal Combustion Engine Propulsion System
    - 2 Cycle Internal Combustion Engine Propulsion System
- Rocket Systems
    - Bipropellant Rocket Propulsion System
    - Monopropellant Rocket Propulsion System

Each of these systems was evaluated to determine its mass and performance characteristics for three mission durations. The mission durations chosen were 1 hour, 2 hours and 4 hours. These durations provide a fairly large range in aircraft capabilities. The thrust level that the propulsion systems were designed to was 35N. This thrust level was based on an estimate for an aircraft with an approximate 5m wingspan and was deemed representative of the thrust level requirements for proposed near-term Mars aircraft.

Electric Propulsion System

The first step in sizing an electric power system to provide propulsion power for the aircraft is to determine the total amount of power needed to produce the desired amount of thrust. Simplistically this power (P) required by the power system is equal to the thrust (T) desired times the flight velocity (V) divided by the system efficiency (η_t).

$$P= TV/\eta_t \tag{1}$$

For a basic electric system the efficiency is given by the following.

$$\eta_t = \eta_e\, \eta_m\, \eta_g\, \eta_p \tag{2}$$

Equation 2 represents the efficiencies of the main power system components, which include the electronic controls, drive motor, gearbox and propeller. The operational efficiency associated with each of these components is given in table 1. These efficiencies are representative approximations, which are subject to change based on a more detailed system and component design.

Component	Efficiency
Control Electronics	η_e 0.98
Motor	η_m 0.90
Gearbox	η_g 0.85
Propeller	η_p 0.85
Total System Efficiency	η_t 0.637

Table 1 Component Efficiencies for Electric Powered Aircraft

Based on the efficiencies listed in table 1, in conjunction with a flight speed of 130 m/s and a thrust level of 35 N, the total power required is 7142 W. To increase the system performance and reduce the power consumption there is the possibility of eliminating the gearbox from the system. However, this would require that the motor spin at the RPM of the propeller. If the propeller is large this RPM would be low (under 1000) and it may be difficult to find a motor that can achieve this while still maintaining a high efficiency.

Electric Motor and Controller

The power level which the electric motor must operate at is about twice that of high-end power levels of present day model aircraft. Using the high power electric motors for model aircraft as well as some larger (much higher power) motors for electric vehicles, a scaling factor of 1291 W/kg was produced for the motor and 6233 W/kg for the motor controller. Based on these scaling values the motor and motor controller masses would be 5.52 kg and 1.15 kg respectively.

Aside from the overall mass of the motor there are some additional factors that must be considered when evaluating which motor to use. The atmosphere of Mars is very thin (similar to the atmospheric pressure at 30 km altitude on Earth) and composed mostly of CO_2. Because of this thin atmosphere, arcing from electrical devices can become a serious problem. Regardless of the type of motor is desired, testing within environmental conditions that simulate the Mars atmosphere would be necessary to insure that arcing will not occur. The two main types of electric motors that can be used for this type of application are a brushless DC motor and a brushed DC motor. A brushless motor reduces the risk of arcing that may be seen with a brushed motor while operating at high power levels within a rarefied atmosphere, while a brushed motor can produce better efficiency at higher power levels (current levels).

Another issue is the cooling that the motor and motor controller may require. Cooling is particularly a concern due to the very thin Martian atmosphere and the extended period of operation at high power levels. Typical model aircraft motors are very lightweight but are designed to operate for short periods of time (on the order of 10s of minutes). Because they are so lightweight they do not have a lot of thermal mass. Therefore they will tend to heat up fairly rapidly and require significant convective cooling to keep them from overheating. Normally this would come from the air passing over the motor. However, in this application it will be operating within the Martian atmosphere and the convective cooling by the atmosphere may not be sufficient. A potential solution would be to place a sleeve over the motor that has fins for enhancing the heat transfer to the air stream or to use a liquid cooling system to keep the motor cool. Either of these approaches has a mass and drag effect on the aircraft. Another issue is the structural integrity of a very small motor connected to a fairly large propeller. There can be significant bending loads placed on the drive shaft. This increases the risk of a mechanical failure of the motor.

Gearbox

The function of the gearbox is to step down the shaft RPM from the motor speed to the desired propeller RPM. Based on this initial analysis it seems that a 10 to 1 reduction ratio will be needed. This will take the motor RPM of about 10,000 down to the propeller RPM of about 1,000. The 10 to 1 reduction is achievable in a single step. This is about as high a reduction that can be reliably achieved with a single step down. Based on some lightweight gearboxes that have been previously built for aircraft applications a scaling number of 0.305 kg/kw for the mass as a function of power transmitted was devised. The power transmitted by the gearbox is the power required to fly (thrust multiplied by velocity: 4550 W) divided by the propeller efficiency (85%). This yields a gearbox mass of 1.63 kg.

Propeller

The propeller design was done in order to get an estimate of the propeller size and mass needed to produce the thrust required by the aircraft. The design was not optimized and the optimal geometry can differ from what is presented here. However, this gives a starting point from which to build and represents one configuration that should be capable of meeting the mission needs.

For this preliminary design a vortex theory code with low Reynolds number corrections was used to generate the estimated performance [1]. The airfoil selected for the propeller was the SD8000-PT low Reynolds number airfoil [2]. This airfoil was chosen because of its very good lift-to-drag characteristics at low Reynolds numbers as well as its post stall lift generation capability. A curve of airfoil lift coefficient (Cl) vs. drag coefficient (Cd) and lift coefficient (Cl) vs. angle of attack (alpha), at a Reynolds number of 60,000 are shown in figures 1 and 2. The geometry of the airfoil is shown in figure 3.

The propeller was assumed to be a fixed pitch propeller. This was done to reduce the complexity of the system. However with a fixed pitch propeller some capability and performance had to be sacrificed. It has to be determined what the best operating blade angle is for the propeller. The blade angle is akin to various gearing ratios in a car. The higher the blade angle (or gear) the lower the operational RPM (for a given thrust) but the greater the torque required. The geometry of the propeller, which includes the twist and the variation of chord with radius, is shown in figure 4. Since the blade is twisted the pitch angle will vary along the blade length. Therefore, the pitch angle specified for the blade represents the angle the blade is at the 3/4 radius point. For this design the pitch angle was 38°. The propeller has six blades and the diameter was chosen to be 2.8 m. This diameter and blade number was sufficient to achieve the thrust level of 35N at 1 km altitude on Mars.

Determining the operating point meant selecting the blade angle, RPM, power required and thrust generated. The selection of these parameters will also influence the capabilities of the other components of the system, such as motor RPM and power available. Propeller performance curves were generated for the geometry described. These performance curves are shown in figure 5. This configuration requires 5353 watts of shaft power, produces an efficiency of 85% and has an operational RPM 1172.

Based on the geometry specified and using a hollow carbon section for the propeller, the mass of each blade could be calculated. The mass of each blade is 1.63 kg. Therefore the total propeller mass is 9.8 kg.

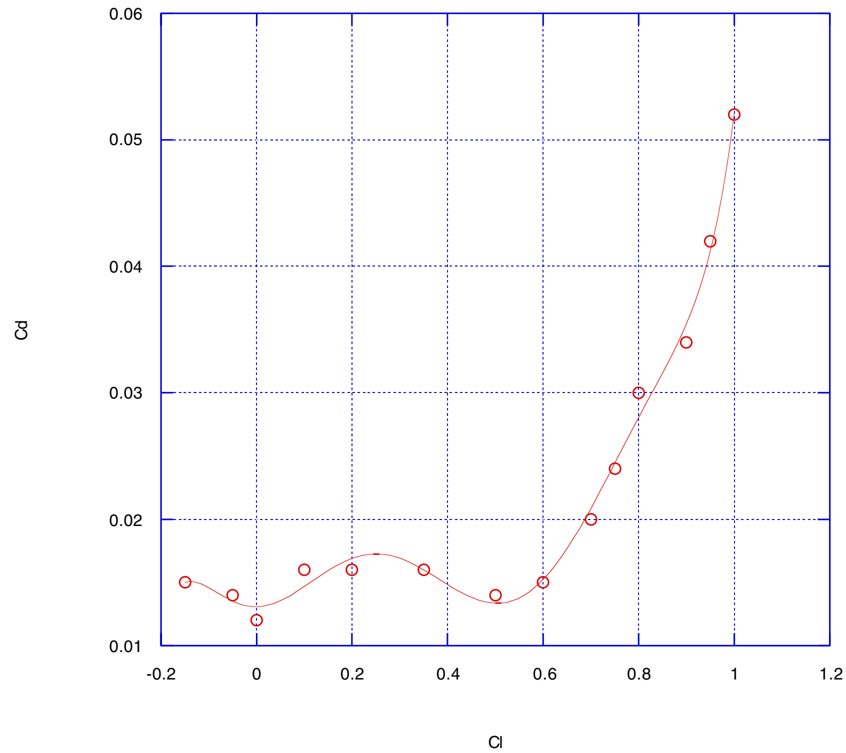

Figure 1 Cl versus Cd for SD8000-PT Airfoil at 60,000 Re [2]

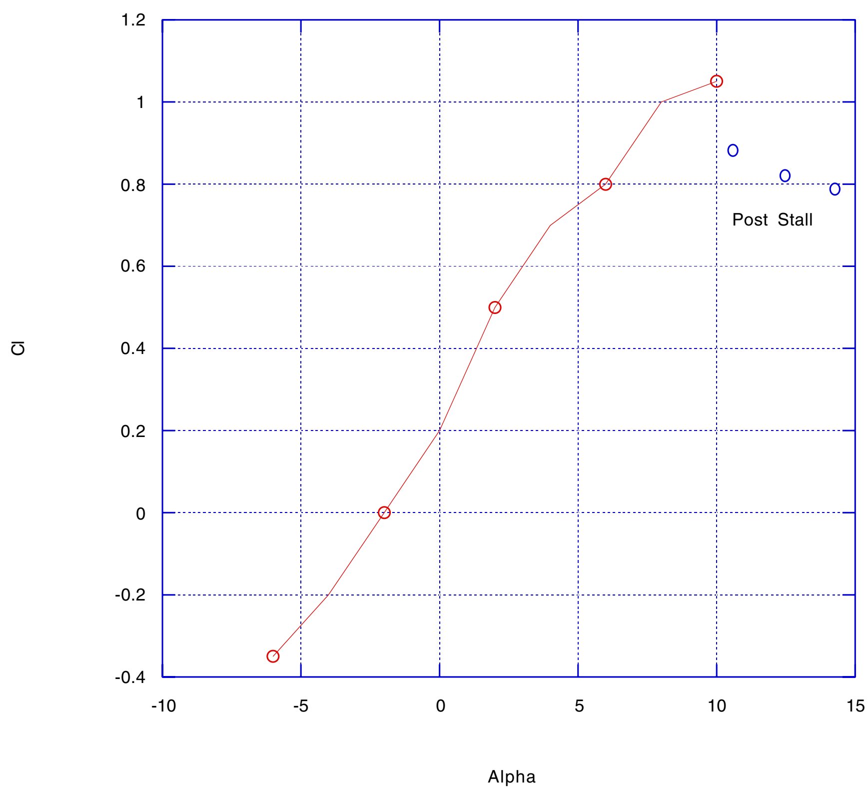

Figure 2 Cl versus Alpha for SD8000-PT Airfoil at 60,000 Re [2]

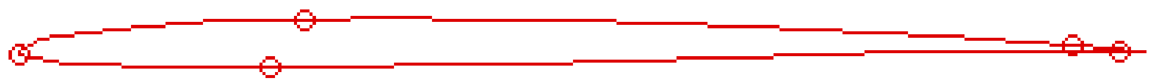

Figure 3 SD8000-PT Airfoil Cross Section [2]

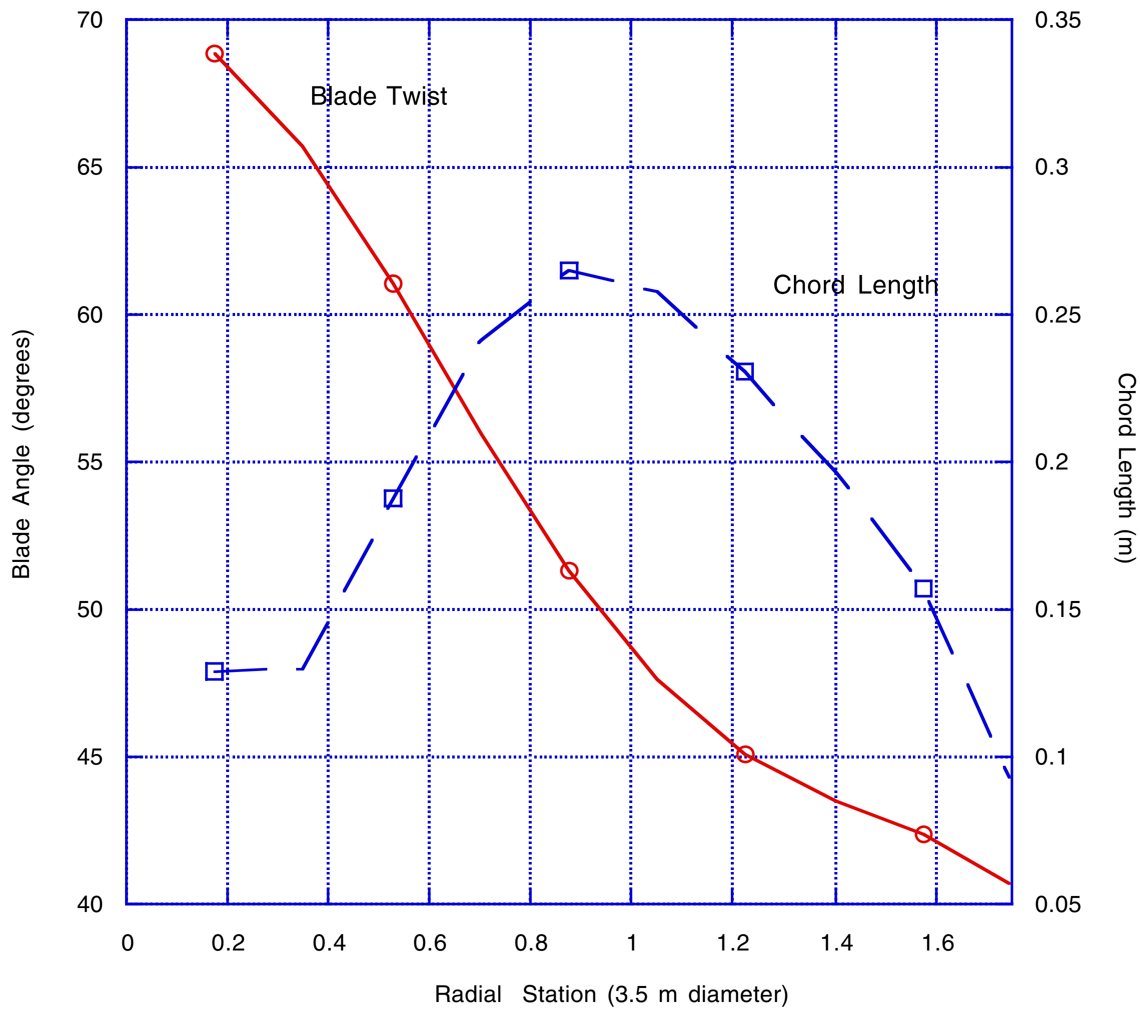

Figure 4 Propeller Blade Twist and Chord Length

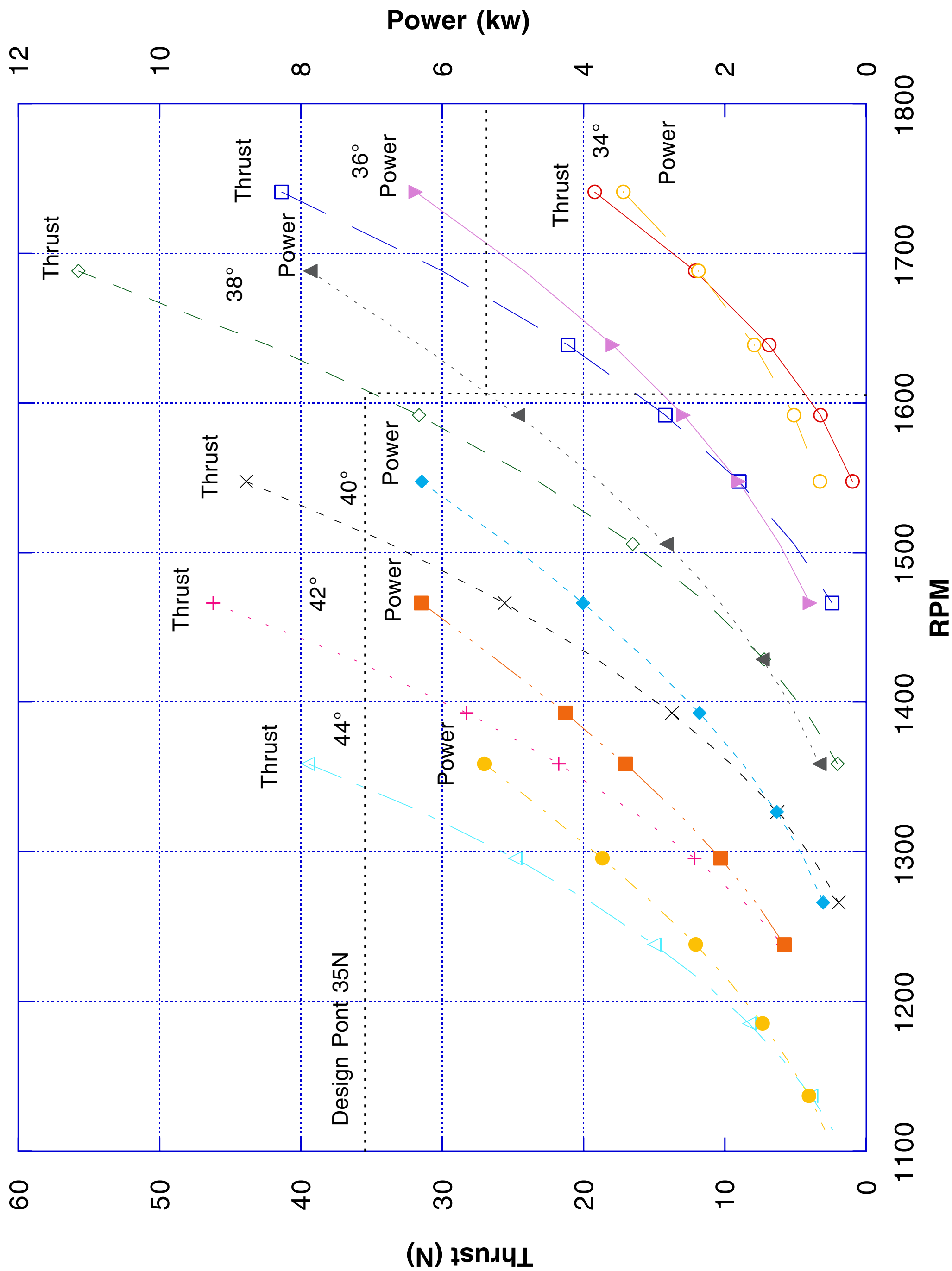

Figure 5 Propeller Performance Curves (2.8 m Diameter, 130 m/s Flight Speed, 6 Blades) [1]

Drivetrain

The drive-train component masses are listed in the following table. These are based on the 35N thrust requirement and a 130 m/s flight velocity.

Component	Mass (kg)
Motor Controller	1.15
Motor	5.52
Gearbox	1.63
Propeller	9.8
Total Drivetrain	18.10

Table 2 Drivetrain Component Masses

Fuel Cell Propulsion System

A simplified fuel cell system can be used to generate electrical power, which in turn can be used to power the electric motor to spin a propeller generating thrust. This system is shown in figure 6. The system consists of a fuel cell stack, pressure regulators, check valves, filters, vent valves, flow and pressure sensors, controller, hydrogen and oxygen pressure tanks and associated piping between the components. The gas flows from the tanks are regulated to the desired pressure and filtered as the gases are passed to the fuel cell. The system is controlled by monitoring the pressure and flow rate in both the hydrogen and oxygen lines and adjusting the flow regulators and exit stream vent control valves to maintain the desired reactant conditions within the fuel cell. The fuel cell will normally operate with the vent flow control valves closed. These valves will need to be opened to purge the fuel cell of water buildup or regulate an imbalance in the flow of one of the reactant gases.

Based on the power required, the fuel cell stack must produce 7142 W of output power during nominal operation. The conversion efficiency of fuel to electricity is approximately 50%. Therefore the flow rate of hydrogen (M) needed can be determined by the following equation, where P_{tot} is the total power output of the fuel cell, η is the fuel cell efficiency estimated to be 50% and F is based on Faradays constant, 96484 (A s/equivalent), which for hydrogen is 192968 (A s/g mole).

$$M = \frac{2P_{tot}}{1.2\eta F1000} \tag{3}$$

This translates into a hydrogen consumption rate of 0.484 kg/hr for the 7.791 kW of power required. Mission durations of 1, 2 and 4 hours were considered. The corresponding hydrogen mass for each mission duration considered is given in table 3. The amount of oxygen needed to react with the hydrogen can be determined from the basic chemical reaction of combining hydrogen and oxygen to from water. This reaction is given below.

$$2H_2(g) + O_2\,(g) \rightarrow 2H_2O \quad (4)$$

Based on this reaction 8 grams of oxygen are consumed for each gram of hydrogen. Therefore the reaction mass flow of oxygen is 8 times that of the hydrogen flow. The oxygen requirement for the 1, 2 and 4 hour missions is given in table 3.

For a storage pressure of 20.7 Mpa (3000 psi) and a factor of safety of 1.8 for the tanks, the specifications on the hydrogen and oxygen system is given in table 3. The gas volumes for both hydrogen (V_h) and oxygen (V_o) were calculated using the equation of state with a compressibility factor for the hydrogen gas. The volumes for both the hydrogen and oxygen are given in table 3. The tank mass is based on a spherical carbon composite tank (3500 Mpa, 507 kpsi yield and 1608 Kg/m^3, 0.058 lb/in^3 density) with a thin metal liner that is resistant to the gas migration out of the tank [3].

$$V_h = 4157.2\ Z\ M_h T_h / P_h \quad (5)$$

$$Z = 0.99704 + 6.4149E\text{-}9\ P_h \quad (6)$$

$$V_o = 259.83\ M_o\ T_o / P_o \quad (7)$$

	1 hour	2 hours	4 hours
Hydrogen Mass	0.484 kg	0.968 kg	1.936 kg
Oxygen Mass	3.872 kg	7.744 kg	15.488 kg
Hydrogen Volume	0.026 m^3	0.053 m^3	0.11 m^3
Oxygen Volume	0.010 m^3	0.021 m^3	0.041 m^3

Table 3 Hydrogen & Oxygen Masses & Volumes for Various Mission Durations

It should be noted that no allowance was made for either the hydrogen or oxygen gas leaking from the tank during the mission. This is particularly a concern for the hydrogen gas due to the high storage pressure and the relatively long transit time to Mars. Cryogenic storage was also considered but the boil off during transit was too great to make this a viable option.

A breakdown of the fuel cell system components and mass estimates for the three mission durations examined are given in table 4. This estimate is based on the 35 N thrust level. The fuel cell stack mass is based on a scaling factor of 1 kg/kW. This is an estimate based on projections of near term fuel cell performance. Using the mass breakdown and power output of the system, a specific energy (W-hr/kg) for each mission duration can be calculated. This specific energy is given in table 4. As the mission duration increases, the specific energy of the fuel cell system will increase. This is because only the tank and reactant masses have to increase duration. As the desired mission duration increases a fuel cell system will become more and more attractive from a mass standpoint.

