VASIMR Engine

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VASIMR is a Variable Specific Impulse Magnetoplasma Rocket engine. It has been in development for several decades but has so far not reached a level of maturity necessary to produce a flight unit. Unlike other electric propulsion engines it has the ability to produce variable impulse simply by varying the power applied to the plasma stream. It makes use of electromagnets to confine the plasma and shape the exhaust, thus doing away electrodes and most moving parts. It has demonstrated performance in a laboratory setting for power levels of 25 kW up to 200 kW, and the testing of the 200 kW variant and the resulting performance will be discussed at length here. A brief history of the engine's development and a performance comparison against chemical and nuclear engines will also be presented. The pros and cons of the design will be weighed for the engine based on its current level of technology readiness.

Nomenclature

Ar = Argon
ICH = Ion Cyclotron Heater
Isp = Specific Impulse
K = Kelvin
N = Newton
Pa = Pascal
P = Thruster power
P_c = Cyclotron power
P_h = Helicon power
RPA = Retarding Potential Analyzer
RF = Radio Frequency
T = Tesla
TRL = Technology Readiness Level
VASIMR = Variable Specific Impulse Magnetoplasma Rocket
ΔV = Change in velocity
W = Watt
η_T = Thruster efficiency

I. Introduction

Since the beginning of the space age over 60 years ago, the vast majority of rocket engines used both for attaining orbit and for use beyond low Earth orbit (LEO) have used some form of chemical propellant. The propellant is ignited to create a gas with very high temperature and pressure which is then released through a nozzle

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at high velocity to provide thrust. While this type of engine has been demonstrated to be extremely useful, the relatively low ISP (few hundreds of seconds) means that enormous amounts of fuel are necessary to propel reasonable payloads beyond LEO. Engines with larger Isp, in theory, could deliver larger payloads for less overall mass and therefore less cost. One such alternative is the VASIMR engine.

VASIMR stands for Variable Specific Impulse Magnetoplasma Rocket. It is an electric propulsion system that uses radio waves to ionize and heat a propellant, and then uses a magnetic field to accelerate the plasma out of the engine. The engine has been the dream of former astronaut Franklin Chang-Diaz since the 1970s.

II. Theory of Operation

The VASIMR engine uses an inert gas such as argon or xenon. The gas is first introduced to a chamber lined with electromagnets. The gas is then subjected to strong electromagnetic waves from helicon RF antenna. The antenna is called a helicon because the shape of the coupler allows it to generate helical RF waves. These waves strip the gas molecules of electrons creating a plasma of ions and electrons. Although the temperature of the plasma at this point in the process is around 5800 K, it is referred to as a “cold plasma”. A schematic of the engine is shown in Figure 1.

