Mars Opposition Piloted Nuclear Electric Propulsion (NEP)-Chem Vehicle

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Many previous studies have examined sending crews to and from Mars. The most economical involved a 'conjunction' class whereby the crew spends around 500 days on Mars surface waiting for a 'cheap' return. The total mission time results in a mission duration around 3 years. Given the current demonstrated crew maximum of a 1 year stint on ISS, it is interesting to look at reducing that time to only two years, thus reducing risk and minimizing time in the Martian System. In order to meet such a short mission an 'opposition' class Mars mission (which includes a Venus flyby) was chosen. The energy required to perform such a mission in only two years (for the 2036 opportunity at least) is about three times that of the 3 year conjunction mission. The rocket equation clearly shows that this mission would then require several times the propellant of the three-year mission unless the Isp of the propulsion system can be increased. Electric propulsion can provide the 3-10x improvement in Isp but even with a nuclear reactor power levels could approach 10 MWe. As an alternative, a smaller reactor (1.5 MWe class) joined together with a chemical stage was found to allow for using each propulsion system to its best advantage: low thrust in interplanetary space and chemical in the gravity wells of Earth and Mars. Indeed, the use of high Isp, low thrust during the interplanetary leg of the journey's reduced the required capture/departure ΔVs by 5-10X. Lowering the NEP power also allowed fitting the power system into a single SLS launch – which limited the radiator area to ~ 2500m^2. For the first look a reactor using fuels created by the SP-100 program with a limit of ~1200K was assumed. Starting in the 'Lunar Gateway' also allowed for use of commercial tankers to fuel the vehicle in a relative benign place. A top level summary of the mission design, concept of operations, as well as a conceptual point design of the vehicle is described.

I. Introduction

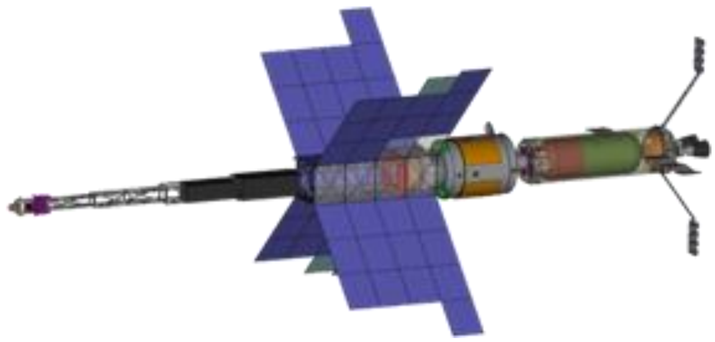

Fig. 1 Vehicle Illustration

Many previous studies have examined sending crews to and from Mars. The most economical involved a 'conjunction' class whereby the crew spends around 500 days on Mars waiting for a 'cheap' return. The total mission time results in over a 1000-day mission duration (about 3 years). Given the current crew member experience level of only one year on the International Space Station (ISS), it is appropriate to ask if Mars missions can reduce that time to only two years, thus reducing risk and minimizing required Mars surface infrastructure. An opposition mission in 2036 was deemed appropriate given the planned Moon missions in the 2020's.

Given the date of 2036, nearer term technologies – primarily nuclear thermal and nuclear electric were deemed as viable for these missions. The energy required to perform such a mission in only two years (for the 2036 opportunity at least) is about three times that of the three-year conjunction mission. The rocket equation shows that this mission would then require several times the propellant of the three-year mission unless the specific impulse (Isp) of the propulsion system can be increased.

Two Space Launch System (SLS) launchers with 8.4m fairings were assumed for the piloted transportation portion of the mission, limiting the size of the system. When using nuclear electric propulsion, the main limiting factor was packaging the required radiator area.

The higher Isp nuclear electric propulsion (NEP) system option is described herein but with a twist: in order to keep the size of radiators packageable in one SLS and keep the reactor technology level high (~1200K) the NEP system had to be combined with a chemical propulsion system. This combination of electric propulsion and high thrust chemical was found to be useful in previous design studies combining solar electric propulsion and chemical propulsion [1]. Such a combination allowed the low thrust system to provide significant change in velocity (ΔV) during the interplanetary portions of the mission, thereby notably reducing the ΔV required by the high thrust system to capture and depart from the Mars gravity well. Here the high thrust 'impulsive' system is more efficient due to the Oberth Effect [2].