Component	1 Hour Duration	2 Hours Duration	4 Hours Duration
Fuel Cell	7.8 kg	7.8 kg	7.8 kg
Hydrogen Tank	1.35 kg	2.75 kg	5.71 kg
Oxygen Tank	0.52 kg	1.09 kg	2.13 kg
Check Valves (2)	0.88 kg	0.88 kg	0.88 kg
Pressure Sensors (2)	1.8 kg	1.8 kg	1.8 kg
Flow Sensors (2)	1.8 kg	1.8 kg	1.8 kg
Filters (2)	0.52 kg	0.52 kg	0.52 kg
Flow Regulators (2)	4.8 kg	4.8 kg	4.8 kg
Control Valves (2)	3.0 kg	3.0 kg	3.0 kg
Propellant Lines	1.5 kg	1.5 kg	1.5 kg
Controller Unit	1.5 kg	1.5 kg	1.5 kg
Power System Total Dry Mass	**25.47 kg**	**27.44 kg**	**31.44 kg**
Power System Total Wet Mass	**29.83 kg**	**36.15 kg**	**48.87 kg**
Power System Specific Energy (W-hr / kg)	**239**	**395**	**584**
Drivetrain	18.1 kg	18.1 kg	18.1 kg
Propulsion System Total Mass	**47.93 kg**	**54.25 kg**	**66.97 kg**

Table 4 Fuel Cell Power Generation System Component Masses for Various Flight Durations [4, 5, 6]

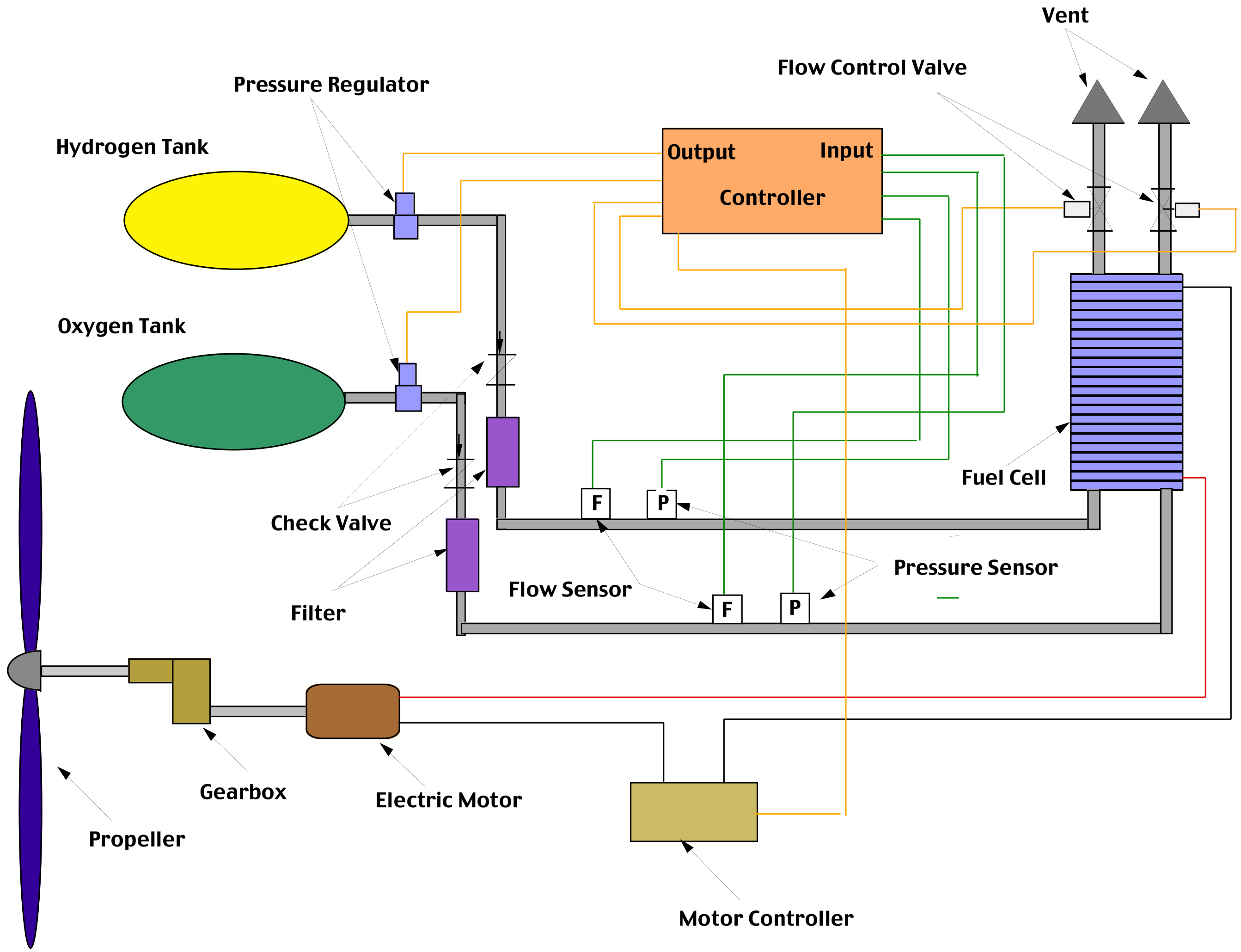

Figure 6 Fuel Cell Propulsion System

Battery Propulsion System

A battery propulsion system, shown in figure 7, is much simpler then the fuel cell system discussed previously. The battery system requires less active controls and no mechanical components in order to operate. This is a benefit for both risk reduction as well as cost and implementation. One of the advantages of the battery system is that it could be easily spread throughout the aircraft without significantly affecting its operation or complexity. Present day battery systems cannot achieve the high energy density (W-Hr / kg) that is achievable with the fuel cell system for the mission durations analyzed. Table 5 lists present and near-term batteries and their operational characteristics. A number of battery types were looked at, and the data presented are from "off the shelf" production batteries. With some development these performance numbers can be expected to improve. Mass, volume and effective specific energy values for each of the batteries considered for the 1, 2 and 4 hour mission durations are given in table 5.

By looking at the data in table 5 it is obvious that some types of batteries will not be applicable to a Mars aircraft mission. This is mainly due to either the battery specific energy being too low or the inability to extract power from the battery at a rate commensurate with the mission duration. Most of the lithium-based batteries, although they have very high specific energies, have very low power discharge rates. Because of this, excess battery capacity must be carried in order to achieve the power level required by the aircraft. This excess capacity effectively reduces the specific energy of the battery for the particular mission duration being considered. Of the batteries listed, the lithium sulfur chloride and lithium manganese dioxide show the most promise. In addition to these lithium batteries, the silver zinc batteries may also be applicable. It should be remembered that this table is based on present day technology and the batteries had not been optimized for aircraft operation. With some development it is reasonable to assume that the mass and volume of the battery can be reduced on the order of 10% to 15% for this application. If a reduction beyond that is necessary for the battery to be applicable, it would not be a good choice for this application.

The masses, volumes and specific power numbers listed in table 5 are based on 7142 W of power required for the mission duration. The mass of the battery will scale with the duration as long as the battery can be discharged within that timeframe. For example the mass of the silver zinc batteries can be reduced in half by reducing the mission duration from 4 hours to 2 hours. This is because this type of battery has a high rate of discharge and will not be effected by the shortened mission duration. On the other hand shortening the duration does not have an effect on the lithium batteries because they cannot provide power at a faster rate. For example, using the lithium sulfur chloride battery to produce 7142 W of power requires a capacity of 8+ hours no matter how long the mission really is.

Another issue that should be mentioned that has not been considered in the preliminary design is the heat removal from the battery system. The batteries will generate considerable heat during operation and most of this heat will need to be removed. The removal of heat from the batteries could be more difficult then the heat removal from the fuel cell system. This is because the heat generated from the battery system will be distributed throughout the complete battery pack. If the batteries are distributed throughout the aircraft a significant amount of heat transfer material will be needed to channel the heat from the batteries to a heat exchanger.

Total propulsion system masses are given in table 7. The battery mass used for each mission duration is the one from table 6 that had the lowest mass for that duration.

Battery Type / Manufacturer	Minimum Operating Duration to 80% DoD	Operating Temperature Range	Energy Density Full Discharge (W Hr / Kg)
Li Mn O_2 / Ultralife	4.8 hr	–30 °C to 55 °C	240
Li Oxyhalide / Electrochem	5 hr	0 °C to 150 °C	137
Li Sulfur Chloride / Electrochem	8.78 hr	–32 °C to 93 °C	444
Li Bromine Complex / Electrochem	8.6 hr	–55 °C to 72 °C	369
Li Thinyl Chloride / Tadiran	20 hr	–20 °C to 150 °C	361
Li Ion / Yardney	4 hr	–20 °C to + 40 °C	145
Nickel Hydrogen / Eagle Pitcher	4 hr or less	–5 °C to + 35 °C	45
Ni-Cad Eagle Pitcher	4 hr or less	–5 °C to + 30 °C	36.5
Silver Zinc / BST Systems	4 hr or less	–20 °C to + 30 °C	150

Table 5 Battery Specifications for Various Battery Types [7,8,9,10,11,12]

	1 Hour Duration			2 Hours Duration			4 Hours Duration		
Battery Type	Mass (kg)	Volume (m^3)	Specific Energy (W hr/kg)	Mass (kg)	Volume (m^3)	Specific Energy (W hr/kg)	Mass (kg)	Volume (m^3)	Specific Energy (W hr/kg)
Li Mn O_2 / Ultralife	143	0.087	49.9	143	0.087	99.9	143	0.087	199.6
Li Oxyhalide / Electrochem	260	NA	27.5	260	NA	54.9	260	NA	110
Li Sulfur Chloride / Electrochem	141	0.067	50.7	141	0.067	101.4	141	0.067	202.8
Li Bromine Complex / Electrochem	166	0.078	43.0	166	0.078	86.0	166	0.078	172
Li Thinyl Chloride / Tadiran	396	0.205	18.0	396	0.205	36.0	396	0.205	72
Li Ion / Yardney	197	0.148	36.3	197	0.148	72.6	197	0.148	145.2
Nickel Hydrogen / Eagle Pitcher	159	0.200	45	317	0.400	45	635	0.800	45
Ni-Cad Eagle Pitcher	196	NA	36.5	391	NA	36.5	783	NA	36.5
Silver Zinc / BST Systems	56	0.003	127	113	0.006	127	225	0.012	127

Table 6 Battery Mass, Volume and Effective Specific Energy for Various Mission Durations [7,8,9,10,11,12]

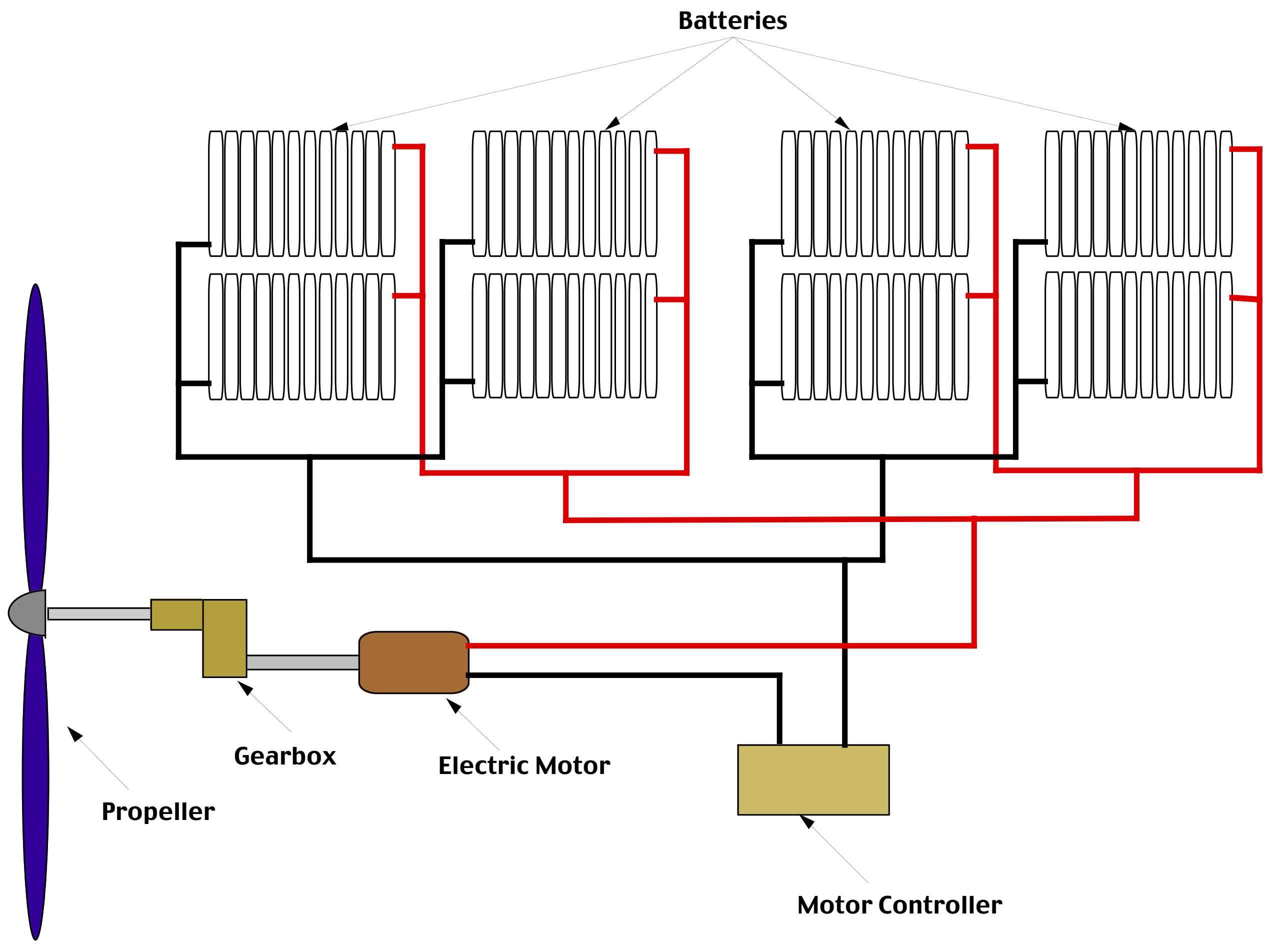

Figure 7 Battery Powered Propulsion System

Component	1 Hour	2 Hours	4 Hours
Battery	56 kg (Silver Zinc)	113 kg (Silver Zinc)	141 kg (Li-Sulfur Chloride)
Wiring	1.5 kg	1.5 kg	1.5 kg
Controller	1.5 kg	1.5 kg	1.5 kg
Drivetrain	18.1 kg	18.1 kg	18.1 kg
Propulsion System Total Mass	**77.1 kg**	**134.1 kg**	**162.1 kg**

Table 7 Battery Propulsion System Mass Breakdown

Combustion Propulsion System

The combustion of fuel to generate thrust dates back to the beginning of aviation. This is the standard method for producing thrust in almost all conventional aircraft. It is accomplished though the use of an internal combustion engine, turbine or some other mechanical device. One advantage almost all of the Earth based aircraft have is the presence of oxygen within the atmosphere. On Mars there is no oxygen available and therefore the combustion engines, which have such a rich aviation heritage here on Earth, cannot be directly applied to Mars flight. Combustion engines, however, can still be considered for Mars flight but the aircraft will need to either carry along an oxidizer or use less energetic fuels that decompose through catalytic reaction.

Standard internal combustion engines can be considered for use with a fuel / oxidizer system. To use a monopropellant through catalytic decomposition, an external combustion engine or piston expander would be required.

The performance of a fuel-driven combustion engine is dependent on the type of fuel and engine (thermodynamic cycle) used as well as the specific engine design. The need to carry along an oxidizer poses a significant weight penalty on the aircraft. Therefore the choice of the fuel / oxidizer combination is critical to optimizing the propulsion system. Factors that need to be considered include: storage volume, environmental constraints (such as freezing temperature and reactivity with other materials) and the amount of energy produced during combustion.

Fuel and Propellant Candidates

The fuel selection will be based on the following criteria:

1. Ability of the fuel to meet the environmental conditions of the mission,
2. Ability of the fuel to be stored in the vehicle
3. Ability of the fuel to meet the performance goals of the aircraft.

The operational constraints on the fuel require it to be capable of being stored for extended periods of time (up to 2 years) with little or no degradation and being capable of withstanding the deep space environment during transit as well as the environment on the surface of Mars. The main environmental issue during transit and on the Mars surface is temperature. Assuming that there is no active thermal control or heating available, the fuel must be capable of withstanding temperatures down to –40 °C for extended periods of time. If the fuel can remain liquid at these temperatures then this greatly simplifies the propellant delivery system as well as eliminates the need for power consuming heaters. Also this reduces the overall risk of the mission by eliminating a failure mechanism which can occur from improperly thawed fuel or a failed heater.

An overall list of potential fuels and oxidizers (and fuel oxidizer combinations) is given in the following sections and included in tables 8, 9 and 10 [1,2].

Fuel/Oxidizer System

Since there is no oxygen available from the environment of Mars the use of a conventional combustion engine requires an oxidizer to be brought along as part of the fuel system. Table 8 and 9 lists fuels and oxidizers that could potentially be used. Various characteristics of the fuels and oxidizers have to be considered when selecting the appropriate combination for used. Ideally they have a low freezing point (below –40°C) to insure that they do not freeze during transit to Mars and would be in the liquid state during storage. Being a liquid is critical due to volume constraints within the aircraft. Other issues such as compatibility and ignition are a significant factor in the selection. Certain fuel/oxidizer combinations will spontaneously react when combined (hypergolic). An example of these combinations are monomethyl hydrazine (MMH) and unsymmetrical dimethyl-hydrazine (UDMH) which are hydrazine derivatives that can operate as the fuel in a bipropellant system. These fuels react instantaneously with certain oxidizers (such as nitrogentetroxide (NTO)) and therefore are hypergolic and do not require an ignition system. This characteristic reduces complexity by eliminating the need for an ignition system and increases system reliability.

Fuel:

Hydrogen

Hydrogen is a stable, non-corrosive, non-toxic material. However in order to be useable for this mission it must be kept in a liquid state. This requires cryogenic storage that would significantly increase the complexity of the mission.

Ammonia

Ammonia is a stable compound that can be stored in Teflon, 18-8 stainless steel, aluminum or polyethylene. It is mildly toxic but can be fatal in concentrated exposure. The main issue with its use for this mission is that it is in the gaseous state under mission conditions.

Hydrazine

Although its most common use is as a monopropellant, hydrazine can also be used as a bipropellant. It has the same general properties as the monopropellant version however its performance is significantly improved when utilized in combination with an oxidizer.

Monomethyl Hydrazine

Monomethyl hydrazine is fairly stable at lower temperatures, however it becomes unstable above 260 °C (500 °F). It can be stored in 18-8 stainless steel, aluminum or Teflon. It is toxic. Its liquid temperature range is well within the requirements for the mission environment.

Unsymmetrical Dimethyl-Hydrazine (UDMH)

Unsymmetrical dimethyl-hydrazine is stable at low temperatures but becomes violently unstable at temperatures above 260 °C (500 °F). It can be stored in most materials including mild steel, 18-8 stainless steel, aluminum, Teflon and polyethylene. It has a lower level of toxicity then hydrazine but more then that of ammonia. Its liquid state temperature range is well within that of the mission requirements.

RP-1

RP-1 is a stable fuel developed for space applications. It is stable up to 370 °C (700 °F) and is compatible with all common metals as well as meoprene, asbestos, fluorocarbons and epoxies. Its toxicity is comparable to that of kerosene. The liquid temperature range for RP-1 is within the operating range for the mission. However, the freezing point is at the estimated low temperature for the mission. To insure that RP-1 doesn't freeze during the mission, some active thermal control would probably be required.

Methane

Methane is stable and compatible with all common metals as well as neoprene, asbestos, fluorocarbons and epoxies. It is essentially non-toxic. The main issue with its use is its low boiling point. This would require it to be used as a gas or stored cryogenically.

Propane

Propane essentially has the same properties as methane. It is stable and compatible with all common metals as well as neoprene, asbestos, fluorocarbons and epoxies. The issues with its use are the same as those of Methane.

Diborane

Diborane is a gas at room temperatures and will slowly decompose. At higher temperatures it decomposes rapidly. It is compatible with most metals and some organic materials. It has moderate toxicity. The issues with using diborane are significant. It would need to be stored as a cryogenic liquid in order to provide for sufficient mission duration as well as minimize the decomposition rate.

Fuel	Boiling Point (°C)	Freezing Point (°C)	Potential Oxidizers	Density (gm/cm^3 at 20 °C)
Hydrogen (H_2)	–253	–259	Oxygen, Fluorine	0.071 (at –253 °C)
Ammonia (NH_3)	–33.4	–77.7	Oxygen, Fluorine, Nitrogen Tetroxide, Chlorine Trifluoride	0.611
Hydrazine (N_2H_4)	113.4	1.5	Oxygen, Fluorine, Nitrogen Tetroxide, Chlorine Trifluoride	1.008
Monomethyl Hydrazine (MMH) (N_2H_6C)	89.2	–52.5	Nitrogen Tetroxide, Chlorine Trifluoride, Inhibited Red Fuming Nitric Acid	0.874
Unsymmetrical Dimethyl Hydrazine (UDMH) ($N_2H_8C_2$)	63.8	–57.2	Oxygen, Fluorine, Nitrogen Tetroxide, Chlorine Trifluoride	0.792
RP-1 ($C_{11.74}H_{21.83}$)	185	–40	Oxygen, Inhibited Red Fuming Nitric Acid	0.801
Methane (CH_4)	–161	–183.9	Oxygen, Fluorine	0.415 (at –164 °C)
Propane (C_3H_8)	–42.2	–187.1	Oxygen, Fluorine	0.585 (at –44 °C)
Diborane (B_2H_6)	–92.6	–164.8	Oxygen Difluoride	0.435 (at –92.6 °C)

Table 8 Fuels and their Phase Change Temperatures [13,14,15]

Oxidizer:

Oxygen

Oxygen is highly reactive and non toxic. It is non-corrosive and is very stable in storage. The main issue with its use is that it would be in the gaseous form under the mission conditions. This will significantly limit the volume of oxygen which can be stored. Liquid oxygen can be used however this brings up significant issues regarding the storage and manufacture of a cryogenic liquid.

Fluorine

Fluorine is highly reactive with almost any material. It can be stored in 18-8 stainless steel or copper but Monel is preferred. It is very important that all materials that come into contact with fluorine are thoroughly cleaned so that there are no contaminating particles for the fluorine to react with. There are no nonmetallic materials that do not react to some degree with fluorine. It is also highly toxic and corrosive to body tissue. Like oxygen it is a gas at mission temperatures. Therefore it would need to be stored cryogenically in order to be used in the mission.

Nitrogen Tetroxide

Nitrogen tetroxide is a stable compound. It is not highly reactive and can be stored in mild steel, stainless steel, aluminum, Teflon and polyethylene. It's toxicity is comparable to that of chlorine. Various formulations of nitrogen tetroxide are available. These formulations vary the percent of nitric oxide in the formulation. This change in the nitric oxide content can effect the freezing point for the propellant. The mixtures of nitrogen tetroxide shown in the table above have varying amounts of nitric oxide. MON 25 (25% mixed NO) has a significantly decreased freezing point over MON 3 (3% mixed NO). This ability to lower the freezing point of nitrogen tetroxide makes it applicable to the mission environment and would eliminate the need for thermal control of the propellant.