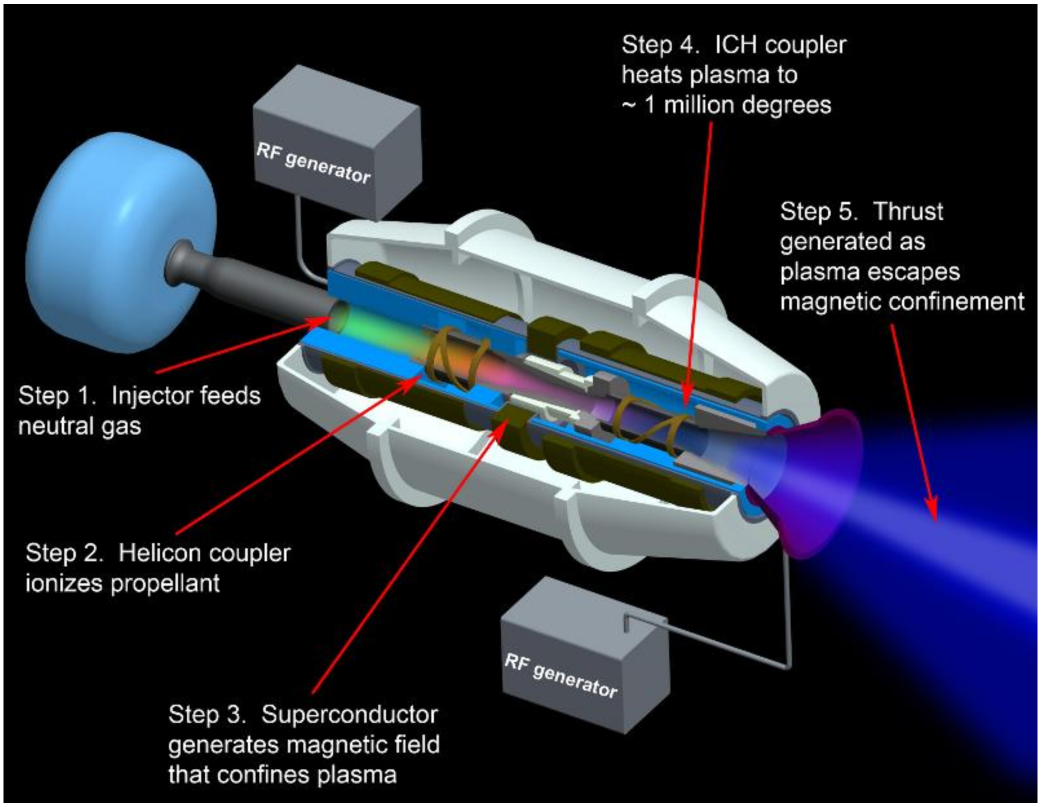


Figure 1 VASIMR engine diagram[1]

Because the particles in the chamber now carry a charge, they can be contained by the magnetic field generated by the electromagnets lining the chamber. The amount of propellant injected into the chamber and the amount of RF energy used to ionize the gas can both be controlled to affect the performance of the engine. In this way the engine is capable of creating low-thrust, high specific impulse exhaust or high-thrust (for an electric engine), low specific impulse exhaust depending on the needs of the mission.

Once the gas has been ionized and heated, another electromagnet is used to compress the gas, similar to the convergent section of a chemical rocket nozzle. At this point a second coupler, known as an Ion Cyclotron Heater (ICH), is used to further heat the ions by bombarding them with RF waves in resonance with the orbits of the ions. This resonance adds efficiency to the transfer of energy to the ions and allows almost all of that energy to be transferred in a single pass cyclotron absorption cycle. The temperature that results is on the order of 1,000,000 K, and the ions and electrons are able to leave the nozzle with a small energy distribution before reaching thermal equilibrium. [1]

III. History of Development

The VASIMR engine has been in development for several decades. Experiments began in the early 1980s at the Massachusetts Institute of Technology (MIT). By the 1990s the original plasma gun was replaced by the helicon plasma creator which made the device operational without the need for an electrode, potentially a huge advantage over other electric rocket engine designs. In 1998 the first helicon experiment was performed at the newly created Advanced Space Propulsion Laboratory (ASPL) at NASA Lyndon B. Johnson Space Center (JSC) in Houston, TX.

The VASIMR name was created in conjunction with the first test of a 10 kW helicon source, dubbed VASIMR Experiment 10 (VX-10). VX-25 and VX-50 followed later, and by 2005 the VX-50 engine had demonstrated full plasma production in the first stage and plasma acceleration in the second stage. This engine was capable of 0.5 N of thrust while using 50 kW of RF energy to produce the plasma. The efficiency of the second stage was calculated at 59%.

In 2005 the Ad Astra Rocket Company (AARC) was formed in order to privatize the VASIMR engine development effort, and former NASA astronaut Franklin Chang Diaz was named chairman and CEO of the new company. Soon after that the company moved off of the JSC site and opened facilities in Webster, TX and in Costa Rica on the campus of Earth University in Liberia.

By 2007, the 100 kilowatt VX-100 engine was operational and demonstrated significant improvements in plasma output over the VX-50 engine. Despite the successful increase in plasma production, the efficiency of the engine did not live up to expectations due to losses in the process that converted electricity to RF energy as well as the high energy overhead associated with running the superconducting magnets that contain the plasma flow. As a result, the VX-100 engine could not match the overall efficiency of other electric engines such as NASA's High Power Electric Propulsion (HiPEP) system which achieved total efficiencies on the order of 80%.