The NEP-Chemical transportation vehicle design is described herein. Explanation of requirements is followed by an overall design evolution (with trades called out.). Next, mission design is described followed by the concept of operations (CONOPS). After a summary of the final configuration and the top-level spacecraft mass and power equipment lists, each subsystem is briefly summarized. Lessons Learned and Next Steps concludes this document.

II. Study Background and Assumptions

A. Assumptions and Approach

The study goal was to explore viable 2-year roundtrip class Mars mission concepts for the 2030s. A 2035 opposition opportunity was chosen as representative. Instead of starting at low Earth orbit, an assembly orbit near the lunar gateway (NRHO) was chosen to evaluate its impact. A mixed launch fleet of SLS and commercial launches was assumed, with the number of each being a figure of merit. Performance to the NRHO was assumed to be 45t and 15t for the SLS and CLVs respectively.

1. Figures of Merit (FOM)

The Compass Team used largely qualitative parameters to define figures of merit (FOM) and guide the subsystem. The FOM for this design focused more on system performance than on current technology readiness level (TRL) or cost. For this design study, the Compass Team assumed the following FOM:

- Feasibility
- Number of SLS
- Number of CLV
- Simplicity
- Risk
- Reuse

2. Redundancy

Single fault tolerance was assumed for this preliminary iteration. Future studies that focus on Loss of Mission versus Loss of Crew and abort options will further investigate required redundancies. Top-level redundancies include spare hall effect and chemical thrusters and multiple path radiator loops.

B. Growth, Contingency, and Margin Policy

The mass growth, contingency, and mass margin policy used by the Compass Team is congruent with the standards described in AIAA S–120A-2015(2019) [3]. The Compass Team typically uses a 30% growth on the bottoms-up power requirements of the bus subsystems when modeling the amount of required power. There is an exception, however, for the electric propulsion subsystem. If present, only 5% growth is applied to the power requirements needed for the electric thrusters. No additional margin is carried on top of this power growth.

C. Mission Description

1. Guiding Mission Trades

Preliminary analysis was completed comparing the performance of a NEP-Chem hybrid mission with an all-NEP mission. The reactor power level and temperature were varied and the inert mass of the deep space transport (DST) was adjusted based on the specific mass of the power system from Figure 2 below. For each of the three reactor temperatures, a design point was chosen that corresponds to the maximum radiator area that can fit in the SLS payload envelope (2500 m2). The conclusions from the reactor analysis were: 1) a 1200 K reactor could be developed on schedule with moderate risk, 2) a 1500 K reactor was marginally possible with high risk, and 3) an 1800 K reactor has too many development challenges to meet the mid 2030s launch goal. The NEP-Chem and all-NEP mission both utilized a Venus flyby and upper limit of 700 day interplanetary time-of-flight (TOF) in order to keep crewed time of the end-to-end mission to approximately two years.