Chlorine Trifluoride

Chlorine trifluoride is a stable oxidizer that can be stored in 18-8 stainless steel, nickel and Monel. However most common metals can be used if they are free of contaminants. It is highly toxic with a toxicity comparable to fluorine. Its liquid state temperature range is more then sufficient to meet the mission requirements.

Inhibited Red Fuming Nitric Acid (IRFNA)

IRFNA is subject to decomposition at elevated temperatures and its decomposition rate is directly related to temperature. It can be stored in 18-8 stainless steel, polyethylene and Teflon. It is toxic and corrosive to body tissue. The liquid temperature range of IRFNA is sufficient to keep the oxidizer in a liquid state throughout the proposed mission duration.

Oxygen Difluoride

Oxygen Difluoride is stable at normal room temperature but becomes increasingly unstable at elevated temperatures. It can be stored in 18-8 stainless steel, copper, aluminum, Monel and nickel. Nonmetallic materials are generally not compatible. It is highly toxic and corrosive to body tissue. The main issue is that it is a gas at mission temperatures. In order to be useable it would need to be stored cryogenically in order to be used in the mission. This would add significant risk and complexity to the overall mission.

Oxidizer	Boiling Point (°C)	Freezing Point (°C)	Density (gm/cm^3)
Oxygen (O_2)	–183	–218.8	1.143
Fluorine (F_2)	–188.1	–219.6	1.505
Nitrogen Tetroxide (MON3) (N_2O_4)	21.2	–11.2	1.45
Chlorine Trifluoride (CLF_3)	11.8	–76.6	1.825
Inhibited Red Fuming Nitric Acid (IRFNA) ($0.835HNO_30.140NO_20.020 H_2O0.005HF$)	~60	~ –62.2	1.56
Oxygen Difluoride (OF_2)	–145	–223.9	1.521
Nitrogen Tetroxide (MON25) (N_2O_4)	–9	–54	1.45

Table 9 Oxidizers and their Phase Change Temperatures [13,14,15]

Monopropellant:

To reduce the complexity of the system, some monopropellant fuels can be used. These fuels do not require an oxidizer but react by being passed over a catalyst. A common example is hydrazine that decomposes through the following reaction.

$$3N_2H_4 \rightarrow N_2 + 4NH \tag{8}$$

Although no oxidizer is needed, the heating value of these fuels is about one half that of a conventional hydrocarbon fuel combusted with oxygen. It should be noted that because of the need for a catalyst, hydrazine (as a mono-fuel) is not suitable for internal combustion engines. It can however work in external combustion engines such as a expander engine or Stirling cycle engine.

The catalyst is the main element in the decomposition of hydrazine. The catalyst bed enables the spontaneous and complete decomposition of hydrazine. Increased reliability and simplicity are the main advantages of a single fuel system. This is achieved by not requiring an ignition system and eliminating the oxidizer tank and feed system. The catalyst is not consumed in the reaction and tests have shown that certain catalysts can initiate decomposition of hydrazine at temperatures as low as –54 °C. The catalyst carrier (material that holds the activating metal) has to have as large a surface area as possible. The best and most versatile material for the active metal in the catalyst is iridium. Present day catalysts have on the order of 133 m^2/gm of surface area with 32% of its mass made up of the activation metal (iridium). The iridium catalyst is capable of several hundred spontaneous cold starts and steady-state operation of several hours. This type of catalyst can decompose 40 gm/ cm^2 s of hydrazine [14]. The properties of the various monopropellants are summarized in table 10.

Hydrogen Peroxide

Hydrogen peroxide has a flight heritage dating back to the 1930's, although until recently, the technology has been dormant. It is seeing a revival of sorts, primarily as an oxidizer in a bipropellant combination, but also as a monopropellant. The main advantage of hydrogen peroxide is that it is non-toxic. However, the freezing point is higher then what is necessary to perform the mission without thermal control. There are a few options for dealing with this freezing issue. A passive thermal system may be used to maintain the temperature above freezing. This may be possible since its freezing point is within 30 °C of the expected environmental conditions. Hydrogen peroxide decomposes in the presence of a catalyst such as carbon, steel or copper. For storage it doesn't react with certain materials such as aluminum, tin, glass, polyethylene or Teflon.

Ethylene Oxide

Ethylene oxide remains liquid over a temperature range which is more then adequate to meet the mission requirements. It is generally stable but the polymerization rate is increased in the presence of some materials. Storage materials it is compatible with include 18-8 stainless steel, aluminum, mild steels, copper, nylon and Teflon. This material is relatively toxic and must be handled with caution.

Nitromethane

Nitromethane's temperature range is nearly within the range required by the mission. It would require some insulation or thermal control to assure it would not freeze. It doesn't react with 18-8 stainless steel, aluminum or polyethylene. These materials can be used for storage. The main issue with its use is that it may detonate under conditions of confinement, heating and mechanical impact. Any of which can possibly be experienced during this mission. Also, it is relatively toxic and must be handled with caution.

n-Propyl Nitrate

n-Propyl Nitrate is capable of remaining liquid well within the temperature range of the mission. It is relatively stable and insensitive to mechanical or thermal shock. It can be stored in containers made of either 18-8 stainless steel, aluminum, polyethylene, Teflon, nylon, Orlon, Dacron or Mylar. This material has no serious toxicity problems which allows for easy handling.

Hydrazine

Hydrazine monopropellant has been used in spacecraft for the last 30 years, mainly for low-thrust applications like satellite station-keeping. Decomposition is achieved by a catalyzed reaction with a metal oxide. Materials which are compatible with hydrazine and will not react include Teflon, 18-8 stainless steels, polyethylene and aluminum. The main issues with using hydrazine are that it is highly toxic and has a high freezing point (approximately 1.5°C). Because of this high freezing point, significant heating would be required throughout the mission in order to maintain the propellant in its liquid state.

Hydrazine Propellant Blend (HPB)

HPB represents a family of monopropellant formulations composed of hydrazine, hydrazinium nitrate (HN) and water. The addition of HN and water serve to depress the freezing point and increase the performance of plain hydrazine. Several HPB's were developed and tested in the 1960's and 1970's, primarily for military applications. HPB's are receiving renewed attention as a low freezing point monopropellant. Presently NASA is sponsored HPB development work at Primex Aerospace.

Hydroxylammonium Nitrate (HAN)

HAN is a family of monopropellants composed of an oxidizer rich salt, a fuel component and water. These types of propellants have been under development by NASA over the last decade. HAN based monopropellants offer a high density low freezing point, nontoxic alternative to hydrazine.

Monopropellant	Boiling Point (°C)	Freezing Point (°C)	Density (gm/cm^3)	Combustion Temperature (°C)	Specific Impulse (Isp)
Hydrogen Peroxide($0.9H_2O_2 0.1H_2O$)	141.1	–11.5	1.39	757	148
Ethylene Oxide (C_2H_4O)	10.6	–112.8	0.87	1004	199
Nitromethane (CH_3NO_2)	101.2	–29	1.14	2193	245
n-Propyl Nitrate ($C_3H_7NO_3$)	110.5	–101.1	1.057	1078	210
Hydrazine (N_2H_4)	113.4	1.5	1.008	633	199
Hydrazine Propellant Blend (HPB) HPB-1808 (18% HN, 8% water)	100	–20	na		230
61% Hydroxylammonium Nitrate $(NH_3OH)NO_3$ (HAN) 14% Glycine ($H_2NCH_2\,CO\,OH$)	100	–35	1.41		190

Table 10 Monopropellants and their Phase Change Temperatures [13,14,15]

Of the fuel and oxidizers listed above, the following combinations were determined to be the best candidates for use. The selection was made based on combustion properties, environmental considerations and space flight heritage.

Fuel	Oxidizer	Heat of Combustion (kJ / kg)	Oxidizer / Fuel Ratio (by mass)	Density Fuel / Oxidizer (kg/m3)
Hydrogen (H_2)	Oxygen (O_2)	144,000	8	18.35 / 375.46 (at 3000 psi)
Monomethyl-Hydrazine (MMH) (N_2H_6C)	Nitrogen Tetroxide: 25% mixed Oxides (N_2O_4)	28,347	1.75	874 / 1450
Hydrazine (N_2H_4)	NA	19,429	NA	1008
Unsymmetrical dimethyl-Hydrazine (UDMH) ($N_2H_8C_2$)	Nitrogen Tetroxide 25% mixed Oxides (N_2O_4)	39,960	2.25	792 / 1450

Table 11 Combustion Properties and Characteristics for Candidate Fuels and Oxidizers [13, 16,17]

Piston Expander

The piston expander is an engine that can burn almost any mono or bipropellant. This type of engine is used extensively by the Navy for torpedo propulsion. Due to its light weight and ability to burn a number of different fuel types, this engine may be well suited for a Mars aircraft application. Engines designs that are used by the Navy would not be directly usable in a Mars aircraft. There are a number of development issues which would need to be addressed in order to apply this type of engine to the proposed Mars aircraft.

The two main obstacles to the engine's use are cooling and scaling to the correct power level. The torpedo motor is cooled during operation by circulating sea water through passageways within the engine. This would not be possible for a Mars aircraft and an alternate cooling scheme would need to be devised. The smallest production torpedo motor produces orders of magnitude more power then what is needed by the proposed Mars aircraft. Therefore the engine would need to be significantly scaled down to meet the power and corresponding weight requirements for this type of aircraft and mission. This scaling would require significant development time and cost. Utilizing a version of the torpedo motor in an aircraft was previously examined under the ERAST program in the mid 1990’s. A prototype engine was under development and some preliminary testing was performed. A diagram of this engine is shown in figure 8.

The concept of using a piston expander propulsion engine as the source of power for the propeller dates back to the first attempts at designing a Mars aircraft in the late 1970's. This design was based on the Mini-Sniffer high altitude unmanned vehicle program. Utilizing the Mini-Sniffer development, JPL tried to adapt this aircraft to a Mars atmospheric flight vehicle. The Mini-Sniffer was a small remotely piloted aircraft designed to fly at an altitude of up to 30 km (100,000 ft) on Earth. This was a propeller driven aircraft with a hydrazine burning piston expander engine. The engine was designed by James Akkerman from NASA Johnson Space Center and dubbed the Akkerman engine.

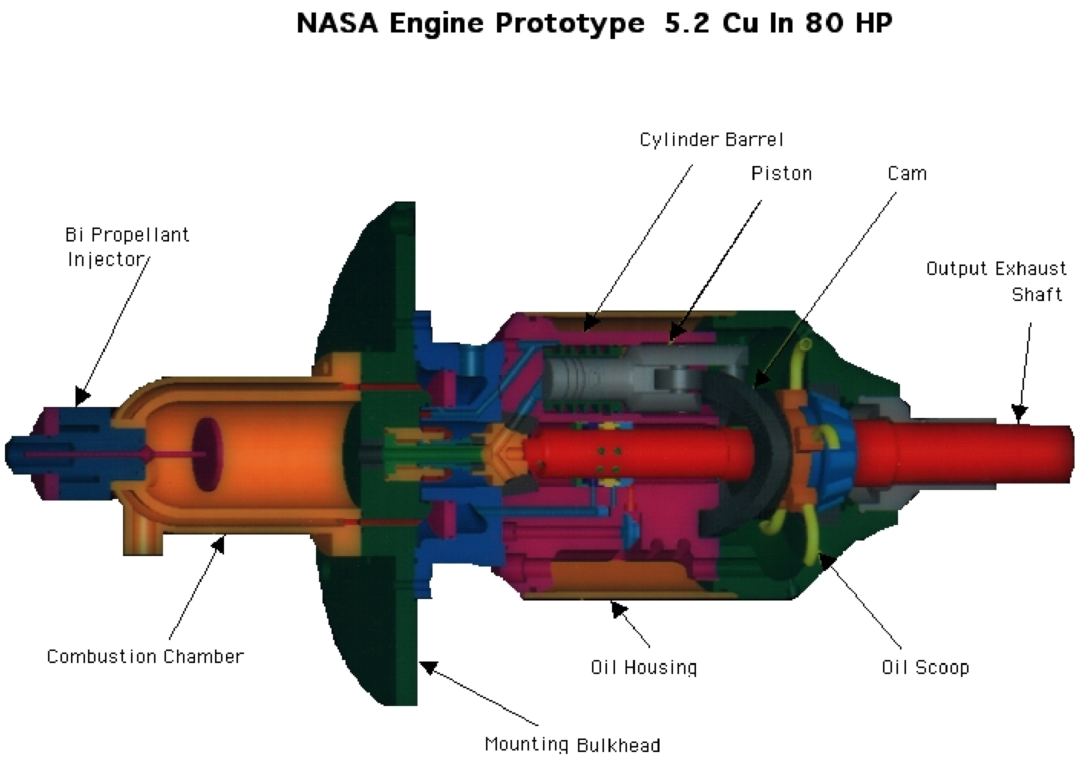

Figure 8 Bipropellant Piston Expander Engine

A prototype engine was constructed and some testing was performed. The Akkerman engine burns hydrazine fuel in a catalyst bed chamber. The prototype was wind tunnel-tested and flight-tested to altitudes of 20 kft. The engine was designed to provide 22kW (30 hp) of power to the propeller. A scaled down version of this engine was proposed for use in a Mars aircraft by Development Sciences Inc. under contract by JPL in the late 1970's. Their estimate for the scaled Akkerman engine was that it would produce 11 kW (15 hp) and weigh 6.8 kg (15 lbs), approximately 1.62 kW/kg for the engine. No prototype of this scaled-down engine was ever produced. The best measured specific fuel consumption for the original prototype engine was 2.7 kg/ kW-hr (4.5 lb/HP hr). However by the end of the 1970's the program was put on hold and all development on the hydrazine engine was stopped. A picture of the Mini-Sniffer aircraft is shown in Figure 9 [18].

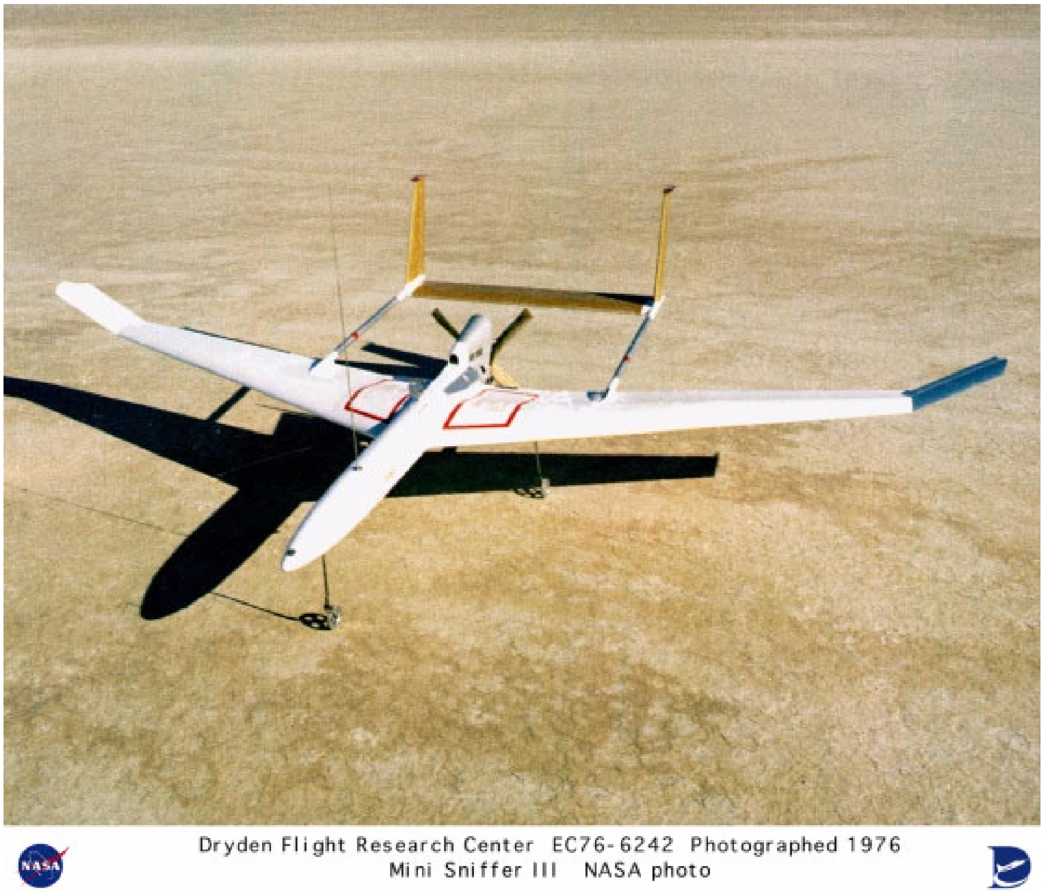

Figure 9 Mini-Sniffer Aircraft

A diagram of a piston expander propulsion system using a monopropellant fuel is shown in figure 10 and using a bipropellant system is shown in figure 11.

Utilizing a heat of combustion for hydrazine (h_c) of –19,429 KJ/Kg and the specific fuel consumption (SPF) of 0.37 kW-hr / kg from the measured fuel flow given above, produces an engine thermal efficiency (η_t) of about 7%. This efficiency is calculated from the following equation

$$\eta_t = SPF\,(3600)\;/\;(-h_c) \tag{9}$$

This measured value is for a prototype engine so it would be reasonable to estimate that after some development, the engine efficiency can reach up to 10%. This low thermal efficiency is characteristic of this type of combustion engine.

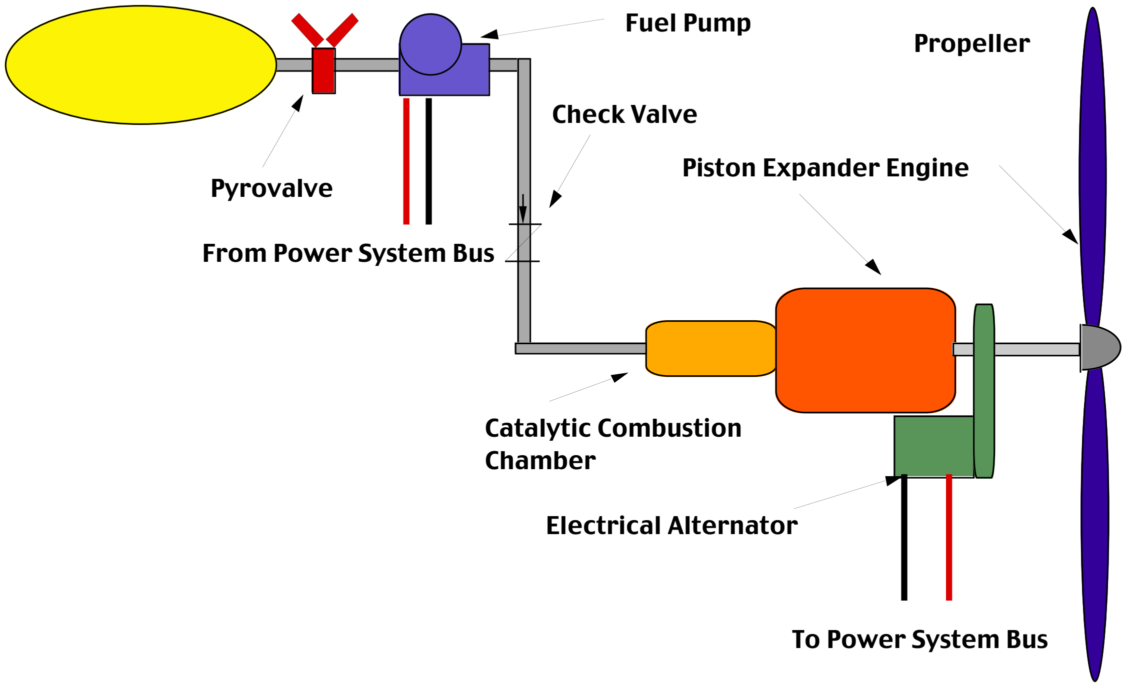

Figure 10 Monopropellant Piston Expander Propulsion System

The fuel requirements for the piston expander engine are given in table 12. These are based on the properties of the fuels listed in table 11 and the flight power requirements for the aircraft (7142 W). It is estimated that the combustion pressure within the combustion chamber will be on the order of 13.8 MPa (2000 psi). Based on this pressure and an assumed 1 second response time, the combustion chamber can be sized (68 cm^3 for hydrazine, 105 cm^3 for MMH + NTO and 83 cm^3 for UDMH + NTO). The liquid fuels and oxidizers are stored in titanium tanks at pressures of 690 kPa (100 psi) with a tank structural factor of safety of 1.8. The gas storage tanks are composed of wound carbon fiber with an inner metal liner with a storage pressure of 20.68 MPa (3000 psi) and a structural factor of safety of 1.8.

A breakdown of the system component masses shown in figures 10 and 11 are listed in table 13 for both the 2 and 4 hour mission duration.