Starting in 2008, Ad Astra began tests with a fully operational VX-200 engine. This version used 30 kW of RF energy to create the plasma, in this case from Argon gas, and the remaining 170 kW was used for the second stage of the engine to further heat and accelerate the plasma. By this stage of development, the energy conversion efficiency of the RF helicon process had improved to ~98%. Overall system efficiency however was still in the range of 60-70%, with most of the remaining 30-40% coming from secondary ionizations caused by the plasma crossing magnetic field lines. Most of this "waste" energy creates heat that must somehow be rejected in the design. Still, the VX-200 demonstrated that it was capable of creating 5 N of thrust with 50 km/sec exhaust velocity yielding an optimal Isp of around 5,000 s, and could maintain this operation for up to 25 seconds. A picture of the VX-200 engine in operation is shown in Figure 2.

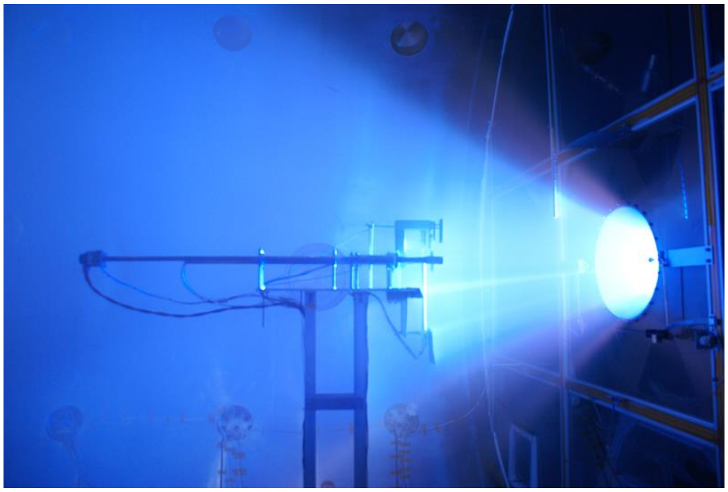


Figure 2 VASIMR test with exhaust measurement instrumentation[1]

The VX-200 was intended to be used as a technology demonstrator on the International Space Station (ISS). The ISS requires periodic orbit boosting to compensate for the high atmospheric drag, and the VASIMR engine would potentially be ideal for this purpose and could yield significant cost savings over the chemical thrusters currently being used for this purpose. The flight unit VX-200 was to have consisted of two VX-100 engines with opposite magnetic poles to cancel out the ~1 Tesla magnetic field that could have resulted in attitude torques due to interaction with the Earth's magnetic field. Several NASA Space Act Agreements were signed with Ad Astra to develop the ISS demonstration, but in 2015 NASA cancelled plans to fly the engine on the ISS any time soon. Part of the difficulty lies in the fact that the total power output for the ISS's solar arrays is 200 kW, so a supplemental power system of some kind would be necessary and the firing durations would be relatively short. Despite the loss of a chance to flight test the engine on the ISS, in 2015 NASA still gave Ad Astra $10 million to help continue to develop the technology.[2]

IV. VX-200 Experimental Setup and Results

The VX-200 is to date the most technically mature version of the VASIMR engine, so we will take a closer look at the testing involved and the performance parameters that resulted as described by Longmier, et al (2011) [3]. The VASIMR engine was never designed to be used inside the atmosphere, so a vacuum chamber is necessary to test the engine and measure the actual thrust generated by it. For the test setup in 2010, the majority of the engine except for the RF generators, power supplies (using facility power rather than solar or nuclear) and the cryocooler were exercised in a vacuum environment. The size of the vacuum chamber was significantly bigger than the engine to allow force measurements downstream of the exhaust and at angles off of the axial line of the engine nozzle. A diagram of the test setup is shown in Figure 3.

