MAPPP Nuclear Mission

Mars Activation Plan for Permanent Presence

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I. MISSION STATEMENT

MAPPP's prime goal is to send a crew of 50 people to the Martian surface in order to build a preliminary Mars base and to then set up the orbital infrastructure required for the function of such a base. This includes habitation facilities, In-Situ Resource Utilization (ISRU) equipment, landing pads, and a global navigation/communication satellite network. The total mission duration will be 6 months, with the vehicle spending one month going to Mars, 4 months in Martian orbit, then returning to Earth with most or all of the crew over the course of another month.

II. DEVELOPMENT

Systems design will be accomplished through traditional methods while the development of a construction plan for LEO would likely utilize NASA underwater testing facilities on the Earth's surface, taking roughly a decade to complete planning. The rocket would be constructed in LEO over the course of 5 years using a large number of launches of conventional launch vehicles, likely Starships or Starship-derived vehicles. The vehicle will be constructed in LEO due to its proximity to Earth. A project of this scale will require on-site human intervention and LEO is the safest place from radiation and allows crews to quickly return to Earth in the event of an emergency.

III. MEDUSA ORBITAL TRAJECTORIES

The MAPPP vehicle will be a nuclear pulse propulsion vehicle similar to the previously proposed *Medusa* vehicle, thus it will be referred to as such throughout the remainder of our proposal. Medusa uses a large sail in tension to redirect the energy of nuclear pulse charges into thrust, allowing for much better specific impulse than other nuclear pulse propulsion concepts. Medusa will be assembled in LEO at an altitude of 500km through several dozen launches of in-development reusable launch vehicles. Assembling the 5000t vehicle on the ground would require an unreasonably large booster to lift it into orbit (see: *Appendix*). Once the Medusa vehicle is assembled and has undergone checkouts it would be chemically boosted into an elliptical orbit around Earth with an apoapsis of 100,000km. This is done in order to allow the spacecraft to operate its nuclear drive without risk of damaging other satellites or life on Earth, as it will be far enough outside the Van Allen Belts. The spacecraft would then ensure the nominal operation of its engine by circularizing its orbit at apoapsis. At this point, the crew of the vehicle would be launched and rendezvous with the vehicle in high Earth orbit (HEO). Finally, Medusa would start its trans-martian injection burn.

The Medusa vehicle would do a large enough burn in HEO to enable a 1-month transfer to Mars, then propulsively inject into Mars orbit at an inclination of 55 deg (see: Section VIII.c). Satellites would then be deployed and allowed to precess around Mars and landing vehicles would begin ferring payloads from the vehicle to the Martian surface. These landers would be powered by chemical engines. There would be at least two return vehicles landed as well, either of which would be capable of returning the whole crew to orbit along with surface samples for analysis back on Earth.

After approximately 4 months at Mars the Medusa vehicle would do its trans-Earth injection burn; the vehicle does not need to be boosted into a higher orbit as Mars has no appreciable magnetic field to trap charged particles and subsequently damage the satellite constellation. After the month-long return coast to Earth, Medusa would inject itself into a highly elliptical orbit around Earth then slowly aerobrake back down to LEO using its sail as a drag surface. The crew would be returned to Earth before aerobraking using a separately launched vehicle designed with heat shields for a separate aerobraking descent. Final circularization in LEO would occur using attitude control thrusters.

IV. SATELLITE CONSTELLATION

Some proposed maneuvers necessary for this mission include an orbital transfer from Earth's orbit to an orbit on Mars. We would be deploying 20 approximately 500 kg satellites into Martian orbit. These GPS satellites should be in different longitudes of the ascending node to have almost full ground coverage for future missions on Mars. We would be using a satellite constellation similar to Starlink. Each satellite will use laser communication with each other, which means that the attitude of each satellite needs to be controlled precisely so that the communications with each other are clean.