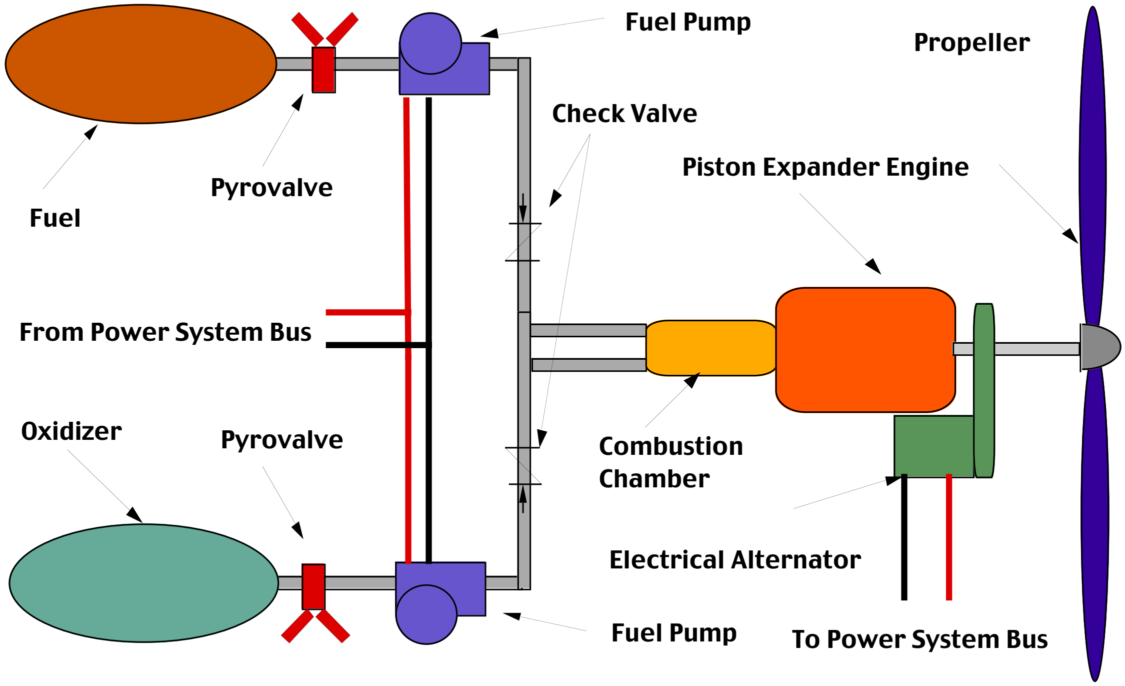

Figure 11 Bipropellant Piston Expander Propulsion System

Duration (hours)		1		2		4	
Fuel / Oxidizer	Flow Rate (kg/hr)	Volume Fuel / Oxidizer (m^3)	Mass Fuel / Oxidizer (kg)	Volume Fuel / Oxidizer (m^3)	Mass Fuel / Oxidizer (kg)	Volume Fuel / Oxidizer (m^3)	Mass Fuel / Oxidizer (kg)
MMH / NTO	9.86 / 17.26	0.0113 / 0.012	9.86 / 17.26	0.023 / 0.0238	19.72 /34.51	0.0451 / 0.0476	39.45 / 69.04
UMDH / NTO	7.02 / 15.79	0.0089 / 0.011	7.02 / 15.79	0.0177 / 0.0218	14.03 / 31.56	0.0354 / 0.0436	28.08 / 63.17
Hydrazine	14.43	0.014	14.43	0.0286	28.86	0.0573	57.71
Hydrogen / Oxygen	1.95 / 15.58	0.106 / 0.0415	1.93 / 15.58	0.2123 / 0.083	3.896 / 31.164	0.4245 / 0.166	7.791 / 62.328

Table 12 Fuel and Oxidizer Specifications for Piston Expander Engine

Fuel / Oxidizer	Hydrazine			MMH and NTO			UMDH and NTO			Hydrogen and Oxygen		
Duration	1 hr	2 hr	4 hr	1 hr	2 hr	4 hr	1 hr	2 hr	4 hr	1 hr	2 hr	4 hr
Fuel Tank	0.29	0.58	1.16	0.23	0.46	0.91	0.18	0.36	0.71	5.50	11.00	22.02
Oxidizer Tank	NA	NA	NA	0.24	0.48	0.96	0.22	0.44	0.88	2.15	4.30	8.62
Pyro Valve	0.3	0.3	0.3	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6
Fuel Pump	0.35	0.35	0.35	0.7	0.7	0.7	0.7	0.7	0.7	4.8*	4.8*	4.8*
Check Valve	0.44	0.44	0.44	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88
Combustion Chamber	0.1	0.1	0.1	0.13	0.13	0.13	0.11	0.11	0.11	0.23	0.23	0.23
Engine	4.8	4.8	4.8	4.8	4.8	4.8	4.8	4.8	4.8	4.8	4.8	4.8
Gearbox	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63
Propeller	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8
Total (dry mass)	**17.71**	**18.00**	**18.94**	**19.01**	**19.48**	**20.41**	**18.92**	**18.92**	**20.11**	**30.39**	**38.04**	**53.38**
Total (wet mass)	**32.14**	**46.86**	**76.65**	**46.13**	**73.71**	**128.90**	**41.73**	**64.91**	**111.36**	**47.90**	**73.10**	**123.50**

Table 13 Piston Expander Engine System Masses (kg) (note * indicates pressure regulator instead of fuel pump)

Conventional Internal Combustion Engine

The internal combustion (IC) engine option is similar to a conventional aircraft IC engine except that it would need to be designed to operate on a bipropellant fuel system and within a low atmospheric density environment. Unlike the piston expander engine described above, a monopropellant would not be usable since the combustion of the fuel needs to take place within the cylinder volume in order for the engine to operate. Piston internal combustion engines are very reliable, well understood pieces of machinery. Conventional aircraft engines are too large for this application. However small IC engines are used regularly for model aircraft. Examples are shown in figures 12 and 13. For this analysis, both 2 and 4 cycle engines were examined. Although they are both classified as IC engines there are significant differences in their operation and performance. In general, 4 cycle engines are more efficient but heavier than 2 cycle engines. The specific power and efficiencies for each are given in table 14.

	4 Cycle Engine	2 Cycle Engine
Specific Power	1.06 kW/kg	1.64 kW/kg
Thermal Efficiency	25%	15%

Table 14 IC Engine Characteristics [19,20]

Figure 12 Fuji BT-86 Two Cycle Engine [19]

Figure 13   O.S. Engines FF320 4 Cycle Engine [20]

Fuel and oxidizer flow rates were calculated using the engine performance and fuel specifications listed in table 11. From these flow rates the required mass and volume of the fuel and oxidizer were calculated. These are shown below in table 15.

The layout of the internal combustion system is the same as that shown in figure 11 for the bipropellant piston expander engine except the engine would be a 2 or 4 cycle internal combustion engine. Based on that diagram, the IC engine performance specifications given in table 14 and the fuel / oxidizer specifications shown in table 15; component mass estimates were determined. These mass estimates are shown in table 16 and 17 for the 2 and 4 cycle engines respectively. As with the piston expander system, the liquid fuels and oxidizers are stored in titanium 690 kPa (100 psi) tanks with a factor of safety of 1.8. The gas storage tanks are composed of wound carbon fiber with an inner metal liner with a storage pressure of 20.68 MPa (3000 psi) and a factor of safety of 1.8.

Duration (hours)		1		2		4	
Fuel / Oxidizer	Flow Rate (kg/hr)	Volume: Fuel / Oxidizer (m^3)	Mass: Fuel / Oxidizer (kg)	Volume: Fuel / Oxidizer (m^3)	Mass: Fuel / Oxidizer (kg)	Volume: Fuel / Oxidizer (m^3)	Mass: Fuel / Oxidizer (kg)
4 Cycle Engine							
MMH / NTO	3.96 / 6.93	0.0046 / 0.0048	3.96 / 6.93	0.0091 / 0.0096	7.92 / 13.86	0.0181 / 0.019	15.84 / 27.72
UMDH / NTO	2.8 / 6.32	0.0036 / 0.0044	2.81 / 6.32	0.0071 / 0.0087	5.615 / 12.633	0.0142 / 0.0174	11.23 / 25.27
Hydrogen / Oxygen	0.78 / 6.24	0.043 / 0.017	0.78 / 6.24	0.085 / 0.033	1.56 / 12.47	0.17 / 0.066	3.12 / 24.93
2 Cycle Engine							
MMH / NTO	6.60 / 11.54	0.0075 / 0.0080	6.60 / 11.51	0.015 / 0.0159	13.19 / 23.01	0.030 / 0.032	26.38 / 46.17
UMDH / NTO	4.68 / 10.53	0.006 / 0.0073	4.68 / 10.53	0.012 / 0.0145	9.36 / 21.06	0.024 / 0.029	18.72 / 42.12
Hydrogen / Oxygen	1.3 / 10.39	0.071 / 0.028	1.30 / 10.39	0.142 / 0.055	2.60 / 20.78	0.284 / 0.111	5.21 / 41.55

Table 15 2 and 4 Cycle Engine Fuel and Oxidizer Specifications

Duration	1 hour			2 hours			4 hours		
Fuel / Oxidizer	MMH & NTO	UMDH & NTO	Hydrogen &Oxygen	MMH &NTO	UMDH & NTO	Hydrogen &Oxygen	MMH & NTO	UMDH & NTO	Hydrogen &Oxygen
Fuel Tank	0.15	0.12	3.69	0.30	0.24	7.37	0.61	0.48	14.73
Oxidizer Tank	0.17	0.15	1.43	0.32	0.29	2.85	0.65	0.58	5.75
Pyro Valve	0.6	0.6	0.6	0.6	.06	.06	0.6	0.6	0.6
Fuel Pump	0.7	0.7	4.8*	0.7	0.7	4.8*	0.7	0.7	4.8*
Check Valve	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88
Engine	4.75	4.75	4.75	4.75	4.75	4.75	4.75	4.75	4.75
Gearbox	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63
Propeller	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8
Total (dry mass)	**18.68**	**18.63**	**27.58**	**18.98**	**18.89**	**36.78**	**19.62**	**19.42**	**42.94**
Total (wet mass)	**36.79**	**33.84**	**39.27**	**55.90**	**49.31**	**56.06**	**92.17**	**80.26**	**89.70**

Table 16 2 Cycle Engine System Masses (kg) (note * indicates pressure regulator instead of fuel pump)

Duration	1 hour			2 hours			4 hours		
Fuel / Oxidizer	MMH & NTO	UMDH & NTO	Hydrogen & Oxygen	MMH & NTO	UMDH & NTO	Hydrogen & Oxygen	MMH & NTO	UMDH & NTO	Hydrogen &Oxygen
Fuel Tank	0.10	0.08	2.21	0.19	0.15	4.41	0.37	0.29	8.82
Oxidizer Tank	0.11	0.09	0.86	0.20	0.18	1.71	0.39	0.35	3.42
Pyro Valve	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6
Fuel Pump	0.7	0.7	4.8*	0.7	0.7	4.8*	0.7	0.7	4.8*
Check Valve	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88
Engine	7.3	7.30	7.30	7.30	7.30	7.30	7.30	7.30	7.30
Gearbox	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63	1.63
Propeller	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8	9.8
Total (dry mass)	**21.12**	**21.08**	**28.08**	**21.30**	**21.24**	**29.81**	**21.67**	**21.55**	**37.25**
Total (wet mass)	**32.01**	**30.21**	**35.10**	**43.08**	**39.49**	**43.84**	**65.23**	**58.05**	**65.30**

Table 17 4 Cycle Engine System Masses (kg) (note * indicates pressure regulator instead of fuel pump)

Combustion Engine Summary

Figures 14 through 16 show the mass comparison of the 3 types of combustion engines that were considered in this analysis for 1, 2 and 4 hour mission durations.

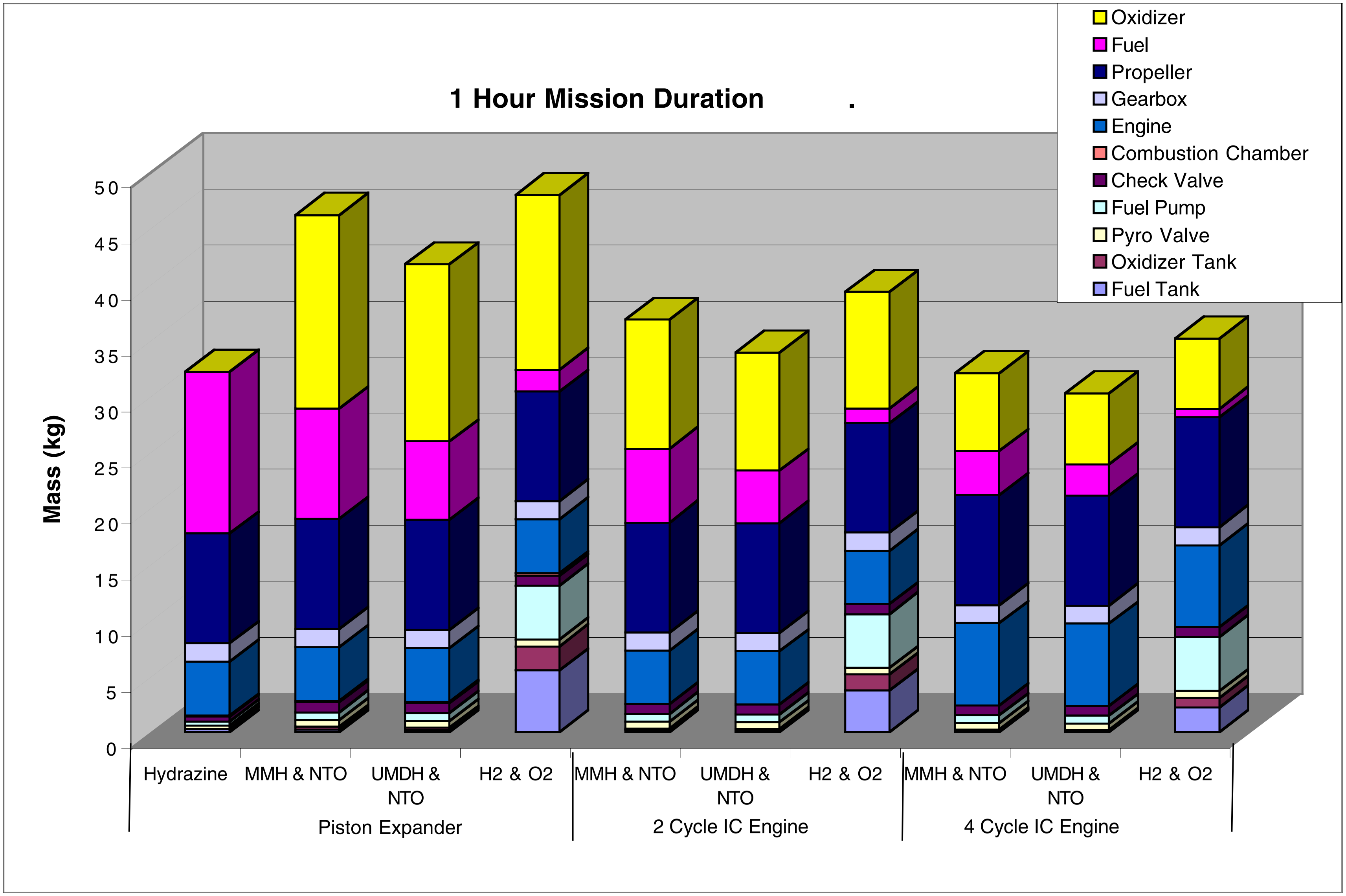

Figure 14 Combustion Engine System Mass Breakdown for 1 Hour Duration Mission

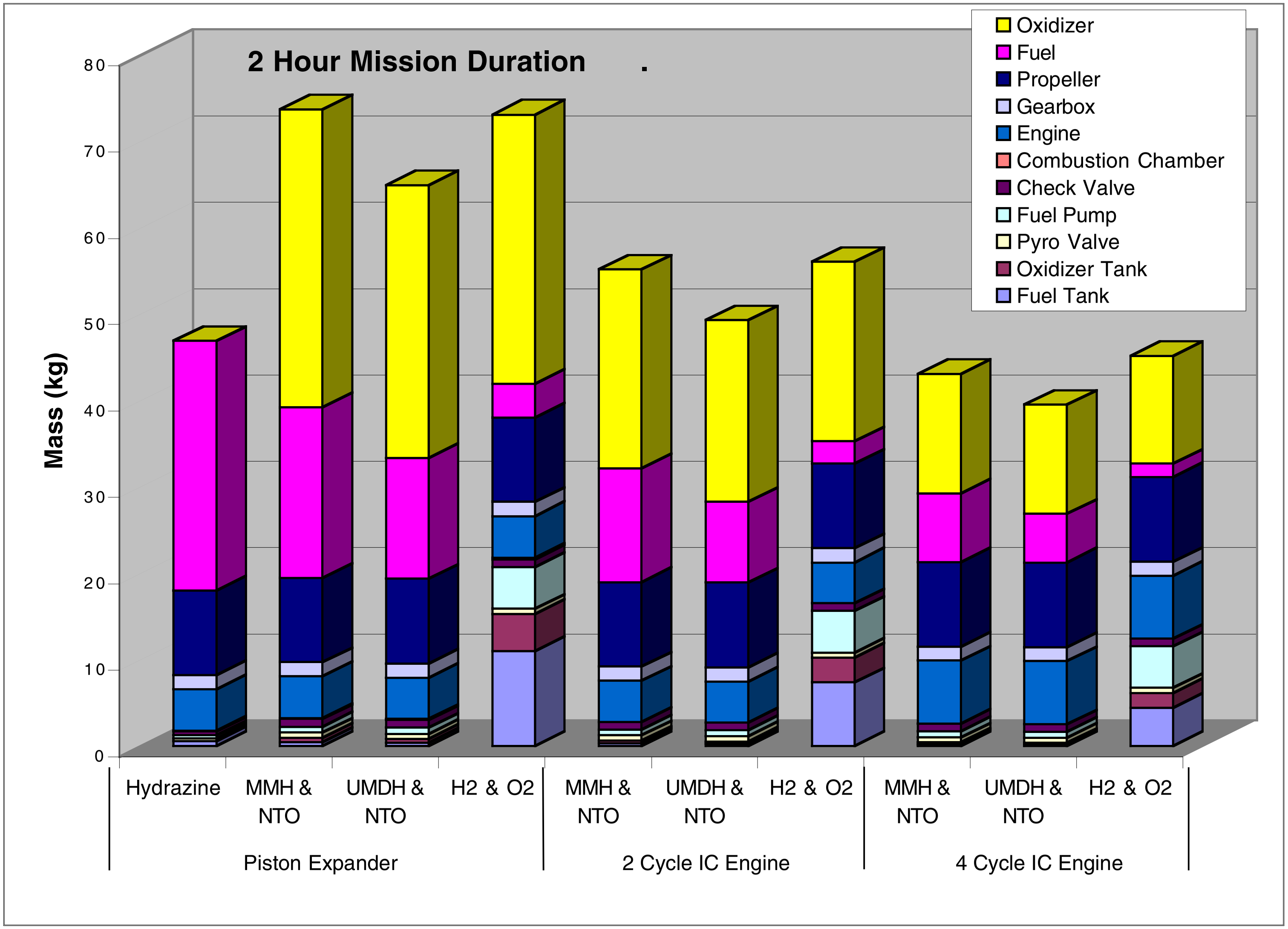

Figure 15 Combustion Engine System Mass Breakdown for 2 Hour Duration Mission

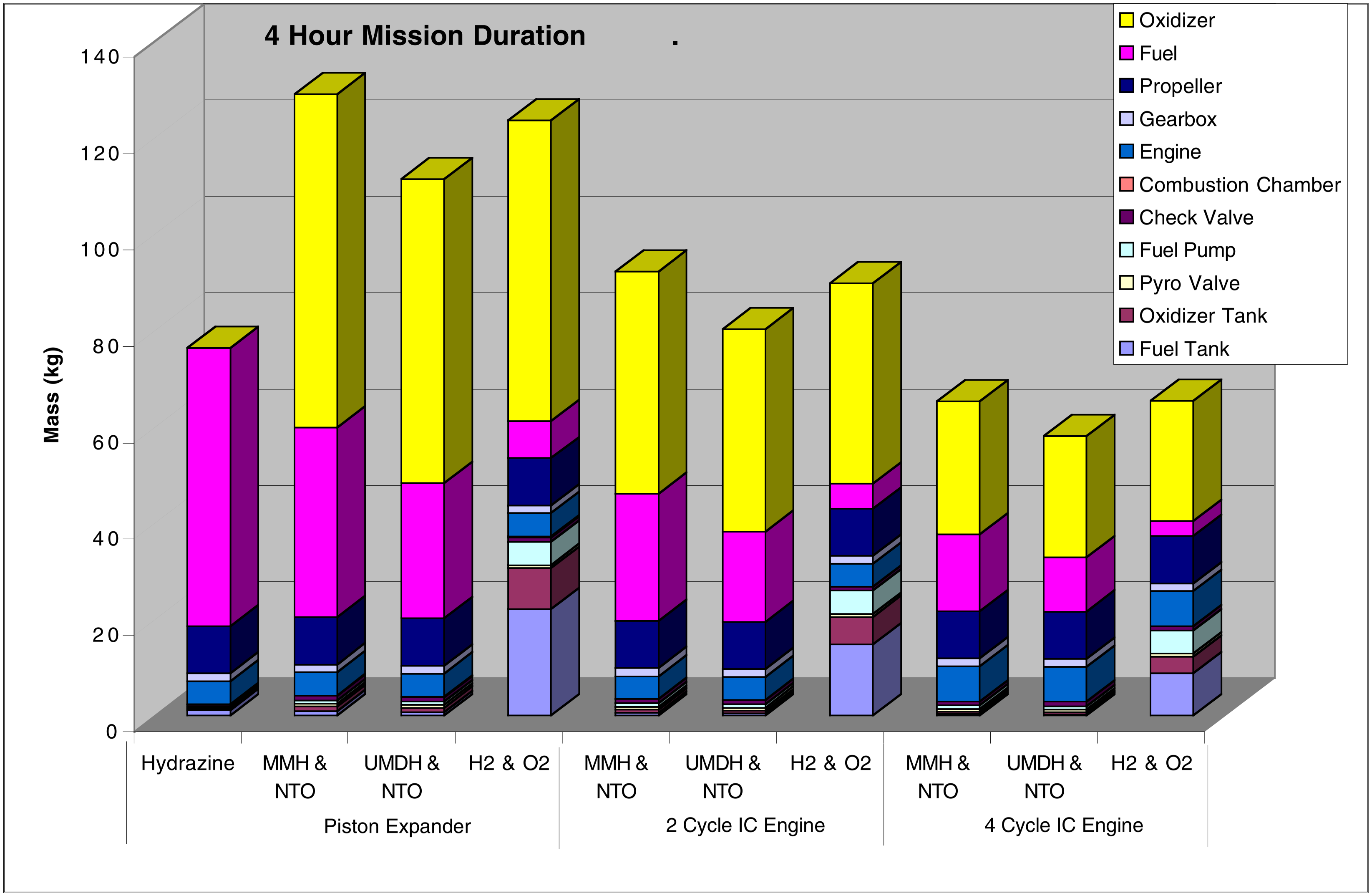

Figure 16 Combustion Engine System Mass Breakdown for 4 Hour Duration Mission

Rocket Propulsion System

Rocket propulsion systems are a very reliable and controllable means of generating thrust. There is a long history of rocket powered aircraft such as the X-1, which was the first aircraft to break the speed of sound, and the X-15, which was the most noted and capable rocket powered aircraft. Due to their high fuel consumption, rocket-powered aircraft have been mostly used as experimental aircraft or to accomplish a specific goal. For Mars flight a rocket-powered aircraft provides some significant advantages over more conventionally propelled aircraft. A rocket system should have no trouble working within the rarefied atmosphere of Mars. The risk associated with a rocket system will be the lowest of all the types of propulsion systems considered.

To evaluate the rocket propulsion system, a number of system designs were considered. These ranged from passive blow-down systems to regulated systems that were controlled either by on / off pulsing or by controlling the propellant flow to the thruster. For each type of control scheme, both monopropellant and bipropellant systems were considered.

Propellant and Performance

There are a number of propellants that can potentially be used to power a rocket-propelled aircraft within the Mars environment. It should be noted that the term propellant means both the fuel and oxidizer combination for a bipropellant system. Ideally a propellant with a low freezing point (below –40 °C) would be used in order to eliminate the need for active heating of the propellant throughout the mission. Table 18 lists a number of bipropellants and monopropellants that may be capable of meeting the temperature requirements as well as being stable throughout the mission. Because of the long transit time to Mars, no cryogenic propellants were considered. Only propellants normally in the liquid state during storage were considered.

To maximize performance, the highest specific impulse propellant should be chosen. Using the specific impulse (Isp) of a given type of propellant, the mass flow required by the rocket to generate a given amount of thrust can be calculated. The thrust generated (T) is given by the mass flow (M) out of the rocket nozzle multiplied by the exhaust velocity (u_e).

$$T = M\, u_e \tag{10}$$

Propellant	Oxidizer / Fuel Ratio	Specific Impulse Isp	Freezing Point (°C)	Combustion Temperature (°C)
Fuel: Monomethyl Hydrazine Oxidizer: Nitrogen Tetroxide (MON25)	1.625	295	–52.5: F –54.0: O	3122
Fuel: Monomethyl Hydrazine Oxidizer: Chlorine Trifluoride	3.00	283	–52.5: F –76.6: O	3318
Fuel: Monomethyl Hydrazine Oxidizer:IRFNA	2.50	274	–52.5: F –62.2: O	2848
Fuel: UDMH Oxidizer: Nitrogen Tetroxide	2.70	286	–57.2: F –54.0: O	3162
Fuel: UDMH Oxidizer: Chlorine Trifluoride	2.85	278	–57.2: F –76.6: O	3306
Fuel: RP-1 Oxidizer: IRFNA	4.90	263	–40.0: F –62.2: O	2881
Monopropellant: Hydrazine	Na	199	1.5	633
Monopropellant: Hydrogen Peroxide	na	148	–11.5	757
Monopropellant: Ethylene Oxide	na	199	–112.8	1004
Monopropellant: Nitromethane	na	245	–29.0	2192
Monopropellant: n-Propyl Nitrate	na	209	–101.1	1077
Monopropellant: HPB-2517	na	220	–54.0	----
Monopropellant: hydroxylammonium nitrate (HAN), glycine and water	na	190	–35.0	----

Table 18 Low Temperature Propellant Candidates [13,14,15]

The exhaust velocity (u_e) can be calculated with the I_{sp} of the propellant and the gravitational constant (g) of 9.81 m/s. It should be noted that the gravitational constant is not dependent on the location of flight (i.e. Mars) but on the location where the I_{sp} was calculated. Therefore the Earth's gravitational constant is used regardless of where the rocket is actually operated.

$$u_e = I_{sp}\, g \tag{11}$$

The above relationships assume that an ideal nozzle is used and that the flow is fully expanded to atmospheric conditions upon exiting the rocket nozzle. Based on this brief analysis, the mass flow and total mass of propellant required can be calculated for a thrust of 35N and flight durations of 1, 2 and 4 hours. The results are shown in table 19.