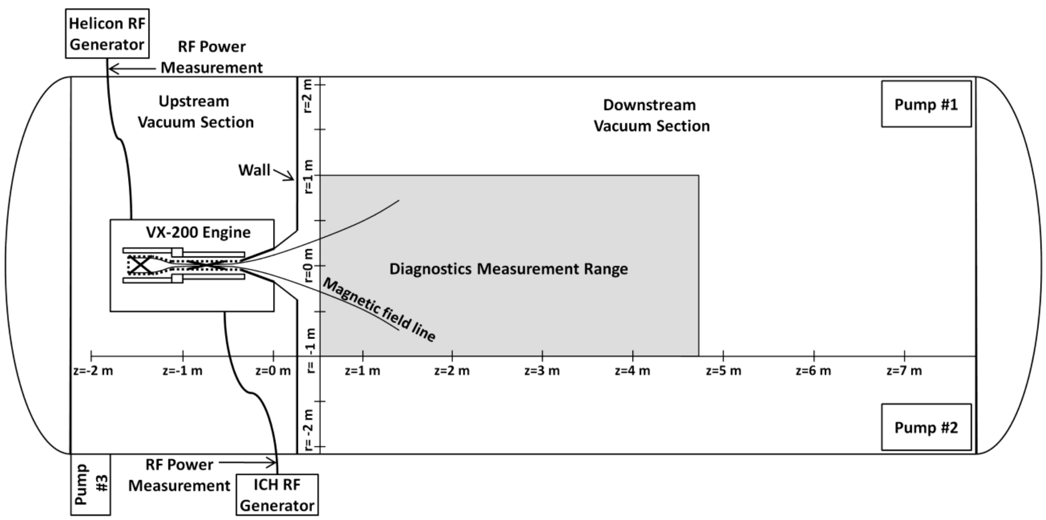


Figure 3 VX-200 test setup inside the vacuum chamber[3]

The helicon generator was rated up to 48 +/-1 kW RF with 91% efficiency and a specific mass of 0.85 kg/kW. The ICH generator was rated up to 172 kW RF with 98% efficiency and a specific mass of 0.51 kg/kW. It should be noted here that these efficiencies are for the electrical RF generation circuits only. They do not represent comparisons of power input to thrust generated. Overall system efficiencies will be discussed later.

The superconducting magnet was cooled to 6 K by a combination cryocooler and cryocompressor that draw up to 15 kW of power and can produce a magnetic field as strong as 2 Tesla. The magnetic field is used to confine the plasma flow, control the speed to allow ion cyclotron coupling with the RF energy. As the magnetic field expands it also converts perpendicular ion kinetic energy to kinetic energy parallel to the axis of the engine, which is the desired thrust vector.

For these experiments two propellants were used, Argon and Krypton, with mass flow rates of 50-160 mg/sec and 100-250 mg/sec respectively. These rates were measured with a calibrated proportional flow control valve and a thermal based mass flow meter positioned at the high pressure section of the propellant feed system.

The vacuum chamber itself was 4.2 m in diameter,10 m long and separated into two sections as shown in Figure 3. A partition was necessary between the two sections to reduce the likelihood of arcing between the plasma flow and the high voltage transmission lines and associated circuitry. The pressure in the engine section was ~1x10^{-3} Pa while the pressure in the exhaust section was ~1x10^{-2} Pa while the engine was firing. The pressure in the exhaust section allowed for a charge mean free path of ~10 cm while the engine was operating.

These tests were typically 30 seconds in duration, and the neutral pressure gradients between the engine and the chamber equalize within the first second of operations. After the first second the operations are considered to be quasi-steady-state for the rest of the test. All of the data for the thruster efficiency calculations presented later are taken from this quasi-steady state timeframe.

A variety of instruments were used to measure the performance of the ion exhaust. An array of molybdenum planar ion flux probes was used to measure the total flux. A set of highly sensitive strain gauges with a resolution of 0.1 mN were used as a plasma momentum flux sensor to measure the force of the plasma exhaust from the engine. This momentum flux sensor was calibrated against the more commonly used inverted pendulum thrust stand using a P5 Hall thruster for comparison. A retarding potential analyzer (RPA) was used to measure the ion energy in the exhaust, which was capable of distinguishing between singly and multiply ionized particles and their respective energy distributions in order to understand the relative contributions from the different species to the overall ion exhaust velocity. Finally, an optical spectrometer was also used to look for spectral lines of neutral Ar, multiply ionized varieties of Ar as well as other impurities to help assess the performance of both the ionizing section of the engine and the efficiency of the cyclotron section. Some of these instruments can be seen to the left of the engine chamber inside the exhaust stream in Figure 4. Figure 5 shows that engine outside of the test chamber.