V. SYSTEMS ARCHITECTURE

Table 1: *Performance Characteristics of Medusa Spacecraft*

Medusa Engine Average Thrust:	1.96MN
Engine Isp:	10,000s
Earth to Mars DeltaV Capability (with payload):	25km/s
Mars to Earth DeltaV Capability (no payload):	25km/s
Payload:	2,500t

The Medusa capsule would necessarily be constructed from a refractory metal such as tungsten or molybdenum to withstand the extremely high temperatures from nuclear pulses, with 3000 degrees C being a rough minimum (Sublette 5.3.1.2). Additionally, preliminary research reveals that a nuclear pulse propulsion sail as described in the Medusa project may actually perform more efficiently and allow for smoother acceleration for the safety of the crew than would an *Orion*-derivative rocket—a more prominent nuclear pulse propulsion vehicle in the literature. Further analysis must be done to determine the feasibility of such a propulsion method, as it involves detonating nuclear explosions in front of the capsule, wherein a sail will capture the impulse and propel the spacecraft forward a short time later after unwinding a winch structure (Solem 1–3). Both designs will require significant analysis of material performance in order to ensure the capsule is not appreciably damaged by either energized particles or extreme temperatures during the mission, particularly in order to ensure radiation exposure is within safe limits for the crew.

VI. NUCLEAR DRIVE RISK MITIGATION

One source of danger to the crew during the MAPPP mission is the radiation from the nuclear pulse drive. This will be mitigated through the use of radiation shielding and by putting the bulk of the vehicle at a large distance from the nuclear detonation, at least 2km. Firstly, the crew compartment will be placed at the back of the capsule section of the Medusa vehicle, allowing the cargo storage area, fuel assemblies, and any additional machinery to act as radiation shields for the crewed section of the vehicle. The remaining requirement for shielding will be met by dedicated radiation shields, however the optimal design of these shields is not a focus of this paper. These shields are assumed to be a part of the 500t set aside for the capsule component of the Medusa vehicle.

The Medusa vehicle completely removes any risk of danger to the crew from the impulsive nature of nuclear blast through a ratchet-motor mechanism. During the nuclear blast the ratchet mechanism allows the tether to be freely reeled out and the sail to be accelerated in order to absorb the force from the nuclear blast. The impulsive periods of the blast only lasts for a few milliseconds, and during the rest of the time the motor is used to reel the sail back in, slowing down the sail and accelerating the Medusa spacecraft itself. The mass of the sail is so large that each pulse only causes a maximum of a few m/s of relative velocity between the sail and Medusa spacecraft, so does affect the required length of the tether.

VII. MISSION TIMELINE

Table 2: *Mission Timeline*

Medusa LEO Construction	**Pre-Launch**	**Earth Ejection Burn and Coast**	**Operations at Mars**	**Return to Earth**
2020-2035	1 month	1 month	4 months	1 month
5 yr Rocket Development 5 yr LEO Construction Plan Development 5 yr Vehicle Construction in LEO	Verify Health of all vehicle components and payloads, boost vehicle into HEO.	Crew is delivered to vehicle Vehicle ejects from Earth and coasts to Mars	Vehicle injects into Low Martian Orbit Satellite release along high inclination orbit from Medusa Gradual Satellite orbital readjustment through ion thrusters and exploiting J2 perturbations Landers deliver payloads to Martian surface and crews begin setting up Martian base	Vehicle ejects from Mars, coasts to Earth, and then injects into HEO Crew is retrieved from vehicle and returned to Earth

VIII. ORBITAL ELEMENTS

A. Earth (LEO to HEO Hohmann Transfer) [Geocentric]

Table 3: *Medusa Geocentric Transfer Elements*

Earth	**Semi-Major Axis** (km)	**Eccentricity**	**Inclination** (degrees)	**Longitude of Ascending Node (degree)**	**Orbital Period** (hours)
LEO	6878	0	39	120	1.57
HEO	106,378	0	39	120	95.9

For the first stage of the mission in Earth orbits, the initial parking orbit and final high Earth orbit result are assumed to be circular given our propulsion adjustments. Thus, the semi-major axes of both orbits are the altitude (500km and 100,000km) added to the radius of the Earth, which is estimated to be around 6378km roughly (Williams). The inclinations of both circular orbits are assumed to be the same, so no change in inclination between the two orbits was taken into account for the Hohmann transfer. Inclinations and Longitude of the Ascending nodes are given for an initial launch from the Kennedy Space center in Florida (Dismukes). Given the gravitational parameter of Earth and each of the orbits' semi-major axes, the orbital period for each orbit was calculated in hours. All elements are geocentric orbital elements.