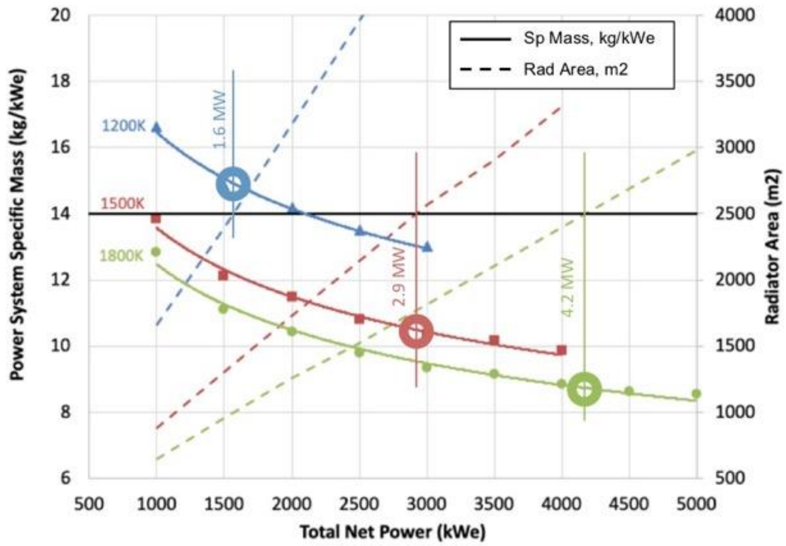

Fig. 2 Power System Specific Mass vs. Total Net Power

Fig. 3 below is a plot of Earth departure mass versus EP power level and compares the results of NEP-Chem (using LOX/LCH4 for the chemical propulsion system) and all-NEP. To achieve the same performance of the NEP-Chem system, an all-NEP system must operate at a substantially higher power level and higher reactor temperature. This results in the all-NEP system requiring a larger "technology leap" to reach the same performance than a lower power, lower temperature reactor when the NEP system is combined with a chemical propulsion system. Due to the desire to put all the radiators for this vehicle in a single SLS launch the radiator area was limited to 2500 m2. This radiator area for a 1200 K reactor corresponds to approximately 1.5 MW EP power (1.6 MW total power).

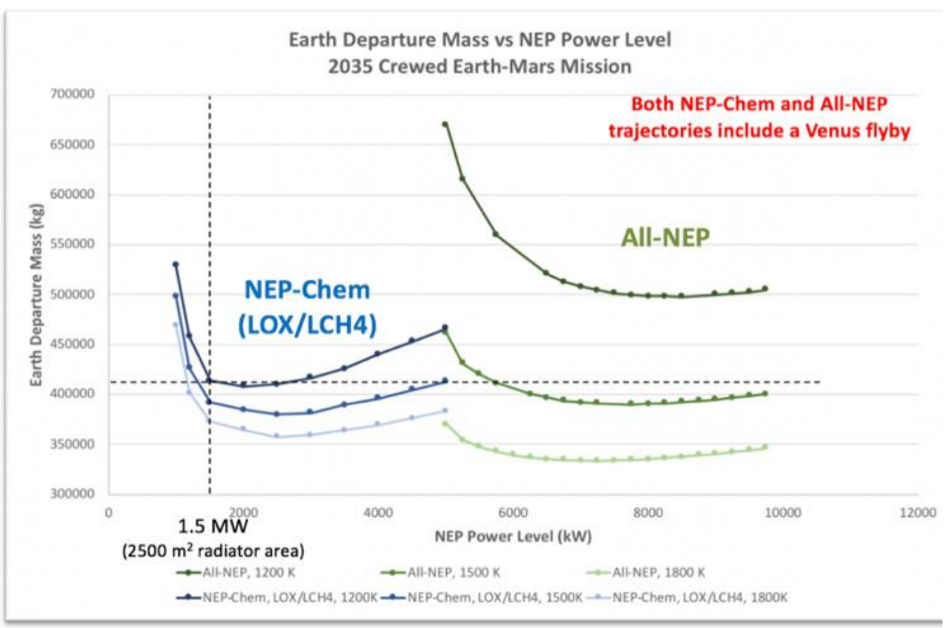

Fig. 3 Comparison of NEP-Chem to all-NEP for Crewed 2035 Earth-Mars Opposition Mission

2. Mission Analysis Event Timeline—Baseline Case 1

The NEP-Chem trajectory uses a hybrid propulsion system to perform a crewed opposition Earth-Mars roundtrip mission departing Earth in 2034 and targets a Mars landing no later than 2036. Chemical engines are used at Earth and Mars to perform the major departure and capture maneuvers in the planetary gravity wells. The NEP system is used during the interplanetary transit to provide sustained acceleration, which reduces the magnitude of the chemical maneuvers. This combination keeps transit times low by eliminating long spiral in and out maneuvers at Earth and Mars and reduces the overall propellant load by limiting chemical ΔV. Ballistic Opposition missions have significant ΔV requirements, between approximately 6 and 10 km/s depending on mission opportunity. Using only a chemical propulsion system results in a vehicle of unrealistic mass due to the propellant load to perform the necessary ΔV. By using the NEP system during interplanetary transits, the ΔV that the chemical system needs to perform is reduced to roughly a third of the ballistic requirement. Since the NEP system is highly efficient, the trade from chemical ΔV to low-thrust ΔV saves a significant amount of total propellant.

To easily compare the effects of various propulsion systems on the interplanetary trajectory, all missions had a common characteristic energy (C3) of 2 km^2/s^2 at Earth departure and arrival. For this reference trajectory, the Compass Team assumed that a series of lunar gravity assists (LGA) would be completed to depart from/return to a lunar distant high Earth orbit (LDHEO).