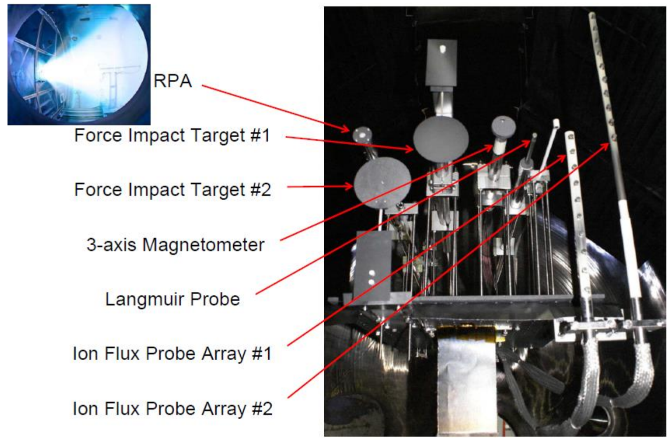


Figure 4 VX-200 test instrument array[4]

Figure 5 VX-200 engine outside the chamber[4]

The engine was operated at a peak magnetic field strength of 2 Tesla, a helicon RF power level of 28 kW, an ICH RF power level of up to 172 kW and a mass flow rate of 107 mg/sec of Ar. The spectrometer did not detect secondary or tertiary ionization states of Ar. The force density and current density were mapped across the area of the exhaust plume to determine how focused the plume was. When including 90% of the total observed ion current density and 90% of the force density, it was determined that the ion current half cone angle was ~30 degrees and the force density half come angle was ~24 degrees. The nozzle efficiency can be defined as the fraction of directed momentum to total ion flow momentum. Using the equation $\eta=1/2(1+\cos\theta)$ with the measured half cone angles yields nozzle efficiencies of 93% and 96% for the current density and force density. The thruster efficiency of the engine is defined as ratio of total thruster power to the total RF power coupled to the plasma. The thruster power is defined as:

$$P = F^2/2\dot{m}$$

where F is the total force as measured by the plasma momentum flux sensor and $\dot{m}$ is the propellant mass flow rate.

The total power coupled to the plasma is equal to the power input to the helicon P_h plus the power input to cyclotron P_c, assuming power to RF conversion efficiencies as outlined above (91%-98%). The thruster efficiency η_T is therefore given as:

$$\eta_T = P / (P_h + P_c)$$

Using $Isp = F/\dot{m}g$, the engine test results are shown in Figure 6 for a helicon power level of 29 kW and a range of ICH power levels between 0 and 172 kW.