B. Earth-Mars Transfer Trajectory [Heliocentric]

Table 4: *Medusa Earth-Mars Transfer Elements*

Semi-Major Axis (AU)	**Eccentricity**	**Inclination** (degrees)	**Longitude of Ascending Node** (degrees)	**Argument of Periapsis**	**Eccentric Anomaly**
4.3404	0.8296	1.592	77.513	57.069	11.178

C. Satellites [Mars-Centered Origin]

Table 5: *Martian Satellite Constellation Elements*

Satellite Numbers	**Semi-Major Axis** (km)	**Eccentricity**	**Inclination** (degrees)	**Longitude of Ascending Node**	**Time of Periapsis Passage** (hours)
1-4	6855	0	55	Equally Spaced Across Groups	Equally Spaced Within Groups
5-8	6855	0	55	Equally Spaced Across Groups	Equally Spaced Within Groups
9-12	6855	0	55	Equally Spaced Across Groups	Equally Spaced Within Groups
13-16	6855	0	55	Equally Spaced Across Groups	Equally Spaced Within Groups
17-20	6855	0	55	Equally Spaced Across Groups	Equally Spaced Within Groups
21-24	6855	0	55	Equally Spaced Across Groups	Equally Spaced Within Groups

Each of the satellites will be in a circular orbit, thus, the eccentricity is 0 and the argument of periapsis is not well-defined. We chose the period and inclination based on the US GNSS constellation. The US GNSS constellation includes 24 satellites all with 55 degrees of inclination in 6 equally spaced orbital planes ("GNSS"). Thus, each orbital plane has a longitude of the ascending node that is 60 degrees from the next plane. The GNSS satellites have an orbital period of half a sidereal day (about 12 hours 37 mins for Mars) ("GNSS"). From this, the semimajor axis can be calculated. For each satellite in the same orbital plane, they should be equally spaced out in their mutual orbit. Since the period of their orbit is about 12 hours, the time of periapsis passage for the four satellites in each orbit should be spaced by 3 hours.

IX. PERTURBATIONS

A. Satellites

The main perturbation on the satellites is the J2 gravitational perturbation. We will be exploiting nodal regression for satellite separation. Apsidal rotation is not a concern, since the satellites are in circular orbits, so there is no well-defined periapsis to be rotated. Additionally, 3rd-body gravitational perturbations from Phobos and Deimos are well approximated as perturbers with circular orbits, as the eccentricity of Phobos is ~ 0.0151 km and Deimos is even less at ~ 0.0005 km (Williams). This once again only leads to nodal regression and apsidal rotation as the only secular perturbations, which aren't considered for the same reason as above. However, Mars also has a significant J3 term. This will cause secular inclination changes since it is an odd harmonic.

Mars's atmosphere is 1% of Earth's ("ESA"). Given this fact and given that the orbital altitude is roughly 6855km, this effect is negligible on the satellite constellation's orbit within their operational time spans. When approximating the atmosphere of Mars using just 1% of the densities of Earth's atmosphere at the same altitudes and for a satellite of the geometry CdA/m = 1/50m^2/kg, the semi-major axis is altered by roughly only -8.2133e-17 km/rev and the inclination by roughly -7.3268e-22 km/rev; all other orbital elements would remain constant due to the circular orbit having an eccentricity of zero ("The U.S."). Once the satellites are in their nominal orbits, nodal regression is unlikely to have any relative changes to one another, as the satellites will still stay equally spaced due to being at the same altitude and inclination. The only significant station-keeping maneuver we will need is to protect against J3 secular inclination changes.

B. Earth-Mars Transfer

While there are numerous perturbations which may affect the heliocentric leg of our transfer such as solar radiation pressure or gravity perturbations—from nearby natural bodies such as the Moon or Jupiter—these are ignored for the initial calculation of delta-v requirements. The geomagnetic perturbations from the Earth are also ignored, as our heliocentric burn begins outside of the Van Allen Belts. The main force which will be driving the delta-v requirement is the gravitational force of the sun, thus our first approximation can isolate only this effect into account as a coarse approximation for the mission.

Additionally, the initial calculations ignore any delta-v required for entry and exit from the sphere of influence of a given planet because, as compared to the heliocentric transfer and other phases already precalculated, they will have only a small effect on our delta-v budget for the mission.