The following propulsion system performance assumptions were made for the chemical and electric propulsion (EP) systems:

- Chemical (modeled impulsively)
    - Liquid Oxygen (LOX)/liquid methane (LCH4): 360 s
    - LOX/liquid hydrogen (LH2): 450 s
- EP
    - Xenon: 2600 s
    - 90% Duty Cycle
    - 59.4% efficient

The mission was optimized using Copernicus. Several impulsive and low thrust burn ‘legs’ were assumed to allow Copernicus place the appropriate type and amount of propulsion (whether high or low thrust) throughout the mission. Fig. 4 lists the optimal burns for the chosen 2035 opportunity. Fig. 5 below shows the hybrid Mars capture and departure sequences. Table 1 below provides the ΔV for all NEP and chemical maneuvers between Earth departure and Earth arrival.

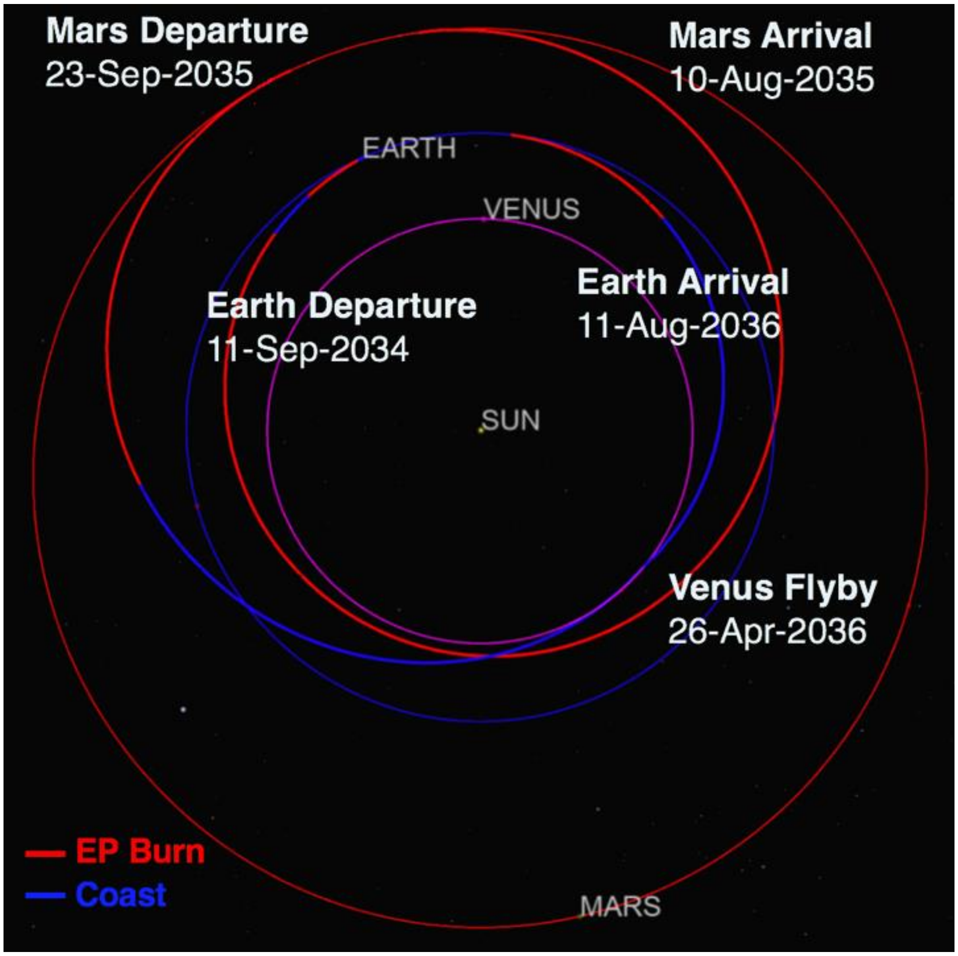

Fig. 4 2035 Earth-Mars Crewed Opposition Mission

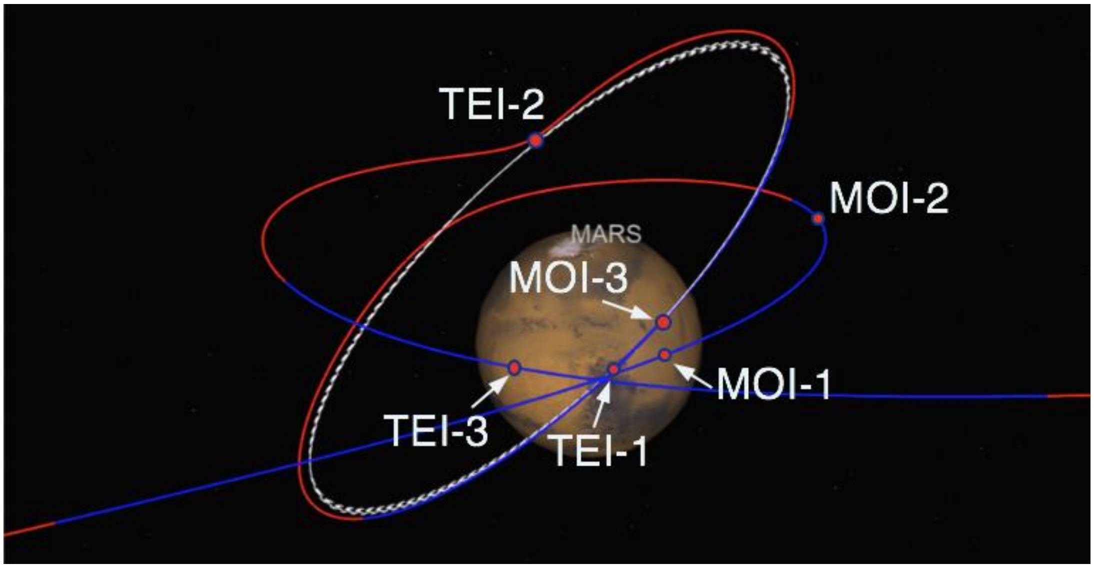

Fig. 5 Hybrid NEP-Chem Mars Capture and Departure Maneuvers

EARTH DEPARTURE	
TMI	272 m/s
EP-1	182 m/s
EP-2	1922 m/s
EP-3	2940 m/s
MOI-1	1190 m/s
EP-4	0 m/s
MOI-2	0 m/s
EP-5	141 m/s
MOI-3	19 m/s
MARS STAY	
TEI-1	29 m/s
EP-6	91 m/s
TEI-2	0 m/s
EP-7	98 m/s
TEI-3	885 m/s
EP-8	4000 m/s
EP-9	0 m/s
EP-10	1479 m/s
EOI	132 m/s
EARTH ARRIVAL	