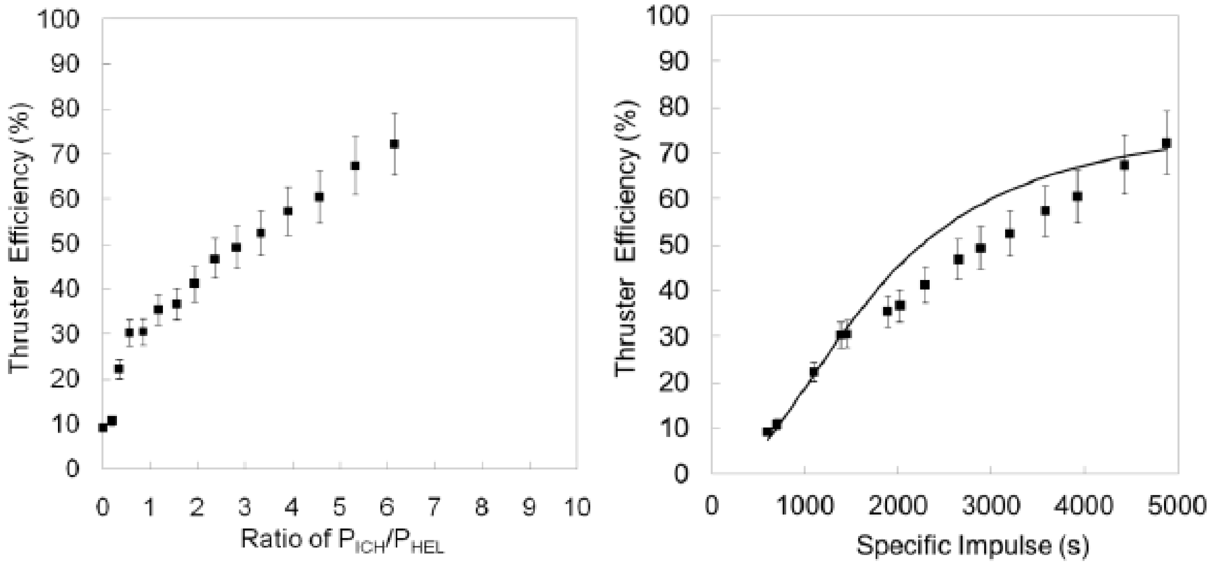


Figure 6 VX-200 Test results showing a) Thruster efficiency vs input power ratio and b) Thruster efficiency vs specific impulse[3]

At maximum power levels the engine demonstrated a total force of 5.8 N and Isp of 4,900 s, and a thruster efficiency of 72% with a propellant mass flow rate of 107 mg/s. The efficiency for the entire system from input DC power to measured thruster power is ~60% with an exhaust velocity of 50 km/s.

Using the following equation from class as a sanity check, with a claimed ionization cost of 80 ev/ion, an ion mass of 0.66×10^{-25}kg and an exhaust velocity of 50,000 m/s, the efficiency is 87%. The experimenters claimed that

$$\eta_p = \frac{1}{1 + \left(\frac{e_{ion}}{0.5 m_{ion} V_e^2} \right)}$$

because the thruster efficiency was still increasing with increasing ICH power that the RF ion heating process was not yet saturated, which may help explain why their measured efficiency was 16% lower than this theoretical limit. The technology is still too immature to qualify as a flight design, so comparisons of specific mass for the engine are not possible.

V. Performance with other elements

The VASIMR engine is designed to allow for variable thrust to accommodate multiple mission scenarios. One of those missions might be transport of a payload from LEO to GEO, where solar panels would be used rather than nuclear energy for power generation. This would necessitate a lower Isp, as can be seen from Figure 6b, where the ratio of ICH power to helicon power more readily accommodates typical power levels available from existing solar panel technologies.

For lower specific impulse values it is necessary to use propellants with larger amu (than Ar) in order to achieve efficiencies greater than 60%. One possibility would be Krypton due to its atomic mass of 84 amu and an ionization energy somewhat less than Ar. Figure 7 shows modeled total system performance of the engine for various elements. Even an element as light as hydrogen can in fact achieve 60% efficiency, but it requires an Isp greater than 30,000 s in order to do it. Figure 7 also shows measured Ar values up to 212 kW for comparison.