Propellant	Specific Impulse Isp	Exhaust Velocity (m/s)	Mass Flow (kg/s)	Total Propellant Mass (kg) 1 Hour Flight	Total Propellant Mass (kg) 2 Hour Flight	Total Propellant Mass (kg) 4 Hour Flight
Fuel: Monomethyl Hydrazine Oxidizer: Nitrogen Tetroxide (MON25)	295	2893	0.0121	43.5	87.0	174.0
Fuel: Monomethyl Hydrazine Oxidizer: Chlorine Trifluoride	283	2776	0.0126	45.3	90.5	181.0
Fuel: Monomethyl Hydrazine Oxidizer:IRFNA	274	2687	0.0130	46.8	93.5	187.0
Fuel: UDMH Oxidizer: Nitrogen Tetroxide	286	2806	0.0125	44.8	89.5	179.0
Fuel: UDMH Oxidizer: Chlorine Trifluoride	278	2727	0.0128	46.0	92.0	184.0
Fuel: RP-1 Oxidizer: IRFNA	263	2580	0.0135	48.8	97.5	195.0
Monopropellant: Hydrazine	199	1952	0.0179	64.5	129.0	258.0
Monopropellant: Hydrogen Peroxide	148	1451	0.0241	86.8	173.5	347.0
Monopropellant: Ethylene Oxide	199	1952	0.0179	64.5	129.0	258.0
Monopropellant: Nitromethane	245	2403	0.0146	52.3	104.5	209.0
Monopropellant: n-Propyl Nitrate	209	2050	0.0170	61.3	122.5	245.0
Monopropellant: HPB-2517	220	2158	0.0162	58.3	116.5	233.0
Monopropellant: hydroxylammonium nitrate (HAN), glycine and water	190	1864	0.0188	67.7	135.4	270.7

Table 19 Propellant Mass for 35N Thrust

MMH/MON25

Of the propellants listed in table 16, MON25 seems to be the best choice. Although not a standard rocket fuel propellant, this propellant provides a high ISP (up to 297s in testing), good thermal characteristics for the Mars environment and is usable in present day thrusters [2].

MON25 is the oxidizer which is made up of nitrogen tetroxide with a 25% NO content. The highest specific impulse values (in combination with MMH) occur with mixture ratios of approximately 1.55. The density (ρ) as a function of temperature (T) of MON25 and MMH is given by the following equations and shown in figure 17 [16,17,21].

MON25:

$$\rho = 1.4308 - 0.0021402T - 4.7391E\text{-}6\ T^2 \qquad (12)$$

MMH:

$$\rho = 0.895 - 0.001T + 1.2358E\text{-}11\ T^2 \qquad (13)$$

The specific heat (Cp) of hydrazine (MMH) can be represented by the following relation, where T is the temperature in degrees Kelvin.

$$Cp = 2.7313\text{-}7.2316E\text{-}5\ T + 1.6377E\text{-}6\ T^2 \qquad [J / (g\ K)] \qquad (14)$$

The products of the reaction between MMH and NTO can be of interest due to potential interaction with science instrumentation or other equipment on board the aircraft. Generally if the aircraft is following a straight path there will be little or no contamination by the exhaust gasses. However, if the aircraft circles back it could potentially fly through its exhaust stream (albeit dispersed). This may be of concern for the science instrumentation and data gathering.

The reaction of MMH and NTO yields methylamine, dimethylamine, N-nitrosomethylamine, N-methylformamide, methanol, methylhydrazinium nitrate, water, nitric oxide, nitrous oxide, nitrogen and carbon dioxide [14].

These combustion products are for MON3 not MON25. Because MON25 is an experimental oxidizer and not presently used in production spacecraft, its combustion products are not well documented. However, since the differences between MON25 and MON3 is the concentration of NO there may be differences in the concentration of nitrogen containing compounds but the products should be similar.

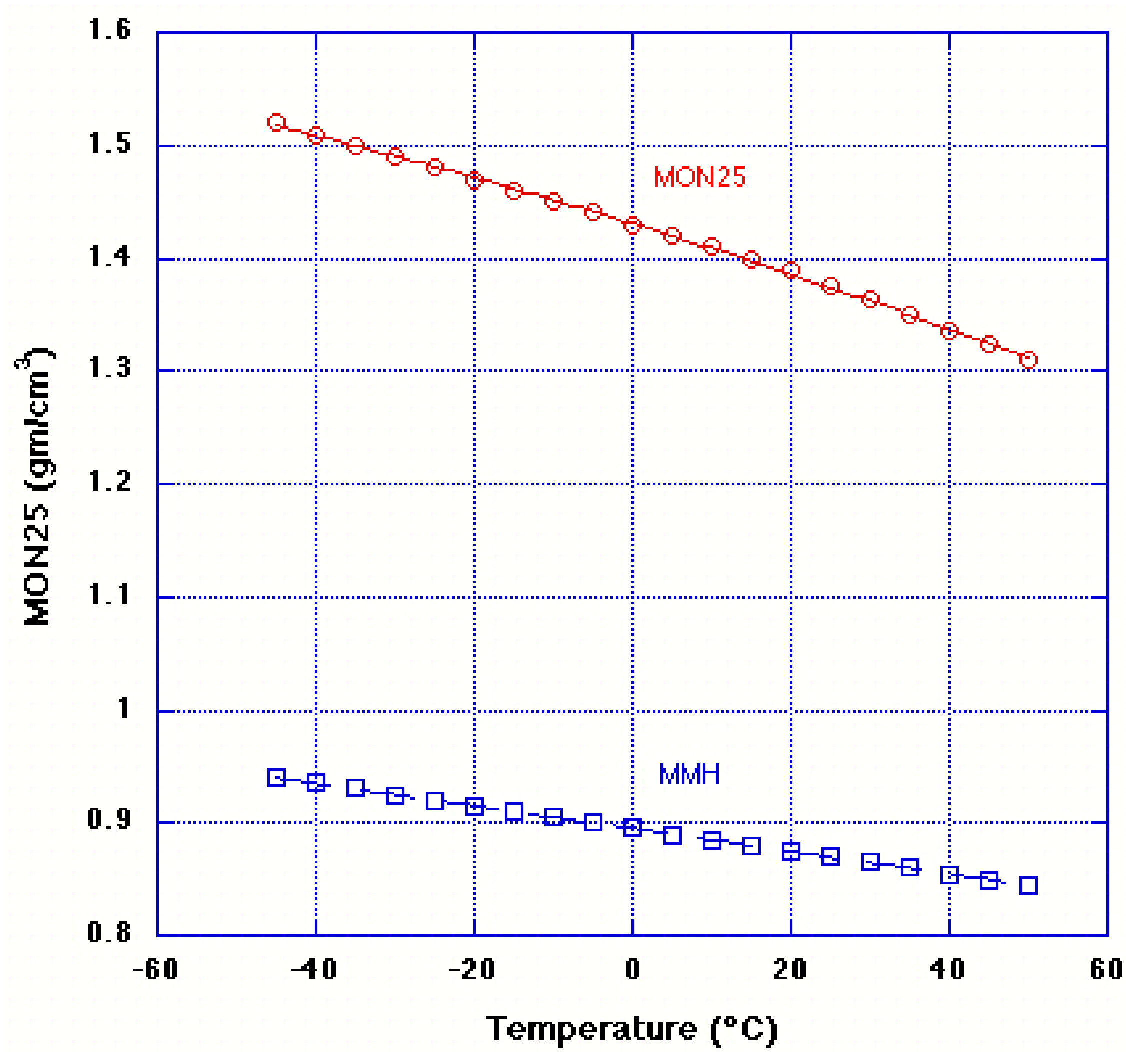

Figure 17 Density of MON25 and MMH as a Function of Temperature [3]

Rocket System Layout: Simple Blowdown System

The simple blowdown system is the most basic and simple type of rocket propulsion system. It consists of propellant tanks (one or two depending on whether it is a monopropellant or bipropellant system) pressurized with helium gas, pyrovalves, and the thrusters. A layout of this system is shown in figure 18 for a bipropellant fueled system. The system is launched with propellant up to the pyrovalves. Pyrovalves are used to prevent vapor migration through the thruster and into the aeroshell (potentially a catastrophic situation). When the system is activated, the normally-closed pyrovalves are fired, bringing propellant to the thruster valves. The system would be controlled by cycling the thruster valves on and off. Because the system is fully pressurized at the start of the mission, the thrust will steadily decrease throughout the aircraft flight. The propellant tanks would need to be large enough to accommodate the fuel as well as the helium pressurant. They could be made smaller if higher pressures were used, but that means a higher initial thrust. To further reduce the system mass, the line filters, thermocouples and pressure transducers can be eliminated. This, however, will increase operational risk.

Because of the non-uniform thrust profile, which will cause the aircraft to be either over or under powered throughout most of the flight, and increased risk, this type of system was deemed not viable for the Mars aircraft application.

Rocket System: Regulated

The regulated system is more complicated and has greater mass then the blowdown system but enables an even thrust level throughout the flight. This type of system, with no active control, is shown in figures 19 and 22 for a bipropellant and monopropellant system respectively. This system has pressurized helium that is used to pressurize the fuel and oxidizer tanks. The helium flow into these tanks is controlled by a pressure regulator which is located on the lines to the fuel and oxidizer tanks. A normally closed pyrovalve is used to seal the helium pressurant from the propellant tanks and to seal the propellant tanks from the thruster during transit to Mars. Check valves and filters are placed in the fuel and oxidizer lines. The system has a transducer for the chamber pressure (thruster performance) and thermocouples for the propellant tanks. The fill lines for the helium and propellant tanks are sealed with normally open pyrovalves which are activated once the system is serviced prior to launch, thereby sealing the access lines. There are checkout lines that are located after each of the pyrovalves. These lines are used to pressure check the system and are sealed manually after the testing is finished.

With the addition of on / off control valves on the thruster, the thruster can be pulsed. This provides some control over the integrated thrust over a period of time. By cycling the thruster on and off the aircraft would glide for a period of time and then resume powered flight. This system is shown in figure 20 for a bipropellant system and figure 23 for a monopropellant system.

For greater control, a variable control valve can be used to vary the thrust of the rocket by regulating the helium flow to the fuel / oxidizer tanks. This enables a system that can be throttled providing variable thrust. This effectively controls the pressure within these tanks and therefore the flow to the engine. However, the amount of thrust regulation achievable will depend on the capabilities of the thruster. To achieve complete thrust throttling capability a variable throat nozzle is required. This adds significant complexity and cost to the design. The variable thrust system is shown in figures 21 and 25 for bipropellant and monopropellant systems respectively

The propellant candidates chosen for analysis with the bipropellant and monopropellant systems is MON-25 / MMH and Hydrazine respectively. The MON-25 / MMH fuel oxidizer combination was chosen because it has a high ISP value and is capable of remaining liquid and operating at temperatures down of less the –40 °C. The low temperature operation eliminates the need for an extensive thermal control system to maintain propellant temperature throughout the mission. The monopropellant hydrazine was chosen because it is the standard monopropellant used for space applications. The other monopropellants, although they have better performance, would require an engine development effort that would add significant cost and time to any aircraft program. The main advantage of hydrazine is that it has good restart characteristics. However, due to its high freezing point, a thermal control system will need to be utilized throughout the mission to insure it does not freeze. A layout of the regulated system components is shown in figures 19 through 23 for both bipropellant (MON-25 / MMH) and monopropellant (hydrazine) systems with and without no / off control capability [22,23].