X. ATTITUDE CONTROL

A. Satellites

Since our satellites will use laser communication to send information to each other, we would need a good ADCS system. The attitude control requirements would fall into the 0.1-degree to1-degree range. On each satellite, we will have redundant reaction wheels for fine pointing, and RCS thrusters to perform momentum dumps. For attitude determination, we would use a sun sensor, star trackers, as well as gyroscopes for angular rate measurements. Attitude determination will be harder around Mars than around Earth since Mars lacks a magnetic field, but with good enough optimal state estimation, the satellites should have a more accurate measure of their orientation. The multiple redundancies are used to ensure that the satellite constellation will be operational for many years—with a minimum life expectancy of at least ten years—such that future missions can be designed to utilize it as a reliable tool and minimize the mass and complexity of their own system requirements,

B. Medusa Vehicle

The main attitude control requirement of the Medusa vehicle is to control the direction in which thrust is applied during burns. The attitude control requirement would be between 0.1-degree and 1-degree. Any more than this would result in a rather significant off trajectory thrust vector. Medusa will use an array of chemical bipropellant hypergolic thrusters for attitude control and small correction maneuvers, and the engine itself is capable of directed thrust by launching bombs slightly offset in one direction. The Medusa vehicle cannot be modeled as a rigid body due to the lack of rigid connection between the sail and the rest of the vehicle, which means that it is very difficult to predict the specific attitude control requirements and techniques required for the vehicle. One technique that has been previously proposed is to spin the vehicle along its long axis to use centrifugal force to keep the sail from collapsing during maneuvers. The vehicle will likely require its communication equipment to be mounted in gimbals so that they can be positioned independently of the rest of the vehicle to maintain contact with Earth.

XI. LAMBERT-SOLVER IMPLEMENTATION

The potential departure dates for our launch from LEO was determined by first checking a register of pre-calculated trajectories from Earth to Mars between 2030 and 2040, with a max delta-v requirement of 20km/s, minimizing for mission duration (Foster). The general trend of the trajectories showed that the most optimal time for launch would be between 2035 and 2036, as indicated by a trough in mission duration, thus giving us a search window of between 2035-01-01 and 2035-11-01 for our solver. The initial position and velocity data for Earth and Mars were taken for this range using JPL Horizons ephemeris generator (Park). The position and velocity of Earth at a time t with respect to the Sun were then used as the initial conditions for the solver, with the initial position of Mars at a time t + dt, where dt is the mission duration. The code then checks for every possible elliptical and hyperbolic orbit given the required mission duration, for every possible departure and arrival date, and then plots the minimum delta-v requirements for each possible dt given our date search.

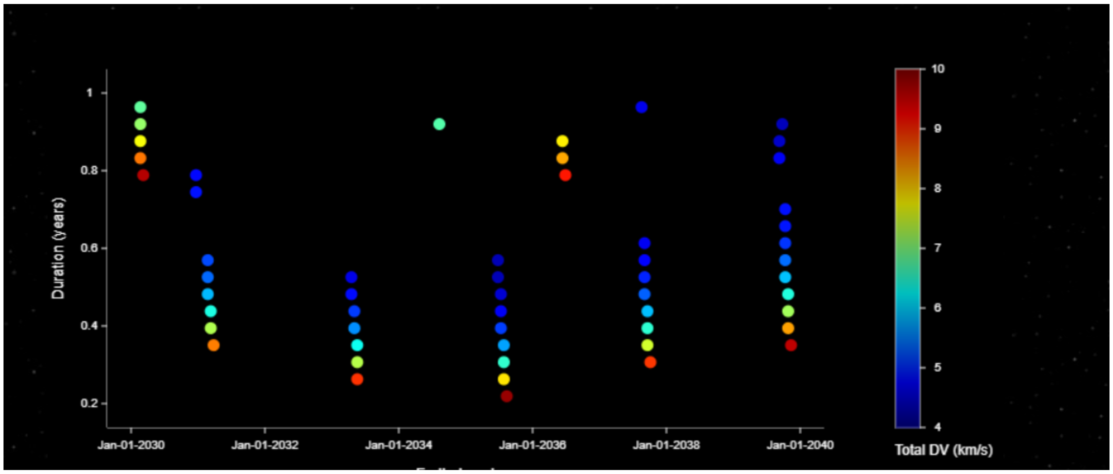

Figure 1: *Earth-Mars Trajectories from 2030–2040 (Foster)*

This results in the following pork chop plot displaying possible delta-vs given the possible launch and arrival dates:

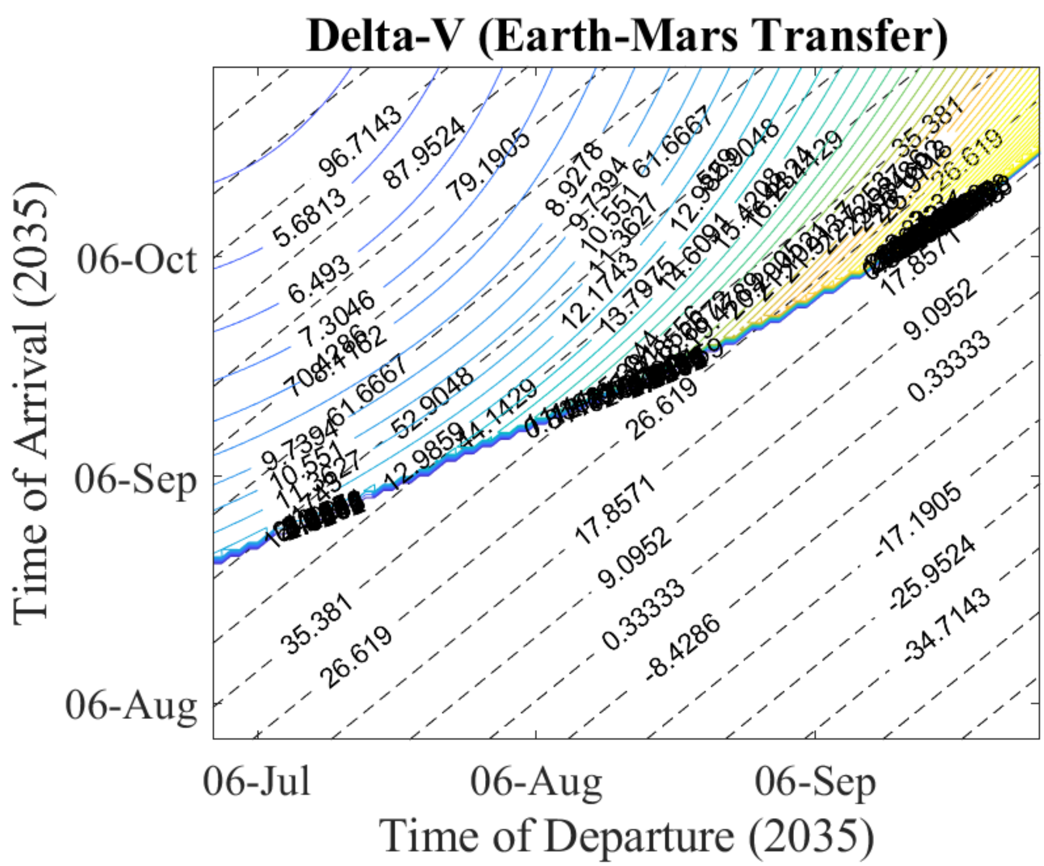

Figure 2: *Minimum Delta-V Trajectories from 2035-01-01 to 2035-11-01*

The minimum delta-v requirement for our transfer was then determined by finding the minimum delta-v plotted which intersects a mission duration of 30 days (plotted as dotted lines), which resulted in ~21.111 km/s. The empty space shown in the bottom right corner is trajectories whose duration would be a negative time—due to how the code calculates the difference in departure and arrival dates—and as such they have been omitted from the final plot. An absolute minimum delta-v can also be seen being approached in the top left corner of the diagram, however this minimum region only spans mission durations of 120–160 days, which is far longer than that of our 1 month requirement; while this is more in line with what current spaceflight technology achieves in terms of delta-v and mission duration (hence being conveniently at the total minimum), our design instead seeks to minimize time down to 1 month by exploiting the incredible delta-v magnitude that the nuclear drive is already capable of producing. Thus, the chosen delta-v is not the absolute minimum, but rather the minimum for our abnormally shorter mission duration, sacrificing extra delta-v from the engines for a much shorter mission time and striking an efficient balance between the two. In reality, the plot demonstrates that our mission duration could potentially be brought down to 20 days (as seen in the top-right asymptote), but the associated delta-vs begin to approach values which are too high even for the nuclear drive to produce, so our 1 month duration was kept.

In doing this calculation we are making a significant simplification by assuming that the initial position will be at the center of mass of Earth and the initial velocity of the MAPPP vehicle will be that of Earth as well, instead of using the position and velocity of the vehicle in its HEO phase. However, the differences in delta-v will be small in comparison to that of the entire transfer, as the sun is already orders of magnitude farther away and the heliocentric velocity of Earth is very close to that of the proposed HEO velocity, so it will suffice for initial calculations.