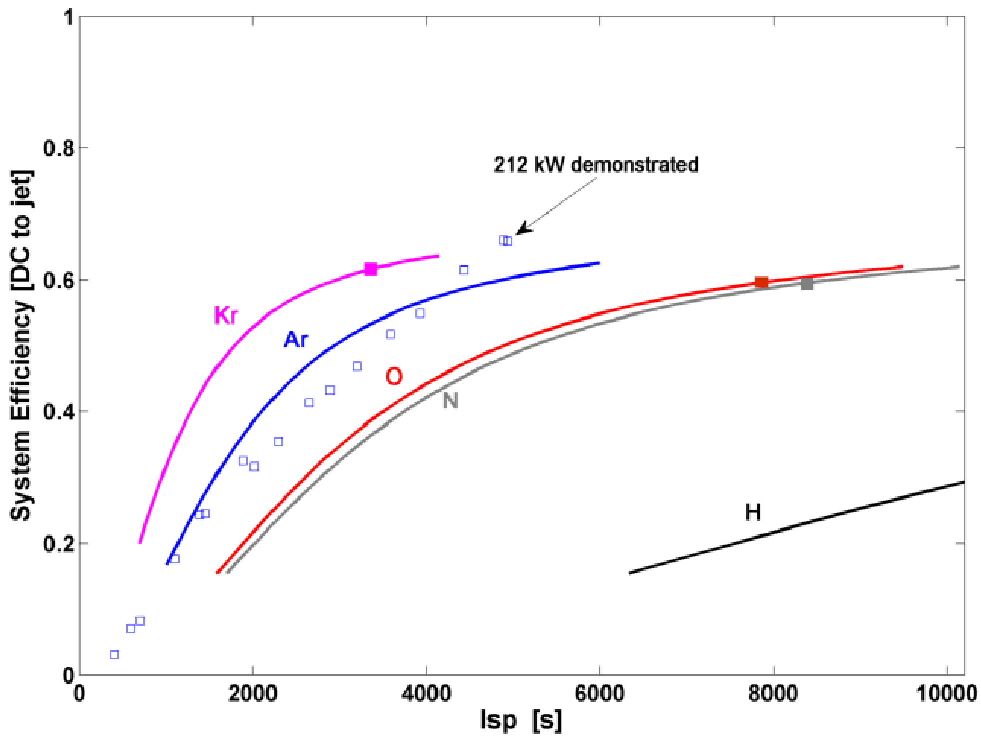


Figure 7 Modeled VX-200 performance for several elements as well as actual measurement points for Ar[3]

VI. Application for an actual mission

Tim Glover, the Director of Development at Ad Astra, presented a mission-specific comparison of the VASIMR engine to both chemical and nuclear powered engine architectures[4]. The mission chosen was a simplified Earth to Mars orbital transfer, using a Hohmann transfer for the chemical and nuclear rockets to Mars orbital insertion (MOI) at an altitude of 250 km.

Assumptions include starting in a 400 km LEO orbit with a 100 mt vehicle and an available Isp of 450 seconds for the chemical rocket and 925 seconds for a nuclear rocket (taken from a version of the NERVA engine).

V_{LEO} at 400 km = 7,676 m/s
V_{TO} perihelion = 32,670 m/s
V_{Earth} = 29,750 m/s
Velocity needed for TMI burn = 32,670 – 29,750 = 2,920 m/s

Post TMI vel = $\sqrt{v_\infty^2 + \frac{2GM_E}{r_{LEO}}}$ =11,241 m/s

ΔV1 = 11,240 – 7,676 = 3,565 m/s

Mars transfer orbit aphelion = 21,490 m/s
Mars Vorbit = 24,130 m/s
Vapproach = 24,130 – 21,490 = 2,640 m/s
Pre-MOI at 250 km periapsis of Mars orbit V= $\sqrt{v_\infty^2 + \frac{2GM_M}{r_{250}}}$ = 5,513 m/s

$$\sqrt{\frac{GM_M}{a} \cdot \frac{r_a}{r_p}}$$

250 km periapsis Vpost-MOI = = 4,619 m/s

$\Delta V2 = 5{,}513 - 4{,}619 = 894$ m/s

Using $\Delta V = I_{sp} g_o \ln(m_i/m_f)$ and iterating for propellant masses for each stage yields a payload mass of 29 mt for the chemical rocket, assuming a two-stage TMI burn and that the tanks are dropped along the way. The flight time is 9 months.

Using $\Delta V = I_{sp} g_o \ln(m_i/m_f)$ with a nuclear thermal system with liquid hydrogen propellant and assuming the engine mass is 6% of the initial mass and separate tanks are used for TMI and MOI maneuvers, the payload mass comes out to 44 mt. The flight time is also 9 months.