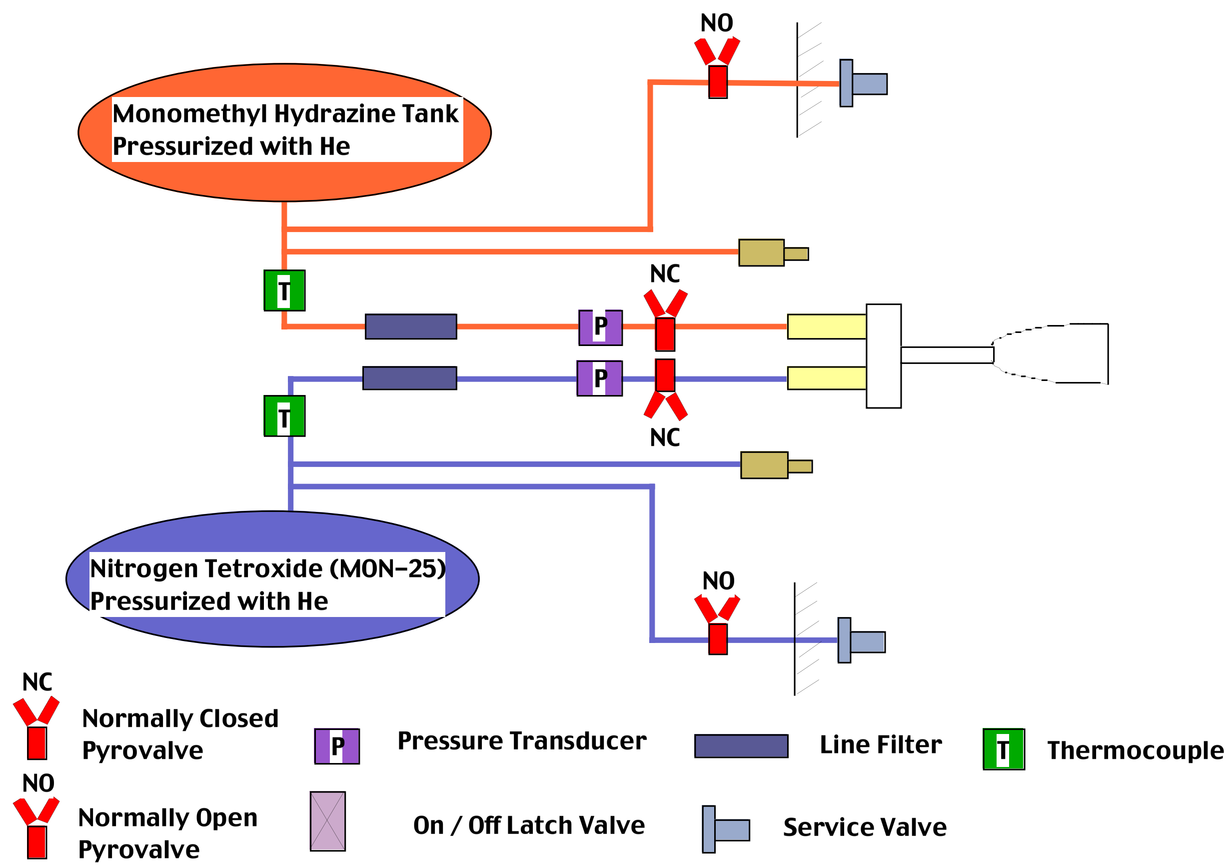

Figure 18 Blowdown Bipropellant Rocket System

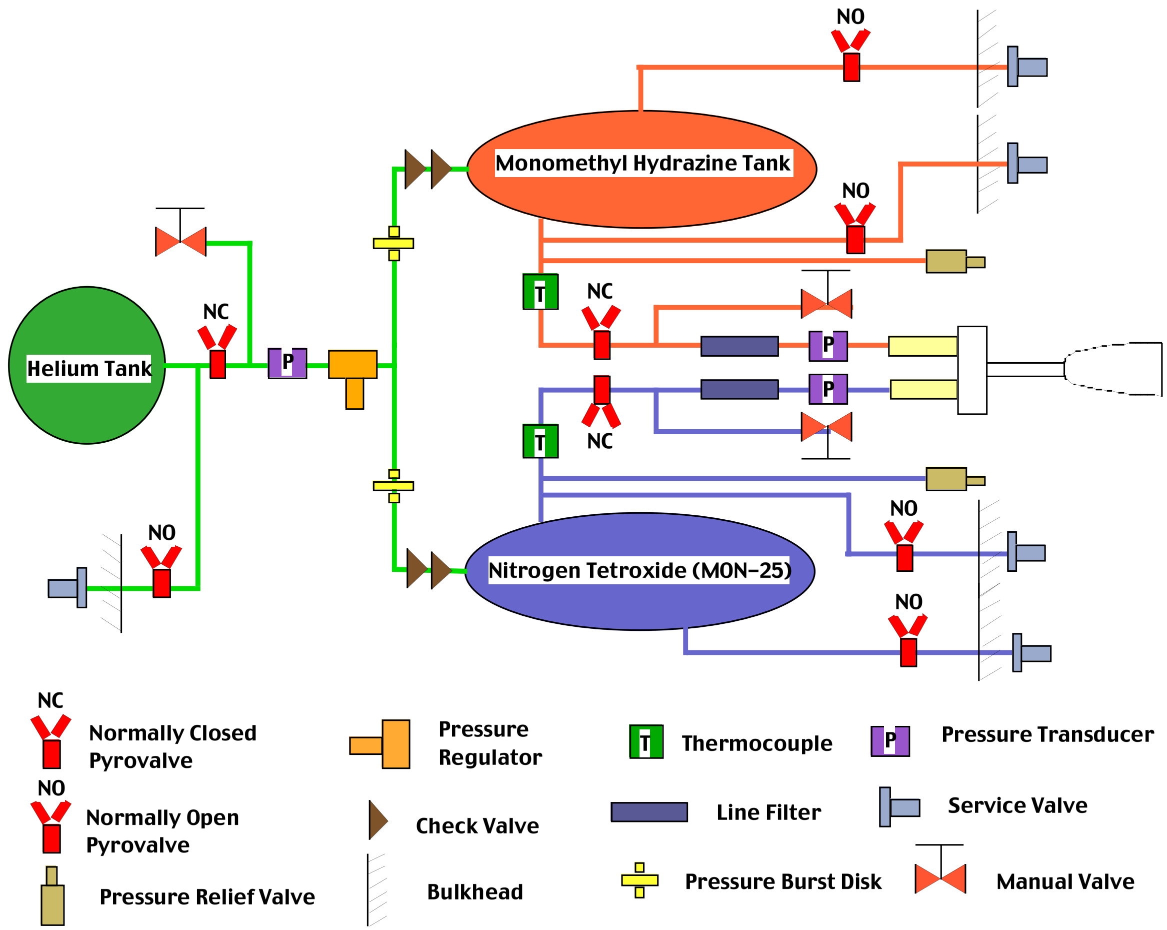

Figure 19 Regulated Bipropellant System with No Active Control

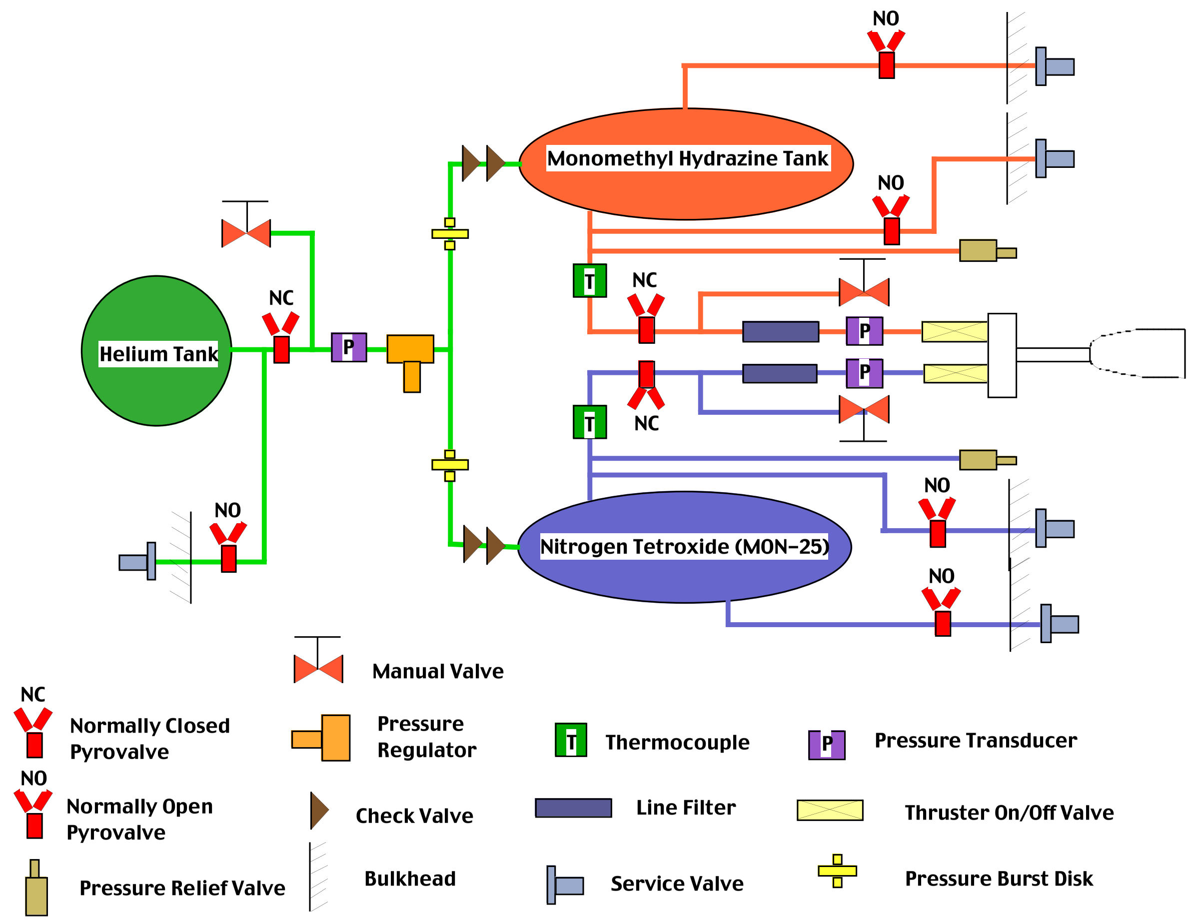

Figure 20 Regulated Bipropellant System with On / Off Control Rocket

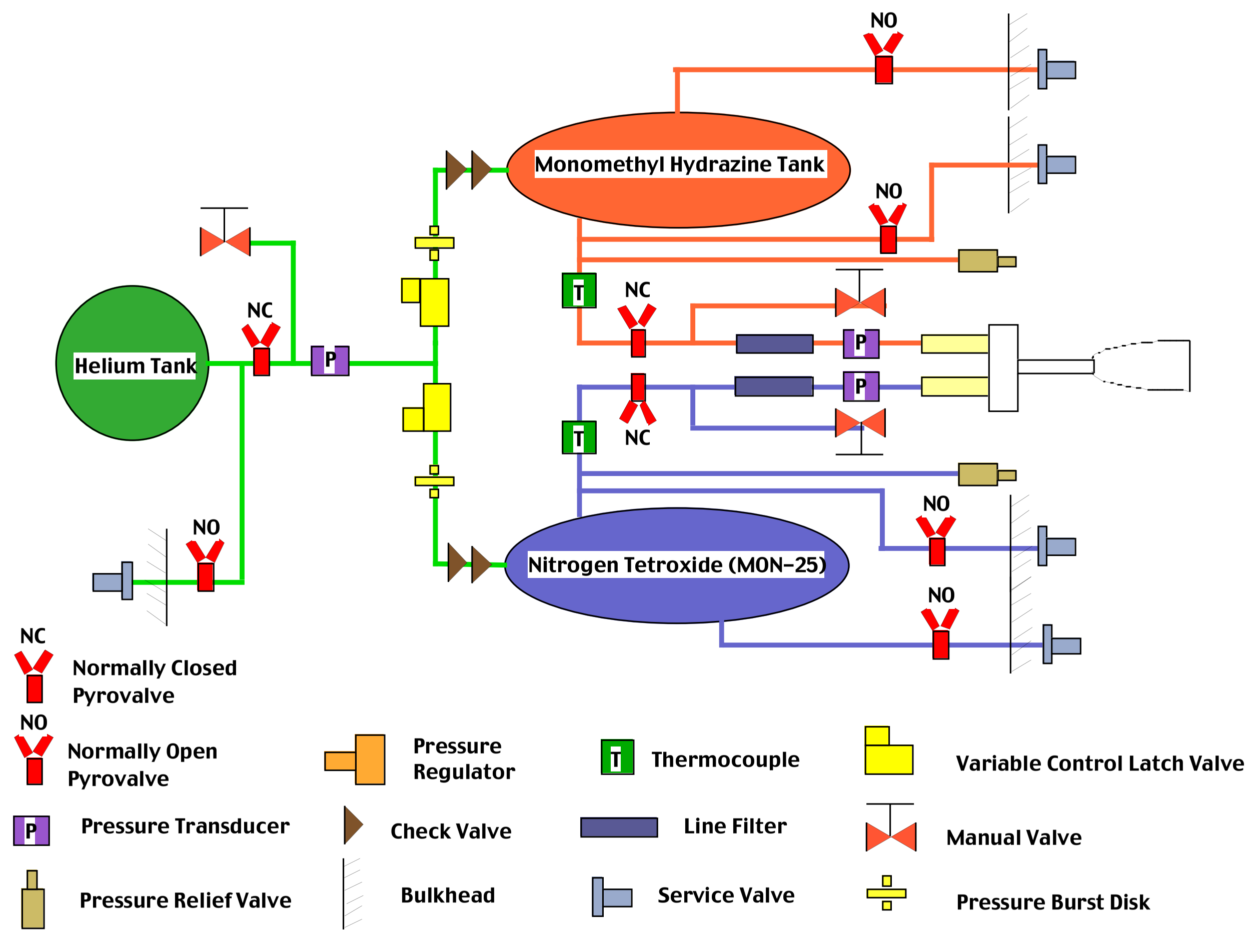

Figure 21 Regulated Bipropellant System with Throttle Control

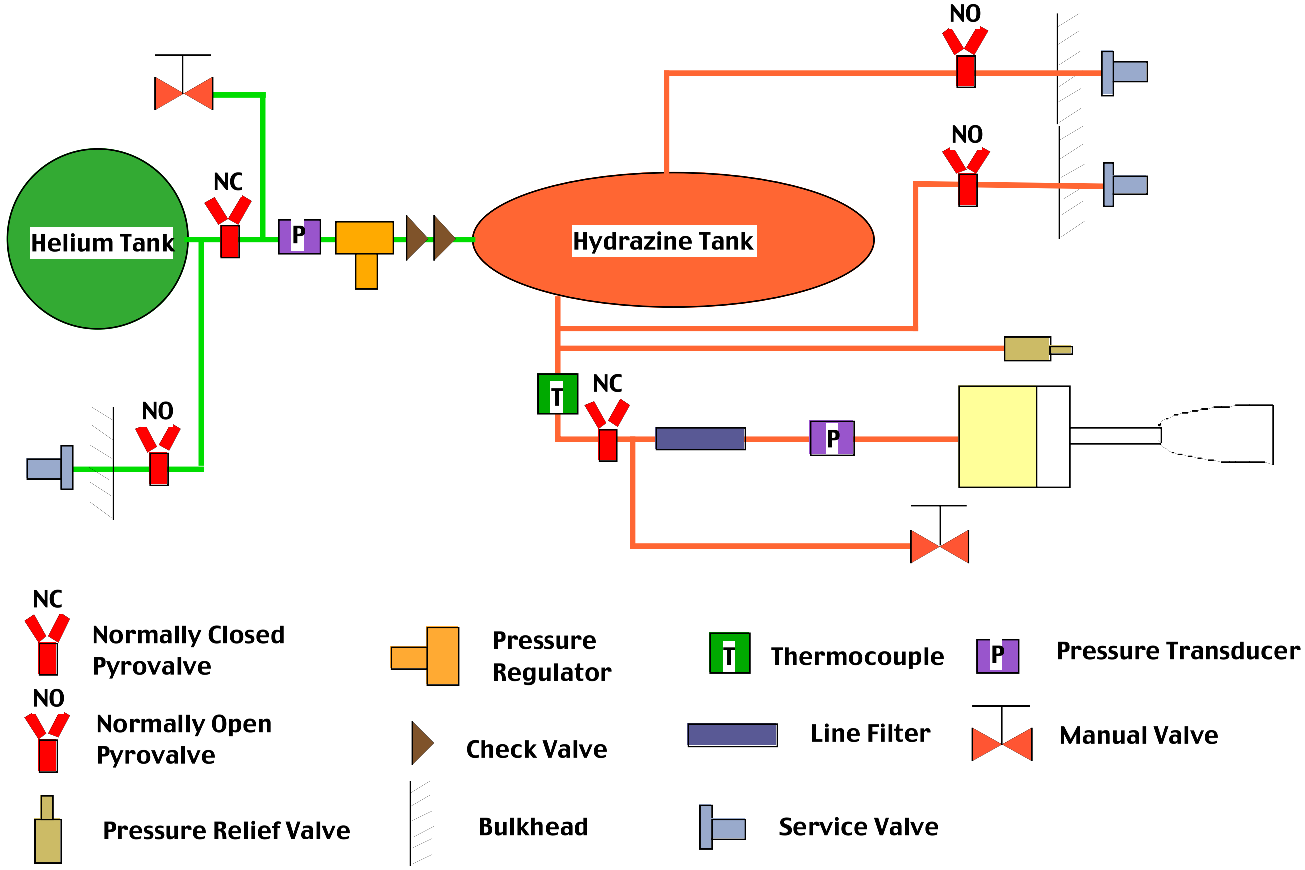

Figure 22 Regulated Monopropellant System with No Active Control

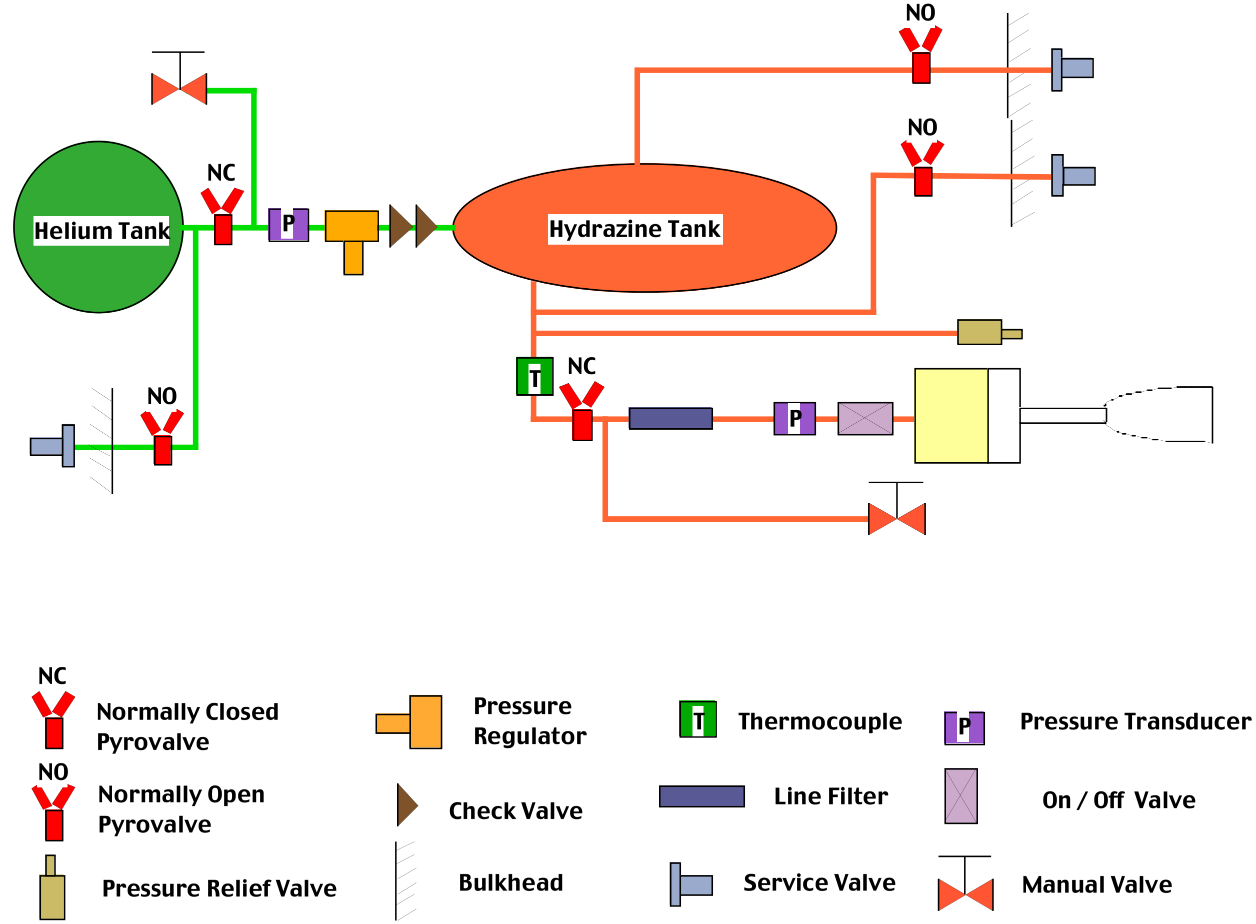

Figure 23 Regulated Monopropellant System with On / Off Control

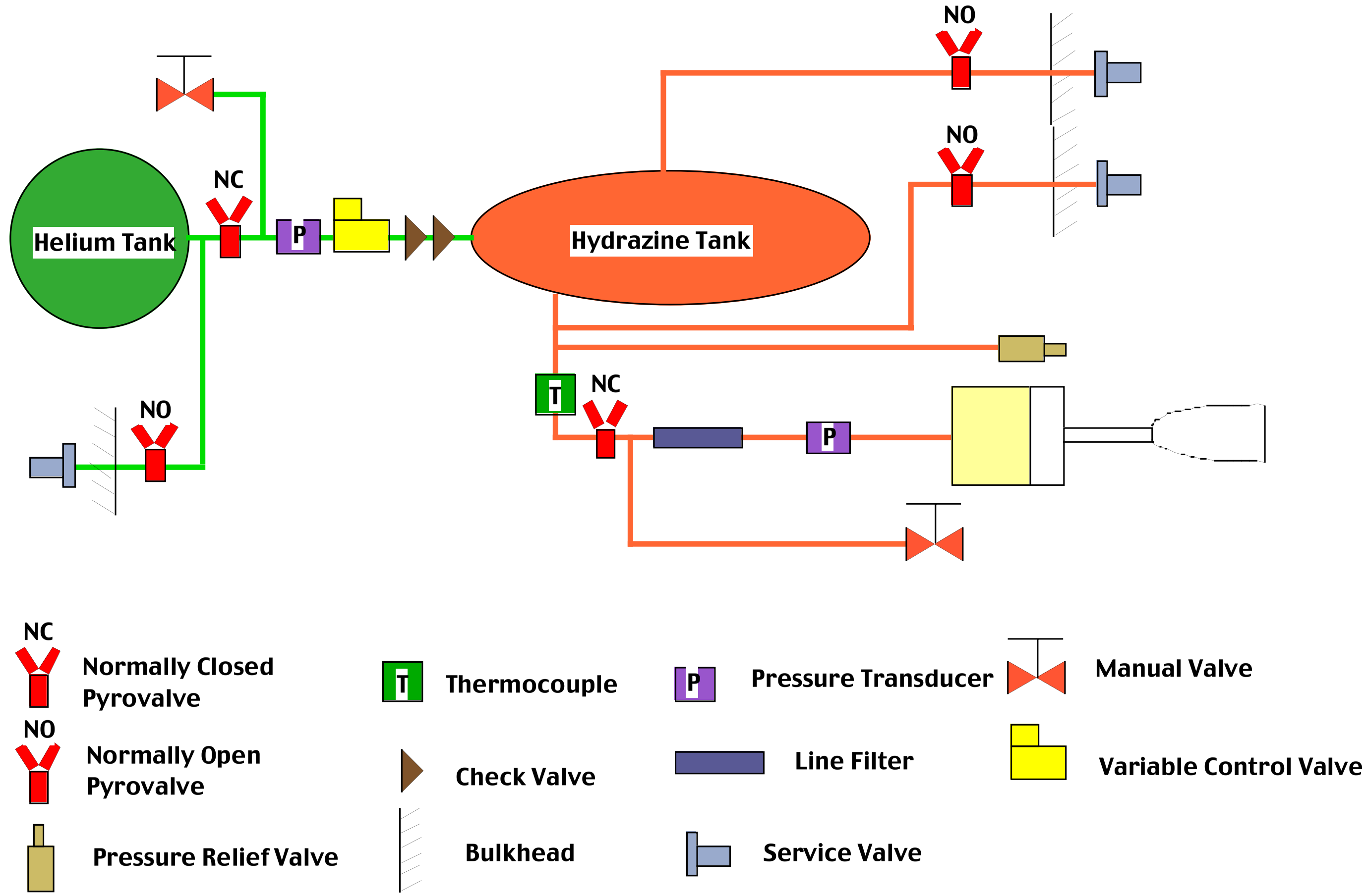

Figure 24 Regulated Monopropellant System with Throttle Control

A breakdown of the masses and volumes for the regulated bi-propellant and monopropellant systems is given in tables 20 through 22 for flight durations of 1, 2 and 4 hours with a continuous thrust level of 35 N. The pressure of the helium tank was assumed to be 20.684 MPa (3000 psi) and the propellant tank pressure was 2.068 MPa (300 psi). The helium tank was assumed to be spherical in shape and made of Arimid fiber composite with a thin metal liner. An additional 10% volume of helium was carried to account for the line volume within the system. The propellant tanks were assumed to be made of titanium and also spherical in shape. Carbon tanks can be used, however a development and qualification program would need to be performed since carbon is not the standard material used for space-qualified propellant tanks. The spherical shape was chosen for simplicity in the analysis. The actual tanks may need to conform to a specific shape within the aircraft fuselage. A factor of safety of 1.8 was used for all tank sizing.

System volume as well as mass is a concern for the feasibility of the propulsion systems. This is especially true for the rocket system since the propellant volume can be considerable. Tables 23 through 25 list the gross volume of the major components for the rocket system for 1, 2 and 4 hour durations. This volume is based on the maximum outside envelope of the component. The line volume is for 3 m of connection lines for the bipropellant system and 1.5 m for the monopropellant system.

Component	Bipropellant No Control	Bipropellant On / Off Control	Bipropellant Throttle Control	Monopropellant No Control	Monopropellant On / Off Control	Monopropellant Throttle Control
Helium Tank	0.528 kg	0.528 kg	0.528 kg	0.889 kg	0.889 kg	0.889 kg
MMH or Hydrazine Tank	1.06 kg	1.06 kg	1.06 kg	3.56 kg	3.56 kg	3.56 kg
NTO Tank	1.06 kg	1.06 kg	1.06 kg	NA	NA	NA
Pressure Transducer	(3) 0.9 kg	(3) 0.9 kg	(3) 0.9 kg	(2) 0.60 kg	(2) 0.60 kg	(2) 0.60 kg
Pyrovalve normally closed	(4) 1.2 kg	(4) 1.2 kg	(4) 1.2 kg	(2) 0.60 kg	(2) 0.60 kg	(2) 0.60 kg
Pyro Valve normally open	(5) 2.15 kg	(5) 2.15 kg	(5) 2.15 kg	(3) 1.29 kg	(3) 1.29 kg	(3) 1.29 kg
Pressure Regulator (fixed or variable)	(1) 0.34 kg	(1) 0.34 kg	(2) 1.5 kg	(1) 0.34 kg	(1) 0.34 kg	(1) 0.75 kg
Check Valve	(4) 0.16 kg	(4) 0.16 kg	(4) 0.16 kg	(2) 0.08 kg	(2) 0.08 kg	(2) 0.08 kg
Thermocouple	(2) 0.04 kg	(2) 0.04 kg	(2) 0.04 kg	(1) 0.02 kg	(1) 0.02 kg	(1) 0.02 kg
Line Filter	(2) 0.16 kg	(2) 0.16 kg	(2) 0.16 kg	(1) 0.08 kg	(1) 0.08 kg	(1) 0.08 kg
Thruster	(1) 1.50 kg	(1) 2.50 kg	(1) 2.50 kg	(1) 1.50 kg	(1) 2.50 kg	(1) 2.50 kg
On / Off Valve	(3) 1.50 kg	(3) 1.50 kg	(3) 1.50 kg	(2) 1.00 kg	(2) 1.00 kg	(1) 1.00 kg
Lines and Fittings	~1.0 kg	~1.0 kg	~1.0 kg	~0.50 kg	~0.50 kg	~0.50 kg
Pressure Relief Valve	(2) 1.05 kg	(2) 1.05 kg	(2) 1.05 kg	(1) 0.53 kg	(1) 0.53 kg	(1) 0.53 kg
Burst Disc	(2) 0.1 kg	(2) 0.1 kg	(2) 0.1 kg	NA	NA	NA
Total Dry Mass	**12.75 kg**	**13.75 kg**	**14.91 kg**	**10.99 kg**	**11.99 kg**	**12.40 kg**
MMH or Hydrazine	16.57 kg	16.57 kg	16.57 kg	64.50 kg	64.50 kg	64.50 kg
Nitrogen Tetroxide	26.93 kg	26.93 kg	26.93 kg	NA	NA	NA
Total Wet Mass	**56.25 kg**	**57.25 kg**	**58.41 kg**	**75.49 kg**	**76.49 kg**	**76.90 kg**

Table 20 Rocket System Mass Breakdown for 1 Hour Flight Duration [3,4,24,25,26,27,28,29,30,31]

Component	Bipropellant No Control	Bipropellant On / Off Control	Bipropellant Throttle Control	Monopropellant No Control	Monopropellant On / Off Control	Monopropellant Throttle Control
Helium Tank	1.03 kg	1.03 kg	1.03 kg	1.78 kg	1.78 kg	1.78 kg
MMH or Hydrazine Tank	2.06 kg	2.06 kg	2.06 kg	7.11 kg	7.11 kg	7.11 kg
NTO Tank	2.06 kg	2.06 kg	2.06 kg	NA	NA	NA
Pressure Transducer	(3) 0.9 kg	(3) 0.9 kg	(3) 0.9 kg	(2) 0.60 kg	(2) 0.60 kg	(2) 0.60 kg
Pyrovalve normally closed	(4) 1.2 kg	(4) 1.2 kg	(4) 1.2 kg	(2) 0.60 kg	(2) 0.60 kg	(2) 0.60 kg
Pyro Valve normally open	(5) 2.15 kg	(5) 2.15 kg	(5) 2.15 kg	(3) 1.29 kg	(3) 1.29 kg	(3) 1.29 kg
Pressure Regulator (fixed or variable)	(1) 0.34 kg	(1) 0.34 kg	(2) 1.5 kg	(1) 0.34 kg	(1) 0.34 kg	(1) 0.75 kg
Check Valve	(4) 0.16 kg	(4) 0.16 kg	(4) 0.16 kg	(2) 0.08 kg	(2) 0.08 kg	(2) 0.08 kg
Thermocouple	(2) 0.04 kg	(2) 0.04 kg	(2) 0.04 kg	(1) 0.02 kg	(1) 0.02 kg	(1) 0.02 kg
Line Filter	(2) 0.16 kg	(2) 0.16 kg	(2) 0.16 kg	(1) 0.08 kg	(1) 0.08 kg	(1) 0.08 kg
Thruster	(1) 1.50 kg	(1) 2.50 kg	(1) 2.50 kg	(1) 1.50 kg	(1) 2.50 kg	(1) 2.50 kg
On / Off Valve	(3) 1.50 kg	(3) 1.50 kg	(3) 1.50 kg	(2) 1.00 kg	(2) 1.00 kg	(1) 1.00 kg
Lines and Fittings	~1.0 kg	~1.0 kg	~1.0 kg	~0.50 kg	~0.50 kg	~0.50 kg
Pressure Relief Valve	(2) 1.05 kg	(2) 1.05 kg	(2) 1.05 kg	(1) 0.53 kg	(1) 0.53 kg	(1) 0.53 kg
Burst Disc	(2) 0.1 kg	(2) 0.1 kg	(2) 0.1 kg	NA	NA	NA
Total Dry Mass	**15.25 kg**	**16.25 kg**	**17.41 kg**	**15.43 kg**	**16.43 kg**	**16.84 kg**
MMH or Hydrazine	33.14 kg	33.14 kg	33.14 kg	129.00 kg	129.00 kg	129.00 kg
Nitrogen Tetroxide	53.86 kg	53.86 kg	53.86 kg	NA	NA	NA
Total Wet Mass	**102.25 kg**	**103.25 kg**	**104.41 kg**	**159.86 kg**	**145.43 kg**	**145.84 kg**

Table 21 Rocket System Mass Breakdown for 2 Hour Flight Duration [3,4,24,25,26,27,28,29,30,31]