XII. DELTA-V BUDGET

The vehicle will have a total delta-v budget of around 50km/s as a result of the extremely high efficiency of its nuclear pulse propulsion engine.

A. Earth LEO to Earth Eccentric Orbit

From the Low Earth Orbit to the High Earth Orbit overall, there is a delta-v required to take the Medusa vehicle from the LEO parking assembly orbit at 500km into a HEO at 100,000km at the start of the mission. Assuming that both the initial and final target orbits are circular, a total delta-v of 4.0824 km/s is needed for the transfer maneuver.

The transfer method used will be a standard Hohmann transfer, even though it is not the most delta-v efficient method, it is the most direct and time-efficient transfer. The first burn of the Hohmann transfer has a delta-v of 2.8213 km/s for the chemical propulsion system out of LEO.

The ratio of the final semi-major axis to the initial semi-major axis is in the range where the bi-elliptic orbital transfer would be slightly more delta-v efficient based on a plot of the orbital methods versus the ratio of the semi-major axes (Savransky); however, since the mission demands a quick transfer, a Hohmann orbit is the appropriate choice.

B. Earth Eccentric Orbit to Earth HEO

The second burn of the Hohmann transfer will be done using nuclear pulse propulsion at apoapsis—far enough away from Earth for safe operation—to turn the 500x100000km orbit into 100000x100000km circular orbit. The second burn of the Hohmann transfer has a delta-v of 1.2611 km/s to get the Medusa vehicle into this circular orbit, and is considerably less important due to the efficiency of the nuclear drive. This maneuver, as well as the previous maneuver with which it is coupled, is modeled as an orbit around a geocentric central body.

C. Earth HEO to LMO

Transfer of the Medusa from the 100000x100000km HEO orbit to a low Mars orbit, ignoring the sphere of influence of Earth due to the HEO already being far enough away to make it fairly negligible for our 1st order approximations. Checking the smallest possible delta-v for our porkchop plot at a mission duration of ~30 days was a delta-v of 21.111 km/s on an elliptic transfer orbit.

D. Earth Transfer to Eccentric Earth Orbit

Transfer of the Medusa vehicle from low Mars orbit to a 500x100000km eccentric orbit around Earth. The delta-v for this maneuver is approximately equal to the delta-v required to transfer from Earth to Mars, therefore having a delta-v of roughly 21 km/s.

E. Aerobraking into LEO

In order to enter LEO the vehicle needs to do effectively the same maneuver as it did to enter an elliptical orbit at the beginning of the mission. However, instead of using chemical propulsion, it would use aerobraking to get about 2.8km/s of delta-v, applied at periapsis, for no vehicle propellant cost. Since the vehicle has a large sail and is designed to survive close exposure to nuclear detonations, it is extremely well suited to aerobraking due to its large resistance to thermal damage and small ballistic coefficient. The aerobraking would take place over a large number of orbits, and once the apogee is at 500km the vehicle would use its attitude control thrusters to raise its periapsis out of the atmosphere.

F. Satellites

For the satellites, no delta-v is required to put them in their respective orbits by the Medusa vehicle. We can deposit all the satellites at once when the Medusa vehicle is in a 55-degree, circular, parking orbit around Mars. The vehicle will deposit the satellites at a lower semi-major axis than the operating SMA. Then, the satellites' onboard thrusters will raise themselves to the operating altitude at different rates. This is because the nodal precession rate is inversely proportional to the semi-major axis, so different satellites at different altitudes precess at different rates. When each group of four satellites has the right longitude of ascending node separation from each other, no further thrust is necessary for the ascending node changes. To get to the right spacing in terms of the time of periapsis, circular phasing can be used (altering SMA to alter orbital velocity). These phasing burns will occur over the span of one orbit, so perturbation effects will not have much time to act on the satellites at different altitudes, therefore they are ignored.

Since all of the satellites are at the same altitude and inclination, they will all be affected by nodal regression at the same rate, so their longitude of ascending node spacing will be preserved. On Mars, the J3 term is not insignificant. Since this is an odd harmonic, it will cause secular inclination changes. Some delta-v will also be needed to maintain each satellite in their respective inclinations as well as guard against the extremely small effects of atmospheric drag.