Using the VASIMR engine and a 500 kW solar array (more than twice that of the entire ISS), a 62 mt payload could be delivered to Mars orbit, but it would take 52 months to do so. In order to achieve an 8 month transfer time similar to that of the chemical or nuclear rocket, the mass of the propellant plus engine would be so high that there would be no mass left over for the payload. It is interesting that for power levels below 1 MW, there is no savings in

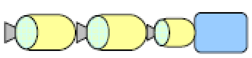
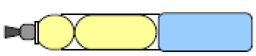
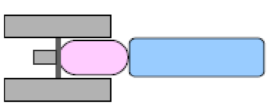
100 mT IMLEO System	Specific Impulse (seconds)	Payload from 400 km LEO to 1-sol Mars orbit (mT)	Total Flight Time (months)
Chemical 	450	29	9
Nuclear Thermal 	925	44	9
1 MW Solar Electric 	5000	48 – 58 ($\alpha = 20$, $\alpha = 10$)	*27*

Figure 8 Comparison of payload and transit time for chemical, nuclear and VASIMR engine architectures[4]

propellant mass, only a reduction in transit time. This limit of 1 MW offers the best trade off of time versus payload, but would require a power source nearly five times that of the ISS. Figure 8 summarizes the result of this trade study. While the assumptions that went into the assessment are at the "rough draft" level, the fact that it was produced by someone at Ad Astra means that it was most likely the most favorable comparison that could be made for the VASIMR design.

VII. Conclusion - Engine pros and cons

The VASIMR engine has several advantages over other electric propulsion methods. The cyclotron heating process is able to transfer nearly all of the RF energy to the ions and have them exit through the exhaust before they lose energy and thermally stabilize with the rest of the chamber. This helps significantly with the overall efficiency of the design. By using electromagnets to confine the ion stream, the engine also does not need to use electrodes. Typical electrothermal engines have electrodes that need to be able to withstand very high temperatures, and they can suffer from electrode erosion due to arc impingement. The heating is also less efficient due to localized heat concentrations near the arc and because of arc instabilities. The magnetic containment of the plasma stream eliminates not just electrodes but virtually all moving parts all together, which gives it an advantage over other designs in terms of simplicity and durability. The variable thrust also allows for applications across a wide range of missions, but it is not clear if that utility balances out with the increase in mass overhead.

While the magnetic containment makes for a good design, the enormous magnetic fields necessary cause other design implications. As stated previously, it is necessary to put two of the engines together to zero out the torque that might be experienced by having the engine interact with an external magnetic field. The effects of high magnetic fields on other electronics or potential crewmembers would also need to be mitigated.

Perhaps the biggest drawbacks involve the enormous amounts of power that would be required to run the engine as well as the massive radiators necessary to reject the excess heat from the process. If solar panels were used at current conversion capabilities of 10 kg/kW, then the 1 MW engine would require 10,000 kg of solar panels. Nuclear energy converted to electricity would be more practical, but is currently unavailable. Assuming the engine is 70% efficient, it would still need to reject 0.3 x 1,000,000W = 300,000 W. The temperature of the exhaust can reach 1,000,000 K. Assuming the radiators can reach 1000 K, 300,000 $W=\sigma T^4 S$ which gives a surface area S of 5.3? m^2. Electric engines that need to reject heat could potentially make use of thermal electric generators to turn some of the waste heat back into electricity, although the conversion efficiency may not be worth the mass penalty.

Ad Astra and Franklin Diaz have claimed as recently as 2016 that the VASIMR engine could get a spaceship to Mars in just 39 days. This claim has led to Ad Astra receiving tens of millions of dollars from NASA to develop the engine. Even they admit that it would require a 10-20 MW power source in order to accomplish[5]. For comparison, the newest nuclear powered aircraft carriers have reactors that produce 300 MW of electricity[6]. Mark Zubrin, the president of the Mars Society, has gone so far as to call the VASIMR engine a hoax, saying it would require, "nuclear electric power systems with 10,000 times the power and 1/100th the mass per unit power as any that have ever been built." [7]

It is clear even from published papers from their own people that the engine is not currently ready for prime time. Without a nuclear component to the engine it is difficult to see it serving as an improvement over current and higher TRL technology, and with nuclear power it is difficult to see how the required amount of heat could be rejected without significantly increasing the mass beyond a useful limit. It does promise the possibility of greater payloads, but only if the customer is very patient with regard to transit times.

VIII. Acknowledgments

This study makes use of data made available by Wikipedia, where needed. The data are used for educational purposes only and we believe the use satisfies the "fair use" clause.

IX. References

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