Component	Bipropellant No Control	Bipropellant On / Off Control	Bipropellant Throttle Control	Monopropellant No Control	Monopropellant On / Off Control	Monopropellant Throttle Control
Helium Tank	2.06 kg	2.06 kg	2.06 kg	3.55 kg	3.55 kg	3.55 kg
MMH or Hydrazine Tank	4.11 kg	4.11 kg	4.11 kg	14.23 kg	14.23 kg	14.23 kg
NTO Tank	4.11 kg	4.11 kg	4.11 kg	NA	NA	NA
Pressure Transducer	(3) 0.9 kg	(3) 0.9 kg	(3) 0.9 kg	(2) 0.60 kg	(2) 0.60 kg	(2) 0.60 kg
Pyrovalve normally closed	(4) 1.2 kg	(4) 1.2 kg	(4) 1.2 kg	(2) 0.60 kg	(2) 0.60 kg	(2) 0.60 kg
Pyro Valve normally open	(5) 2.15 kg	(5) 2.15 kg	(5) 2.15 kg	(3) 1.29 kg	(3) 1.29 kg	(3) 1.29 kg
Pressure Regulator (fixed or variable)	(1) 0.34 kg	(1) 0.34 kg	(2) 1.5 kg	(1) 0.34 kg	(1) 0.34 kg	(1) 0.75 kg
Check Valve	(4) 0.16 kg	(4) 0.16 kg	(4) 0.16 kg	(2) 0.08 kg	(2) 0.08 kg	(2) 0.08 kg
Thermocouple	(2) 0.04 kg	(2) 0.04 kg	(2) 0.04 kg	(1) 0.02 kg	(1) 0.02 kg	(1) 0.02 kg
Line Filter	(2) 0.16 kg	(2) 0.16 kg	(2) 0.16 kg	(1) 0.08 kg	(1) 0.08 kg	(1) 0.08 kg
Thruster	(1) 1.50 kg	(1) 2.50 kg	(1) 2.50 kg	(1) 1.50 kg	(1) 2.50 kg	(1) 2.50 kg
On / Off Valve	(3) 1.50 kg	(3) 1.50 kg	(3) 1.50 kg	(2) 1.00 kg	(2) 1.00 kg	(1) 1.00 kg
Lines and Fittings	~1.0 kg	~1.0 kg	~1.0 kg	~0.50 kg	~0.50 kg	~0.50 kg
Pressure Relief Valve	(2) 1.05 kg	(2) 1.05 kg	(2) 1.05 kg	(1) 0.53 kg	(1) 0.53 kg	(1) 0.53 kg
Burst Disc	(2) 0.1 kg	(2) 0.1 kg	(2) 0.1 kg	NA	NA	NA
Total Dry Mass	**16.27 kg**	**17.27 kg**	**18.43 kg**	**24.32 kg**	**25.32 kg**	**25.73 kg**
MMH or Hydrazine	66.28 kg	66.28 kg	66.28 kg	258.00 kg	258.00 kg	258.00 kg
Nitrogen Tetroxide	107.72 kg	107.72 kg	107.72 kg	NA	NA	NA
Total Wet Mass	**190.27 kg**	**191.27 kg**	**192.43 kg**	**282.32 kg**	**283.32 kg**	**283.73 kg**

Table 22 Rocket System Mass Breakdown for 4 Hour Flight Duration [3,4,24,25,26,27,28,29,30,31]

Component	Bipropellant No Control	Bipropellant On / Off Control	Bipropellant Throttle Control	Monopropellant No Control	Monopropellant On / Off Control	Monopropellant Throttle Control
Helium Tank	4,180 cm^3	4,180 cm^3	4,180 cm^3	7,040 cm^3	7,040 cm^3	7,040 cm^3
MMH or Hydrazine Tank	19,000 cm^3	19,000 cm^3	19,000 cm^3	64,000 cm^3	64,000 cm^3	64,000 cm^3
NTO Tank	19,000 cm^3	19,000 cm^3	19,000 cm^3	NA	NA	NA
Pressure Transducer	105 cm^3	105 cm^3	105 cm^3	70 cm^3	70 cm^3	70 cm^3
Pyrovalve normally closed	257 cm^3	257 cm^3	257 cm^3	129 cm^3	129 cm^3	129 cm^3
Pyro Valve normally open	2,500 cm^3	2,500 cm^3	2,500 cm^3	1,500 cm^3	1,500 cm^3	1,500 cm^3
Pressure Regulator (fixed or variable)	938 cm^3	938 cm^3	1,876 cm^3	938 cm^3	938 cm^3	938 cm^3
Check Valve	128 cm^3	128 cm^3	128 cm^3	64 cm^3	64 cm^3	64 cm^3
Thermocouple	30 cm^3	30 cm^3	30 cm^3	15 cm^3	15 cm^3	15 cm^3
Line Filter	136 cm^3	136 cm^3	136 cm^3	68 cm^3	68 cm^3	68 cm^3
Thruster	368 cm^3	860 cm^3	860 cm^3	368 cm^3	860 cm^3	860 cm^3
On / Off Valve	369 cm^3	369 cm^3	369 cm^3	246 cm^3	246 cm^3	246 cm^3
Lines and Fittings	216 cm^3	216 cm^3	216 cm^3	108 cm^3	108 cm^3	108 cm^3
Pressure Relief Valve	32 cm^3	32 cm^3	32 cm^3	16 cm^3	16 cm^3	16 cm^3
Burst Disc	64 cm^3	64 cm^3	64 cm^3	NA	NA	NA
Total Volume	**47,323 cm^3**	**47,815 cm^3**	**48,753 cm^3**	**74,562 cm^3**	**75,054 cm^3**	**75,054 cm^3**

Table 23 Rocket System Volume Breakdown for 1 Hour Flight Duration [3,4,24,25,26,27,28,29,30,31]

Component	Bipropellant No Control	Bipropellant On / Off Control	Bipropellant Throttle Control	Monopropellant No Control	Monopropellant On / Off Control	Monopropellant Throttle Control
Helium Tank	8,140 cm^3	8,140 cm^3	8,140 cm^3	14,080 cm^3	14,080 cm^3	14,080 cm^3
MMH or Hydrazine Tank	37,000 cm^3	37,000 cm^3	37,000 cm^3	128,000 cm^3	128,000 cm^3	128,000 cm^3
NTO Tank	37,000 cm^3	37,000 cm^3	37,000 cm^3	NA	NA	NA
Pressure Transducer	105 cm^3	105 cm^3	105 cm^3	70 cm^3	70 cm^3	70 cm^3
Pyrovalve normally closed	257 cm^3	257 cm^3	257 cm^3	129 cm^3	129 cm^3	129 cm^3
Pyro Valve normally open	2,500 cm^3	2,500 cm^3	2,500 cm^3	1,500 cm^3	1,500 cm^3	1,500 cm^3
Pressure Regulator (fixed or variable)	938 cm^3	938 cm^3	1,876 cm^3	938 cm^3	938 cm^3	938 cm^3
Check Valve	128 cm^3	128 cm^3	128 cm^3	64 cm^3	64 cm^3	64 cm^3
Thermocouple	30 cm^3	30 cm^3	30 cm^3	15 cm^3	15 cm^3	15 cm^3
Line Filter	136 cm^3	136 cm^3	136 cm^3	68 cm^3	68 cm^3	68 cm^3
Thruster	368 cm^3	860 cm^3	860 cm^3	368 cm^3	860 cm^3	860 cm^3
On / Off Valve	369 cm^3	369 cm^3	369 cm^3	246 cm^3	246 cm^3	246 cm^3
Lines and Fittings	216 cm^3	216 cm^3	216 cm^3	108 cm^3	108 cm^3	108 cm^3
Pressure Relief Valve	32 cm^3	32 cm^3	32 cm^3	16 cm^3	16 cm^3	16 cm^3
Burst Disc	64 cm^3	64 cm^3	64 cm^3	NA	NA	NA
Total Volume	**87,283 cm^3**	**87,775 cm^3**	**88,713 cm^3**	**145,602 cm^3**	**146,094 cm^3**	**146,094 cm^3**

Table 24 Rocket System Volume Breakdown for 2 Hour Flight Duration [3,4,24,25,26,27,28,29,30,31]

Component	Bipropellant No Control	Bipropellant On / Off Control	Bipropellant Throttle Control	Monopropellant No Control	Monopropellant On / Off Control	Monopropellant Throttle Control
Helium Tank	16,280 cm^3	16,280 cm^3	16,280 cm^3	28,160 cm^3	28,160 cm^3	28,160 cm^3
MMH or Hydrazine Tank	74,000 cm^3	74,000 cm^3	74,000 cm^3	256,000 cm^3	256,000 cm^3	256,000 cm^3
NTO Tank	74,000 cm^3	74,000 cm^3	74,000 cm^3	NA	NA	NA
Pressure Transducer	105 cm^3	105 cm^3	105 cm^3	70 cm^3	70 cm^3	70 cm^3
Pyrovalve normally closed	257 cm^3	257 cm^3	257 cm^3	129 cm^3	129 cm^3	129 cm^3
Pyro Valve normally open	2,500 cm^3	2,500 cm^3	2,500 cm^3	1,500 cm^3	1,500 cm^3	1,500 cm^3
Pressure Regulator (fixed or variable)	938 cm^3	938 cm^3	1,876 cm^3	938 cm^3	938 cm^3	938 cm^3
Check Valve	128 cm^3	128 cm^3	128 cm^3	64 cm^3	64 cm^3	64 cm^3
Thermocouple	30 cm^3	30 cm^3	30 cm^3	15 cm^3	15 cm^3	15 cm^3
Line Filter	136 cm^3	136 cm^3	136 cm^3	68 cm^3	68 cm^3	68 cm^3
Thruster	368 cm^3	860 cm^3	860 cm^3	368 cm^3	860 cm^3	860 cm^3
On / Off Valve	369 cm^3	369 cm^3	369 cm^3	246 cm^3	246 cm^3	246 cm^3
Lines and Fittings	216 cm^3	216 cm^3	216 cm^3	108 cm^3	108 cm^3	108 cm^3
Pressure Relief Valve	32 cm^3	32 cm^3	32 cm^3	16 cm^3	16 cm^3	16 cm^3
Burst Disc	64 cm^3	64 cm^3	64 cm^3	NA	NA	NA
Total Volume	169,423 cm^3	169,915 cm^3	170,853 cm^3	287,682 cm^3	288,174 cm^3	288,174 cm^3

Table 25 Rocket System Volume Breakdown for 4 Hour Flight Duration [3,4,24,25,26,27,28,29,30,31]

Nozzle Sizing

In order to maximize the rocket engine performance as well as minimize the impact of the exhaust plume on the science experiments, the nozzle should be designed to fully expand the flow. This will minimize the movement of the exhaust gases from the exhaust stream once they leave the nozzle. To fully expand the exhaust gases the exit pressure (p_e) of the exhaust stream would need to equal the ambient atmospheric pressure (p_a). The size of the nozzle necessary to fully expand the flow within the Mars environment can be estimated by the following basic analysis. This analysis assumes a constant specific heat, ideal gas and isentropic flow through the nozzle. The desired exit Mach number (M_e) can be found from equation 15, where p_o is the stagnation pressure in the combustion chamber, p_a is the atmospheric pressure and γ is the ratio of specific heats for the exhaust gas. Since it is assumed that the exit pressure (P_e) will equal the ambient pressure (P_a) by fully expanding the exhaust flow through the nozzle then the Mach number of the exhaust gases at atmospheric conditions (M_a) is equal to the exit Mach number (M_e) [32]. Therefore, M_a in equation 15 is equivalent to M_e.

$$\frac{P_o}{P_a} = (1 + \frac{\gamma - 1}{2} M_a^2)^{\frac{\gamma}{\gamma - 1}} \tag{15}$$

Once the exit Mach number is determined from equation 15, then the area ratio for the nozzle can be calculated. This area ratio is the ratio between the nozzle throat (A*, point where the flow within the nozzle reaches Mach 1) and the exit (A_e) is given by equation 16.

$$\frac{A_e}{A^*} = \frac{1}{M_a}(\frac{2}{\gamma + 1}(1 + \frac{\gamma - 1}{2} M_e^2))^{\frac{\gamma + 1}{2(\gamma - 1)}} \tag{16}$$

The stagnation pressure just before the nozzle converging section will be about 95% of that at the inlet to the combustion chamber. This is for a combustion chamber that is uniform in area along its length and about 2 times the diameter of the nozzle throat. Common inlet pressure into the combustion chamber is usually on the order of 2.07 MPa (300 psi), however the exact pressure will depend on the rocket system design. Therefore the stagnation pressure within the chamber (p_o) for use in determining the exit mach number is 1.97 MPa (286 psi). For this initial sizing it will be assumed that the ratio of the specific heats of the exhaust gases (γ) is 1.23. This is representative of the specific heat for hydrazine combustion with nitrogen tetroxide.

Once the area ratio is calculated, the actual throat area will depend on the mass flow required to produce the necessary thrust (T). For a fully expanded nozzle this throat area is given by equation 17 [32].

$$A^* = \frac{T}{p_o \sqrt{\frac{2\gamma^2}{\gamma - 1}\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left(1 - \left(\frac{p_e}{p_o}\right)^{\frac{\gamma-1}{\gamma}}\right)}} \tag{17}$$

The required length of the nozzle can be determined by assuming a standard 15° cone angle for the nozzle in conjunction with the throat and exit radius values. Utilizing the values and equations given above, the resultant geometry for the nozzle with a thrust level of 35N is summarized in table 26.

Parameter	Value
Exit Mach Number	5.49
Nozzle Area Ratio (A_e/A*)	153.67
Throat Area (A*) and Diameter	0.0942 cm^2 / 0.346 cm
Exit Area (A_e) and Diameter	14.5 cm^2 / 4.292 cm
Nozzle Length	7.37 cm

Table 26 Ideal Nozzle Geometry for Mars Atmosphere and 35 N Thrust

Off Design Rocket Engine Operation

As propellant is consumed during the flight, less thrust is needed. Therefore it would be advantageous to reduce the thrust level throughout the flight. The thrust level of the engine can be reduced by lowering the propellant tank pressure. The rocket systems shown in figures 21 and 24, for bipropellant and monopropellant systems respectively, would be capable of reducing the propellant tank pressure and thereby reducing the engine thrust accordingly. A throttling down of approximately 30% should be achievable by the rocket motor. This estimate of thrust reduction is based on the capabilities of a similarly designed 8.9 N engine. The limiting factor on reducing the thrust of the rocket engine (for a bipropellant system) is the fuel injectors. These injectors are designed to operate at a specific propellant pressure and their performance will degrade as the pressure is lowered. Increasing the thrust of the rocket engine is not advisable. Increasing the rocket thrust beyond the design point will stress the engine both pressure-wise and temperature-wise and can cause it to fail. The following analysis estimates the engine performance for a thrust reduction from 35 N to 25 N (approximately 29%). This reduction in thrust corresponds to an Isp decrease of from 295 to 255, respectively.

Since the geometry of the nozzle will remain the same as the combustion pressure is reduced, the flow will become overexpanded within the nozzle. This means that the exit pressure will be less then atmospheric pressure. Compression of the gas back to atmospheric pressure will take place outside the nozzle through a series of shock waves. The combustion chamber pressure for a given thrust level is given by equation 18 [32]. The gas constant (R) and combustion temperature (t_o) for the MMH and NTO propellant was estimated to be 311 J/K Kg and 3395 K, respectively. The Isp was assumed to decrease linearly with thrust and is given in equation 19. The Isp value is based on the Earth's gravitational constant (g_e) of 9.81 m/s^2.

$$P_o = \frac{T + P_a A_e}{\frac{A^*}{\sqrt{Rt_o}} \sqrt{\gamma (\frac{2}{\gamma - 1})^{\frac{\gamma+1}{\gamma-1}}} I_{sp} g_e + \frac{A_e}{(1 + \frac{\gamma - 1}{2} m_e^2)^{\frac{\gamma}{\gamma-1}}}} \tag{18}$$

$$I_{sp} = 4T + 155 \tag{19}$$

Utilizing the combustion chamber pressure from equation 18, the mass flow of propellant and exit pressure can be calculated. These are given by equations 20 and 21, respectively [32].

$$\dot{m} = \frac{A^* p_o}{\sqrt{Rt_o}} \sqrt{\gamma (\frac{2}{\gamma + 1})^{\frac{\gamma+1}{\gamma-1}}} \tag{20}$$

$$\frac{P_o}{P_e} = (1 + \frac{\gamma - 1}{2} M_e^2)^{\frac{\gamma}{\gamma-1}} \tag{21}$$

Because the flow through the nozzle is choked, which means there will be a normal shock at the nozzle throat for the range of combustion chamber pressures encountered in this analysis, the exit Mach number is based on the geometry of the nozzle and is given in equation 21. This Mach number does not change with the variation in combustion chamber pressure. The results of the above analysis are shown in figures 25 and 26. Figure 25 shows the combustion chamber and exit pressure change over the thrust range and figure 26 shows the mass flow and Isp change over the thrust range.

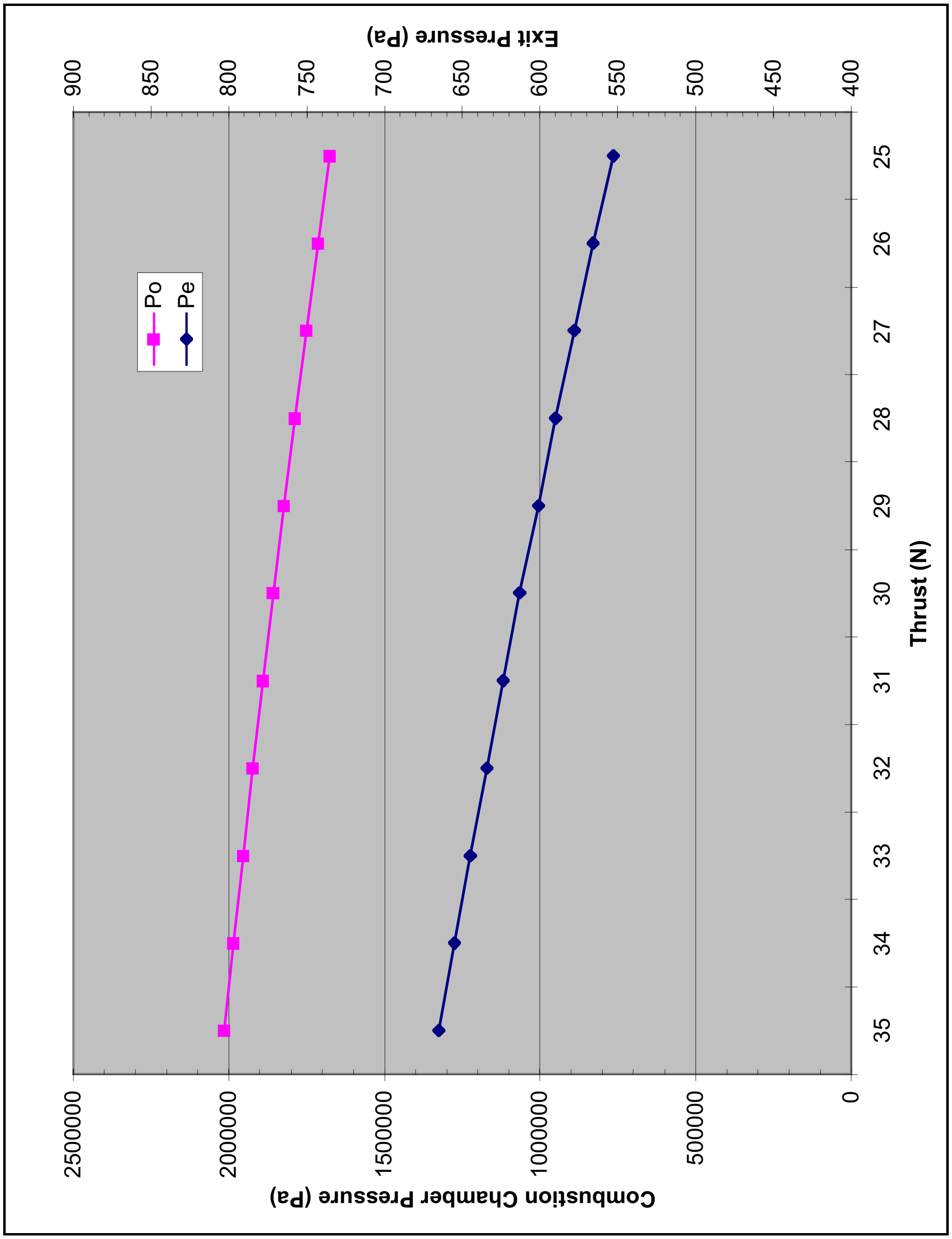

Figure 25 Combustion and Exit Pressures vs Thrust for Off Design Engine Operation

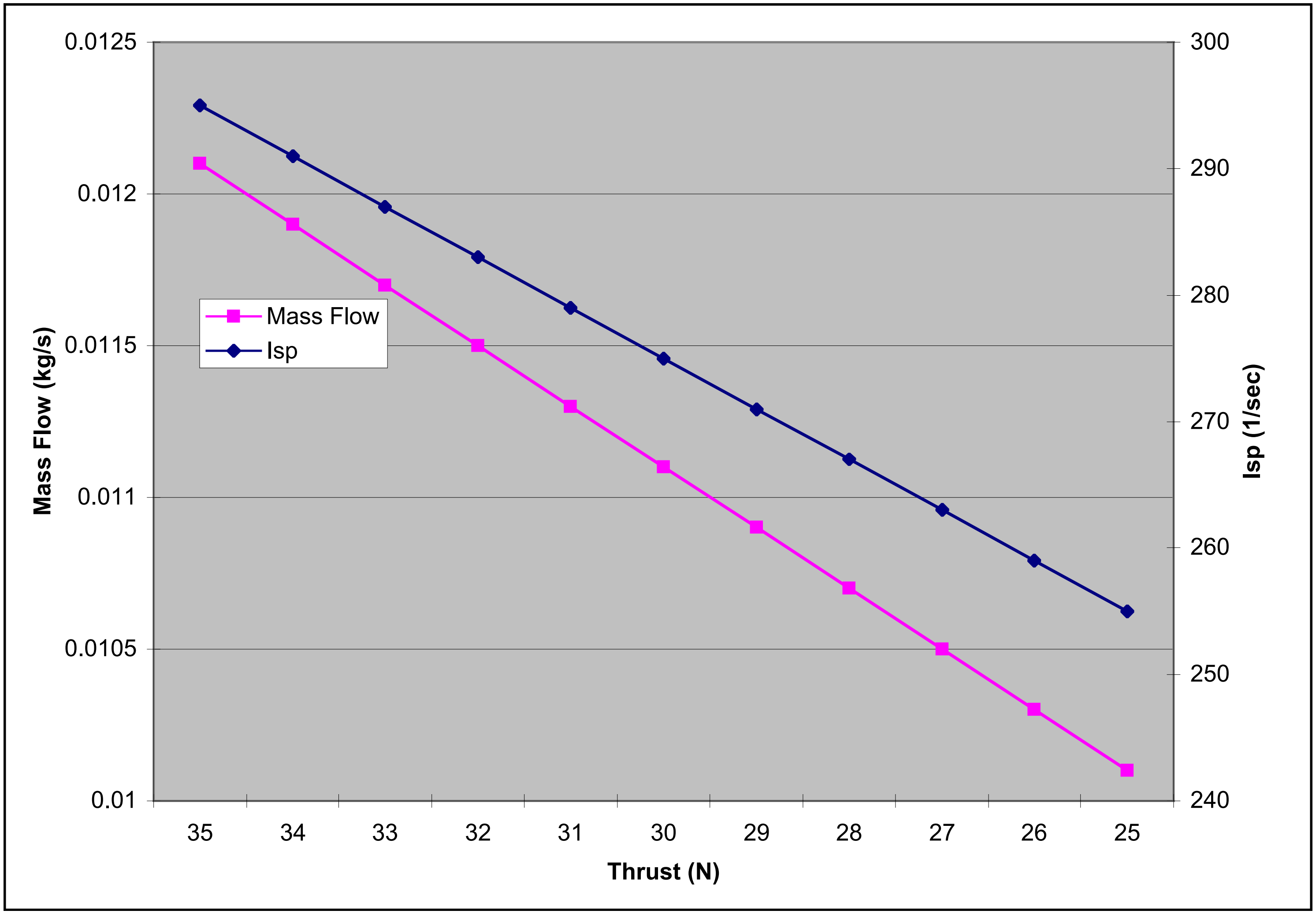

Figure 26 Mass Flow and Isp vs Thrust for Off Design Engine Operation

Thruster Cycling

Controlling the amount of thrust can also be achieved by cycling the rocket motor on and off during operation. This type of operation is represented by the rocket system shown in figures 20 and 23 for a bipropellant and monopropellant system respectively. This cycling produces an integrated thrust value that is less then that of continuous operation. The average thrust value over a period of time will depend on the cycling frequency. There are, however, some performance and mission impacts to cycling the engine to regulate thrust.