G. Total Delta-V Budget Table

Table 6: *Total Mission Delta-V Budget*

Mission Stage	**Delta-V Required** (km/s)
Earth LEO to Earth Eccentric Orbit	2.8213
Earth Eccentric to Earth HEO Orbit	1.2611
Earth HEO to Mars Transfer	21.111
Earth Transfer to Eccentric Earth Orbit	~21
Aerobraking into LEO	2.8
Satellites	0
Total	48.9934

Given the breakdown of the mission delta-v table, the mission requires a total of 48.9934km/s delta-v for the listed mission stages. This is given the different calculations and assumptions made on varying stages of the overall mission, as listed above.

XIII. GRAVITY ASSIST

While a gravity assist around the Moon for the transfer trajectory was considered, ultimately the delta-v gained from doing such a maneuver is insignificant compared to the nuclear pulse propulsion that will already be driving the MAPPP vehicle before even reaching the Moon's sphere of influence. Additionally, because the Medusa propulsion system is so much more efficient than standard chemical propulsion, the optimizing variable for our mission is primarily the duration, thus a gravity assist would only take time away from the mission, saving delta-v that is not strictly necessary for our system.

XIV. EARTH LAUNCH

The Medusa vehicle has a total mass of about 5,000t as shown in Appendix A, and will require an additional 6,000t of chemical propellant to boost it into a highly elliptical orbit. This large amount of mass will be lifted into orbit using approximately 120 launches of the in-development Starship Launch Vehicle, which can lift more than 100t into LEO per flight (SpaceX). The starship vehicle is chosen as it is the only vehicle in development as of the writing of this paper that can lift large payloads into orbit quickly and cheaply. A multiple launch strategy is clearly needed as no existing or in-development launch vehicle can lift anywhere near 5,000t into orbit in a single launch. While this is a large number of launches, it is reasonable because Starship is designed to be rapidly and fully reusable, meaning that the only hardware that needs to be replaced from flight to flight is the payload itself. Additionally, most of those flights will only carry propellant used to refuel the chemical booster, so there is no payload to be integrated on these flights. Assuming a conservative launch rate of one flight per week per launch pad and three launch pads, the entire vehicle could be assembled in orbit in less than a year using a fleet of starships. The initial burn out of LEO into an elliptical orbit will be done by a Super Heavy booster, which will SSTO itself into orbit with no payload and then be refueled twice. The booster itself cannot carry enough propellant to boost Medusa into an elliptical orbit in a single burn, so the burn will be split up into 2 parts, with the booster being refueled before each burn by several dozen starship launches.

APPENDIX A

The problem of estimating the mass of the Medusa spacecraft is quite difficult since the highly discrete and impulsive loads produced by a sequence of nuclear detonations is quite different from the continuous forces produced by conventional rocket engines. The first step in designing the vehicle is selecting a nuclear device to make the vehicle around. The mass of the sail of the Medusa spacecraft scales with the amount of energy released on the detonation on the nuclear device, so for our vehicle we chose the smallest nuclear bombs possible, specifically the W54 nuclear device which has a yield of 1kt and a mass of 20kg (Pike). Additionally, a sail radius (R) of 1000m was selected. Smaller sail radii are more mass efficient but are more prone to damage from the nuclear detonation since the incident energy is more concentrated. The optimal radius of the sail is a point of future optimization.

The general mode of operation of the spacecraft is that the sail absorbs a portion of the energy of the detonation as rigid body acceleration, the mass distribution of the sail is specifically chosen so that all parts of it are accelerated at the same rate during the pressure pulse after detonation. The pressure pulse duration is very small, less than a millisecond, so the sail is rapidly accelerated during that pulse then slowly decelerated over the course of a second by being reeled back into the Medusa. The Medusa vehicle will have a mechanism to completely decouple the motion of the sail and vehicle during a detonation, preventing the impulsive force from driving loads on the main vehicle or causing crew discomfort.

The sail itself is in the shape of a hemispherical cap. This is chosen since spheres are the optimal shape to carry pressure loads and the math is relatively simple. The driving considerations on the sail are that it must undergo rigid body acceleration during the pressure impulse and that it must not fail from hoop stress during the pulse. Since it is a hemispherical shape, the center of the sail has no hoop force, but the hoop force increases as the angle between the axis of the sail and the point of interest increases as shown below:

$$Fhoop\ = Pmax * R * sin^2(angle) \qquad \text{(Eq. 1)}$$

Intuitively, this relation is squared because as angle increases both the radius the non-axial pressure acts on increases, and the component of pressure that is non-axial. In order for the sail to undergo rigid body acceleration, the mass must scale with the projected area. This requires the sail thickness to vary as shown below:

$$Sail\ Thickness\ = Pmax * cos(angle)/((density\ of\ material)(sail\ acceleration))$$
(Eq. 2)