First, the performance of the rocket engine will be less than that of steady state operation. This is because on each cycle the rocket engine goes through a startup phase in which transients in injector and chamber pressure reduce performance.

The ability of the rocket motor to withstand the cycling is well established. This mode of operation is common in satellite applications, where the thruster will go through many on / off cycles. Most of the satellite reaction control thrusters have gone through extensive cycling qualification testing. For example in the qualification testing of an Atlantic Research Corporation (ARC) 10 N thruster, it went through over 1 million cycles and over 1,000 cold starts. This testing is very representative of what could be expected for the aircraft rocket system since satellite reaction control thrusters are around the same thrust level as that required by the aircraft. The capabilities of this type of thruster are well matched to this aircraft application. During the propellant evaluation that was done for the Mars micro-mission aircraft program, pulse testing was performed on an ARC 10 N thruster using MMH / MON25 propellant. Tests were performed for on / off pulse rates of 0.1s / 0.1s , 0.2s / 0.2s and 0.5s / 0.5s.

The 0.5s / 0.5s pulse test was terminated because ignition of the propellant was not occurring. The other two pulse tests ran successfully. This lack of ignition would need to be experimentally investigated if MMH and MON-25 were selected as the propellant.

The Isp for the 0.1s pulse test was 208.4 and for the 0.2s pulse test it was 224.1. This compares to an Isp of 288.4 for steady state operation (60 s run time) under the same operating pressure and propellant mixture ratio. The thrust levels for the pulse tests were also significantly reduced. For the steady state case the thrust was 10.28 N compared to 0.81 N for the 0.1 s pulse test and 1.62 N for the 0.2 s pulse. This reduction in Isp is due to the startup effects of cycling the thruster. As the cycle times get larger the effect will diminish. Figure 27 shows an approximation of the effect on Isp from pulsing the engine. This figure is based on the limited data given above.

In another series of tests performed by Kaiser Marquardt, also for the Mars micro-mission aircraft program, ignition delay was plotted versus thrust level. At the design thrust level (10 N) of the engine, the ignition delay was 7.5 ms. As the operational thrust was reduced this delay increased to 11.5 ms at 4.5 N thrust.

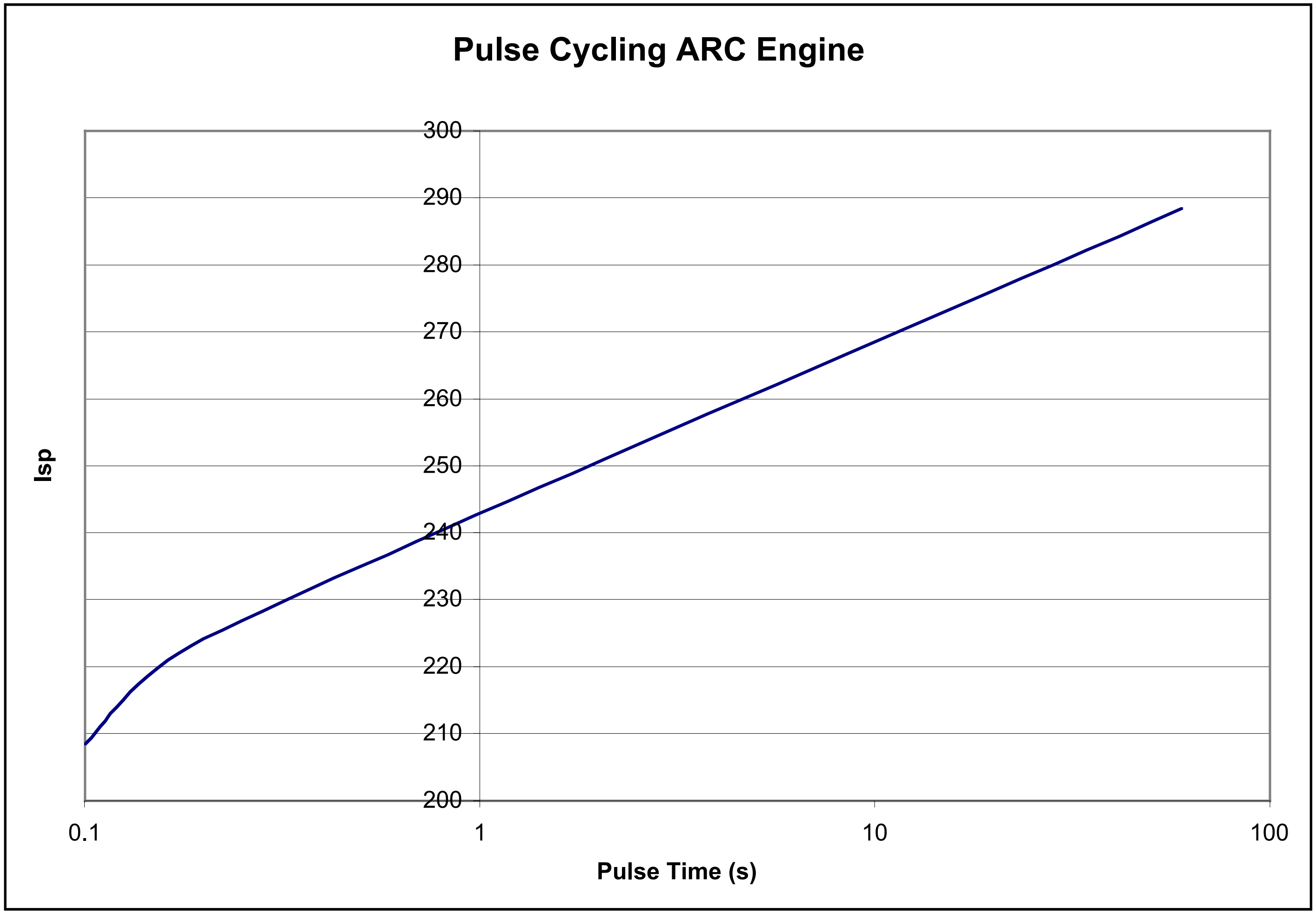

Figure 27 Estimate of the Effect of Pulse Duration on Isp [16]

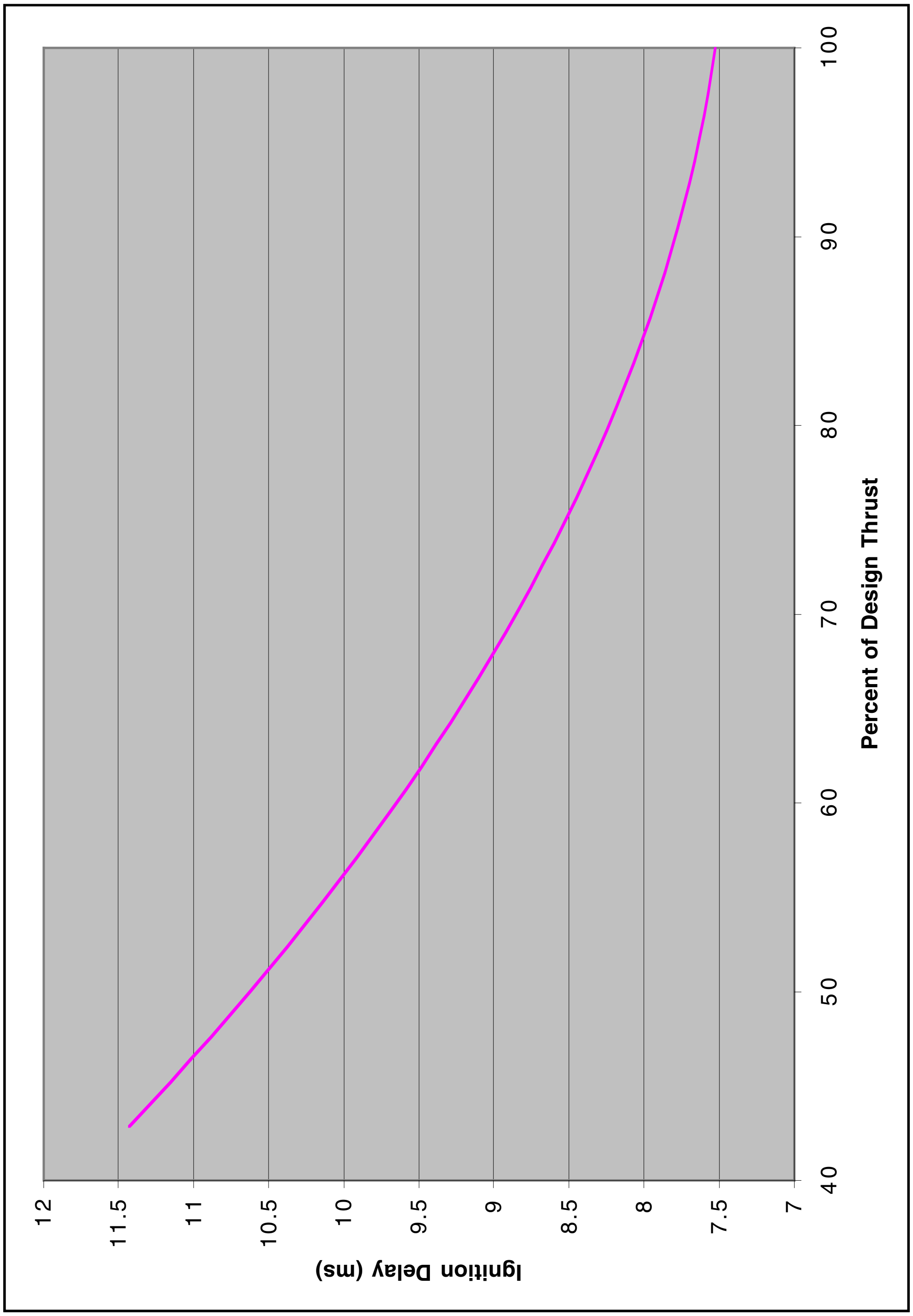

Figure 28 Estimate of Ignition Delay as a Function of Design Thrust [16]

This behavior should be consistent with higher thrust level engines closer to the 35N that was assumed for this analysis. Figure 28 was generated from the data supplied by Marquardt and is an estimate of how pulsing is affected by off-design operation. From figure 28 it can be seen that as the engine is operated further from its design thrust level, ignition delay increases. Therefore the reduction in performance due to pulsing the engine will increase as the engine is operated further from its design thrust level.

An additional characteristic of pulsing the engine is the loading or vibration it places on the vehicle. This does not affect the engine's performance but it may have an impact on the science instruments or other hardware on the aircraft. A plot of a start-up acceleration profile is shown in figure 29. This figure was generated by Kaiser Marquardt during the testing of their 10 N thruster. The higher thrust level of the 35N thruster may increase the g-levels experienced, but the profile should be similar. From this figure it can be seen that vibration g-loads of over 30 g's were experienced within the first 2 ms of startup.

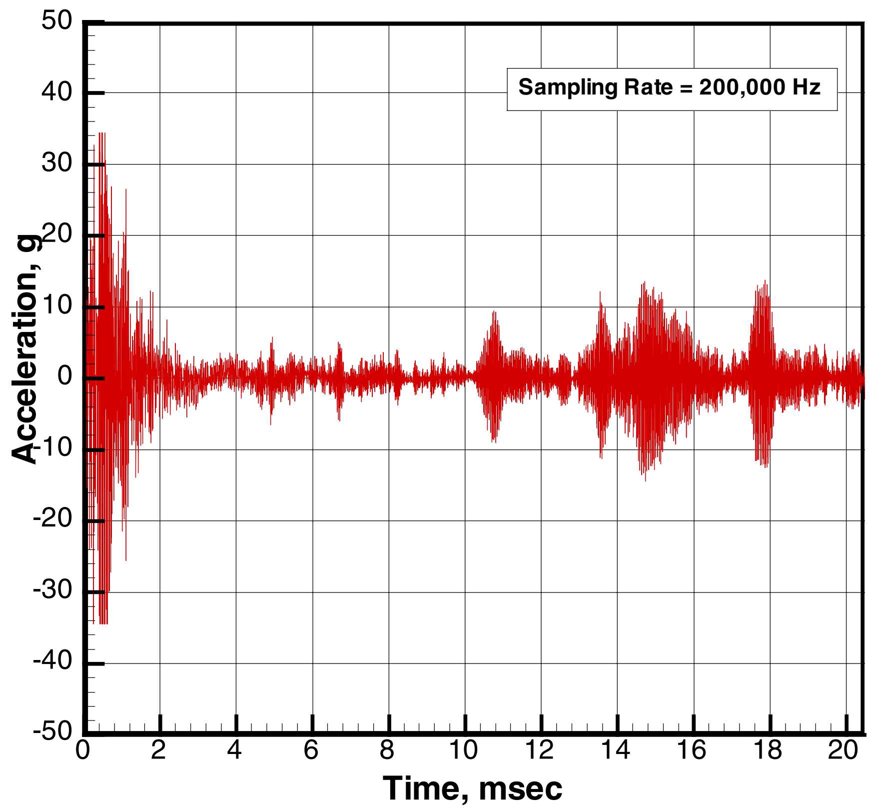

Figure 29 Accelerometer trace of 10N Kaiser Marquardt Startup [17]

Propulsion System Comparison

The main method for evaluating the propulsion systems is by their mass. The aircraft mass is a critical factor in the design feasibility and any reduction in mass can greatly enhance the aircraft's performance and capabilities. Since the systems considered are very different from each other in their design, their technology readiness level (TRL), their operational risk and in the way they affect the other aircraft systems; comparing them is somewhat like comparing apples and oranges. The systems were broken down into three categories: electrical systems, combustion engine systems and rocket systems. The lightest mass system of each of these types is compared for the 1, 2 and 4 hour mission durations. This comparison is shown in figure 30. To try and maintain some consistency between the systems, MMH and NTO were chosen as the propellant for both the bipropellant rocket system and the combustion engines.

Based on the mass comparison shown in figure 30 it can be seen that the 4 cycle internal combustion engine produces the lowest mass system for all mission durations. The fuel cell system is more competitive as the duration of flight increases and is almost equal to the 4 cycle engine for the 4 hour case. Compared to the other systems, the rocket system mass grows considerably as the mission duration increases. Based on this, the rocket system should not be considered for mission durations greater then 2 hours.

In selecting a propulsion system, the desired duration is a critical factor. However other aspects such as development time and cost must also be considered. The rocket system has the lowest performance but has the highest TRL due to its extensive use in the satellite industry. The combustion engine, although widely used on Earth, would require a significant amount of development for use in a Mars aircraft. A new engine would need to be designed, tested and qualified for this application. This is because of the differences in the type of fuel that would be used and the environments of Mars and deep space.

If development cost and time is not an issue, the 4 cycle internal combustion engine propulsion system is the best candidate, for flight durations up to approximately 4 hours. The fuel cell system should also be considered for longer durations (near and greater then 4 hours). Figure 30 indicates that for durations greater then 4 hours the fuel cell system would be the lightest.

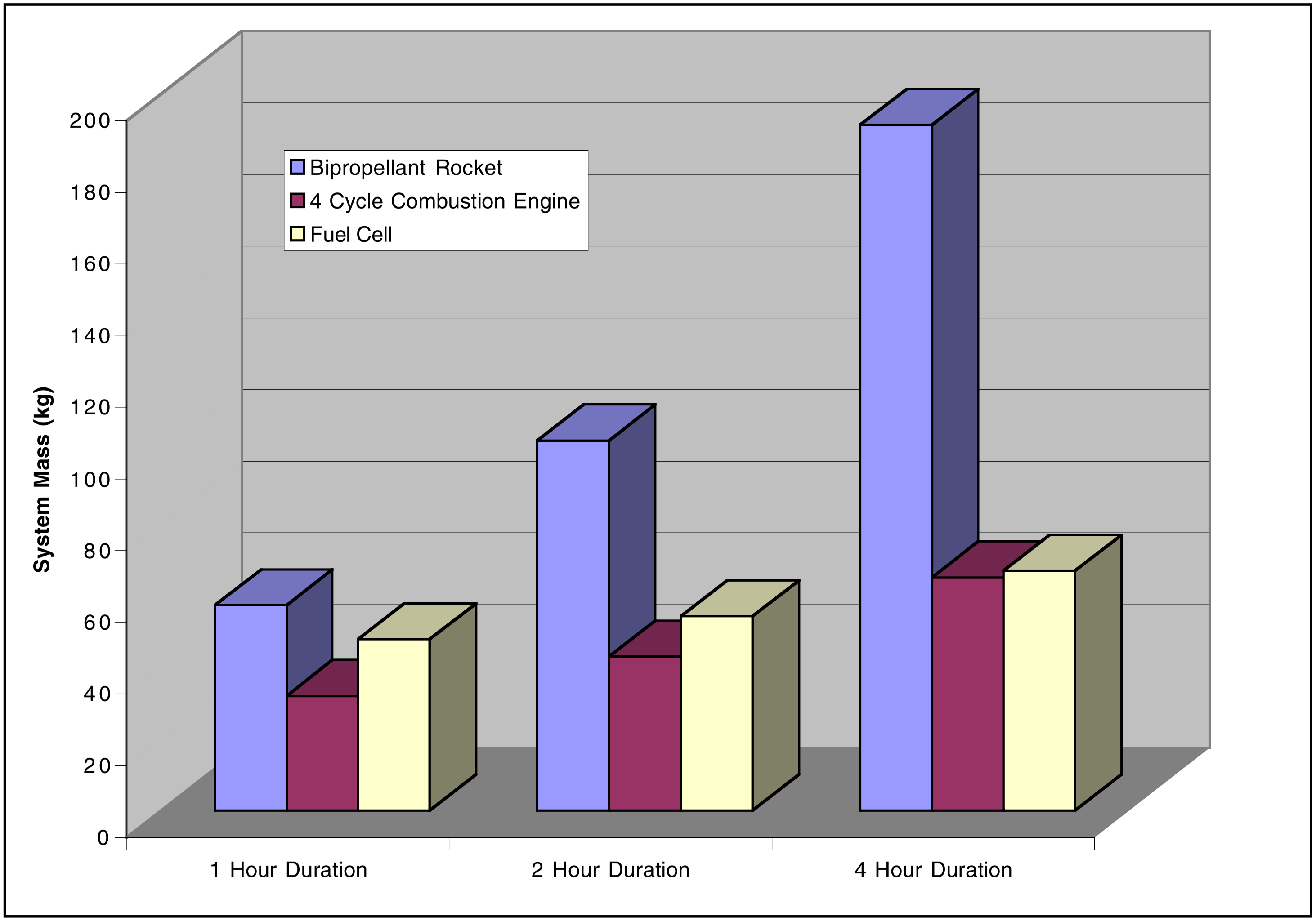

Figure 30 Mass Comparison Between the Various Types of Propulsion Systems

Near-Term System Selection

For a near-term flight of moderate to short duration (approximately 90 minutes or less) where extensive propulsion system development is not possible, a bipropellant fueled rocket system would be the best choice due to its present high level of development (technology readiness level (TRL) of 7-9) and relatively low implementation risk.

The system is based mainly on components from satellite reaction control systems. This provides a lot of design heritage and eliminates the need for space qualification of most of the components within the system. The oxidizer used in this system is nitrogen tetroxide with 25% nitric oxide content (MON-25). This differs from the standard nitrogen tetroxide oxidizer which utilizes 3% nitric oxide content (MON-3). The reason for utilizing the MON-25 oxidizer is that its freezing temperature is –54 °C compared to –11 °C for MON-3. This reduced freezing point is a significant benefit for the system design and reliability. The estimated minimum storage / operating temperature is –40 °C for the propulsion system. Therefore MON-25 eliminates the need for an active heating system during the mission. This reduces the parasitic mass of the propulsion system and its complexity. Also the overall aircraft reliability is increased since there is little to no chance of the oxidizer freezing and thereby causing the propulsion system to fail.

An estimate of the present level of development for the propulsion systems that were considered in the context of a Mars aircraft mission are shown in table 27. A definition of each TRL level is given in appendix A.

System	TRL
Bipropellant Rocket	7 to 9
Monopropellant Rocket	7 to 9
Battery	4 to 6
Fuel Cell	4
Piston Expander Engine	4 to 5
4 Cycle Combustion Engine	4 to 5
2 Cycle Combustion Engine	4 to 5

Table 27 Propulsion System Development Level for a Mars Aircraft Application

The monopropellant rocket system was not selected because its performance was less than that of the bipropellant rocket system. Although it is a simpler system requiring less hardware mass, the increase in fuel consumption due to its reduced performance more then offset the component mass saved. Overall, for the same flight range, the monopropellant system and propellant was heavier then that needed for the bipropellant rocket system.

Two electric propulsion system concepts were studied. The battery powered system weight was comparable to the bipropellant rocket system. This was mainly due to the inability to take advantage of some of the high specific power batteries that are available because of their low rates of discharge. This low discharge rate meant that to meet the continuous power demand of the aircraft, the battery capacity had to be much larger then what was actually needed to meet the mission duration requirement. The battery system also had significant issues with eliminating heat generated during operation as well as volume and packaging constraints within the aircraft. Because of theses concerns the battery system was not selected as the propulsion system.

The fuel cell system was lighter then the bipropellant rocket system. However the relative newness of the technology and the complete lack of experience in utilizing a fuel cell as the main power system in an aircraft made it a very high risk item. Because of the relatively low TRL level (4) for a fuel cell in this type of application, it was not considered a viable option for this mission.

Both of the types of combustion engines considered were lighter then the bipropellant rocket system. The main concerns with these systems is operating them on either a bipropellant or monopropellant fuel system. In order to accomplish this, a development program would need to be implement to produce an engine that can operate within the Mars environment. For near term missions or missions with limited development funding, this would not be feasible. Heat transfer issues with operating these engines within the Mars environment are also a concern.

In addition to the items listed above, both the electric and combustion engine systems would require a propeller to generate thrust. There are significant concerns with operating a propeller within the Mars environment. To generate the thrust required, the propeller will need to operate near the sonic limit at its tip. This places the operation of the blade in a low Reynolds number (~15,000) high subsonic Mach number environment. The aerodynamics of operating a propeller within this regime is out of the experience of conventional propeller design. A development program would need to be undertaken to produce a propeller that can confidently operate under these conditions. In addition to its aerodynamic operation, the size of the propeller requires it to be stowed and then deploy from the aeroshell with the aircraft. Due to the aerodynamics and required deployment, the propeller represents a considerable risk item for the operation of the aircraft. Eliminating the need for a propeller significantly reduces the overall operational risk of the aircraft and is a significant factor in selecting the bipropellant rocket system as the baseline propulsion system for a near-term Mars aircraft.

Appendix A.—TRL Level Definition

Level 1	Basic Principles Observed and Reported
Level 2	Technology Concept and/or Application Formulated
Level 3	Analytical and Experimental Critical Function and/or Characteristic Proof-of Concept
Level 4	Component and/or Breadboard Validation in Laboratory Environment
Level 5	Component and/or Breadboard Validation in Relevant Environment
Level 6	System or Subsystem Model or Prototype Demonstration in a Relevant Environment (Ground or Space)
Level 7	System Prototype Demonstration in a Space Environment
Level 8	Actual System Completed and Flight Qualified Through Test and Demonstration (Ground or Space)
Level 9	Actual System Flight Proven Through Successful Mission Operations

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The propulsion system is a critical aspect of the performance and feasibility of a Mars aircraft. Propulsion system mass and performance greatly influence the aircraft's design and mission capabilities. Various propulsion systems were analyzed to estimate the system mass necessary for producing 35N of thrust within the Mars environment. Three main categories of propulsion systems were considered: electric systems, combustion engine systems and rocket systems. Also, the system masses were compared for mission durations of 1, 2, and 4 h.

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