The thickness must decrease with increasing angle since the angle of the sail material changes yet the volume per project area must remain constant. The sail acceleration term is a relatively free variable which is limited by the material properties of the sail material. The tensile stress in the sail is given by the following equation:

$$Stress = \frac{Fhoop}{Sail\,Thickness} \quad \text{(Eq. 3)}$$

Since the sail is in pure hoop tensile stress, we can select a material that has a very high strength to weight ratio in pure tension, namely pure carbon fiber strands. Specifically, we have selected Toray 1100G carbon fiber, which has a tensile strength of 7GPa and density of 1.8g/cm^3. The total mass of the sail is given by the following equation:

$$Sail\,Mass = Pmax * (sin(angle) * R)^2/(sail\,acceleration) \quad \text{(Eq. 4)}$$

However, these equations cannot be solved in isolation, as the specific impulse of the engine is also dependent on the angle which the sail covers. This is based on the ideal specific impulse (occurring when the sail covers a full 180 degrees of the explosion (90 deg half angle)), which is given by Solem in this equation:

$$Ideal\,Isp = \frac{25}{24g}\sqrt{\frac{2E}{5mb}} \quad \text{(Eq. 5)}$$

Where E is the energy of the detonation and mb is the mass of the bomb. We can correct this for sails that do not cover the full 90deg half angle below:

$$Isp = \frac{25sin^2(angle)}{24g}\sqrt{\frac{2E}{5mb}} \quad \text{(Eq. 6)}$$

Additionally, Solem gives an equation for the maximum pressure experienced during the detonation impulse:

$$Pmax = 0.98 * E/R^3 \quad \text{(Eq. 7)}$$

Finally, we have decided on a payload of 2,500t. The choice of payload was based on the smallest nuclear bombs that can be made due to critical mass limitations, the efficiency of a Medusa spacecraft drastically increases with scale since a vehicle at least an order of magnitude can start using thermonuclear bombs with fusion stages, which get much better yield to weight ratios than pure fission devices. We used a non linear solver and the rocket equation to design a single stage vehicle that can take a payload of 2,500t through a delta-v of 25km/s and then return itself through another delta-v of 25km/s, which automatically accounts for the change in mass and Isp as a function of the angle which the sail covers. A pulse charge frequency of 1Hz was arbitrarily chosen, further analysis is needed to determine the optimal rate at which bombs are detonated. The thrust of the vehicle is proportional to the rate of bomb detonation, and this rate is likely limited by the ability of the vehicle to survive the thermal effects of nuclear detonations.

Table 7: *Resulting Performance Characteristics of Medusa Vehicle*

Sail Half Angle:	34.8 degrees
Bomb detonation rate:	1/second
Sail Mass:	431t
Average Thrust:	1.96 MN
Initial Average Acceleration:	0.41m/s^2
Specific Impulse:	10,000s
Sail Mass:	431t
Capsule Mass:	500t
Payload Mass:	2500t
Outgoing Propellant Mass:	1070t
Return Propellant Mass:	270t
Total Propellant Mass:	1340t
Total Initial Vehicle Mass:	4770t

This analysis results in a vehicle with a mission start mass of 4770t, which includes 2500t of payload and 1340t of nuclear bombs. This works out to 67,000 nuclear pulse units, which is completely insane and a rather good reason why a device like this has never been constructed. This is almost as many nuclear bombs as there were during the height of the cold war. We decided to round the total vehicle mass to 5000t to have a round number and since the mass of a vehicle only grows as it matures.

Obviously, this analysis neglects a lot of important vehicle systems including the winch system, bomb storage and dispensing, crew quarters, radiation shielding, and attitude control systems to name a few. These systems are considered to be included in the 500t capsule mass. Interestingly, the tether is relatively light compared to the rest of the vehicle, only about 3t per km, meaning that allowing the vehicle to hang large distances behind the nuclear detonation are potentially feasible. This is important because the radiation and heat fluxes of a nuclear detonation are reduced by the inverse square law, so a long tether can greatly reduce the amount of shielding necessary. Additionally, substantial ability to mass optimize the sail is possible since the center of the sail is extremely strong compared to the forces it sees, so by allowing some induced stress near the center of the sail during the pressure pulse substantial weight losses could be possible.

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