Table 1 2035 Crewed Mars Opposition Mission ΔV History

D. Concept of Operations (CONOPS)

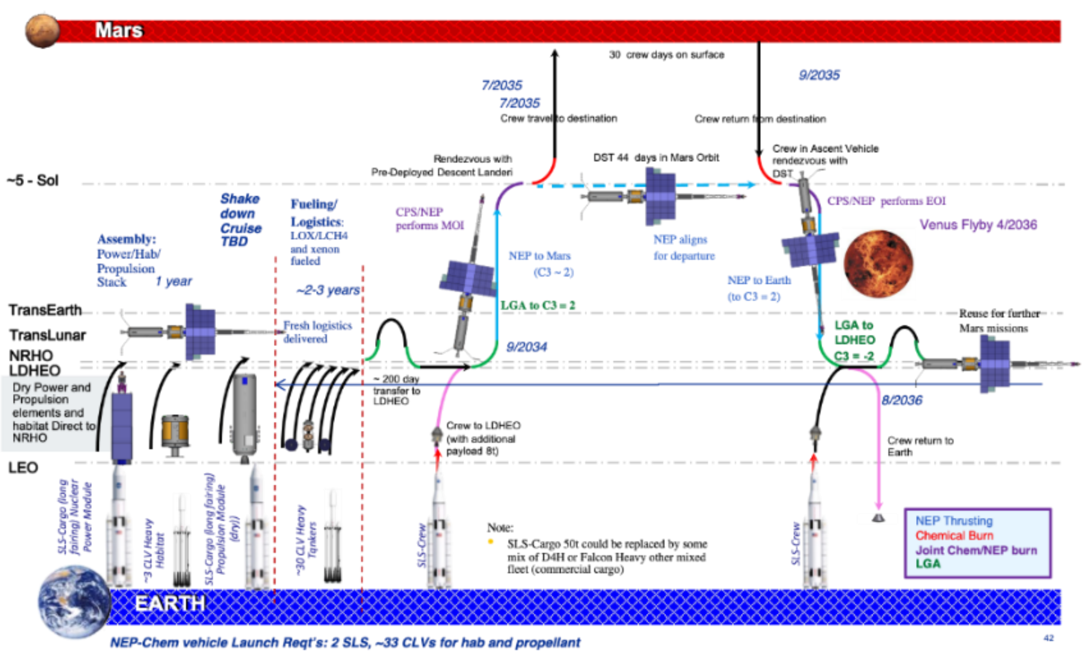

Fig. 6 Concept of Operations

Fig. 5 above shows a top-level CONOPS for the crewed mission. CONOPS phases are further defined below for each specific element (Nuclear power module, propulsion module, and habitat). Overall, the NEP-Chemical solution requires only two SLS launches but about 300t of xenon, Liquid Oxygen (LOX), and Liquid Methane (LCH4) propellants to be transferred to the vehicle by commercial launchers for every Mars mission. Depending upon CLV capability to the NRHO refueling orbit this could require fifteen to thirty commercial launchers.

The CONOPS figure illustrates that the assembly of the three elements could occur in less than a year with the fueling taking one to three years. Before complete fueling a ‘shakedown’ cruise, perhaps in cislunar space, would test the integrated system before final fueling and Mars departure. The flight to and from Mars is described in the mission section.

Commissioning Phase (~ 3 years): Many launches will need to occur over a space of three years to place all of the necessary equipment and propellants into the NRHO. Only two SLS cargo launches will be required: the Reactor power system Element and the combined Electric propulsion and Chemical propulsion Element. Each will have sufficient propulsion, power, and docking equipment to transfer to the NRHO and dock to the other elements. It is important to note that with the 45t limit of SLS performance to NRHO the elements must be launched without their main Xenon and LOX/LCH4 propellants. Thus, fueling will be provided by commercial tankers, both for the initial and follow-on missions. A notional habitat is assumed launched by CLVs. Operational empty mass of the reusable habitat is 22t with 20kW of power required and a trash dump of 15 kg / day assumed during the transit to/from Mars. Once in the NRHO the power, propulsion and habitat elements will dock.

Reactor Commissioning Phase: Once the vehicles are docked, the reactor will be started using the commissioning solar arrays for power. For startup, the control system will activate the control drums ~ 1 kW for 1 hour while simultaneously warming up the power systems. The braytons will be rotated using ~ 10 kWhr of battery energy over 4 hours. A trade still exists as to whether to start the reactor at idle before deploying the radiators or deploy the radiators first.

Outfitting Phase: Outfitting will be supplied by resupply and tanker vehicles launched by CLVs. To outfit the habitat about 3-4 CLV resupply vehicles to NRHO will be needed. For propellant tankers roughly 8-10 CLV of Xenon Tankers (@ ~ 8-10t xenon each) and 18 LOX/LCH4 tankers (@ ~8.5 t propellant each) are assumed.

Mars Mission (~ 2 years) Phase: Once assembled, outfitted and fueled the NEP/Chem vehicle will be sent to the Lunar Distance High Earth Orbit (LDHEO) using electric propulsion and a WSB transfer. Once there an SLS will launch the crew to dock with the NEP/Chem vehicle using Orion. After the unmanned Orion separates the NEP Stack leaves from Lunar Distance High Earth Orbit (LDHEO) with crew using a small chemical burn. Once in interplanetary space the vehicle uses NEP to accelerate and then decelerate to Mars (to reduce the chemical capture ΔV). The approximate time of EP thrusting during the Mars transit is ~70%, allowing long periods when the reactor can be powered down to preserve life and reduce emitted radiation. The vehicle captures chemically in a 5 SOL elliptical orbit where it meets up with the lander previously delivered by a cargo vehicle. The crew descend to Mars surface for a 30 day stay and then return to the NEP/Chem Vehicle using the MAV. After the MAV separates the NEP/Chem vehicle performs a chemical burn to escape Mars and once in interplanetary space uses NEP to return to Earth, also utilizing a Venus flyby. Once recaptured into the LDHEO an unmanned Orion is launched to retrieve the crew. The NEP/Chem vehicle then returns to NRHO using NEP and a WSB transfer for refit and refueling for the next mission.

III. Baseline Design

A. System-Level Summary

Fig. 7 below shows a top-level schematic block diagram of the subsystems of the *Piloted Mars Opposition NEP-Chem Vehicle* concept.

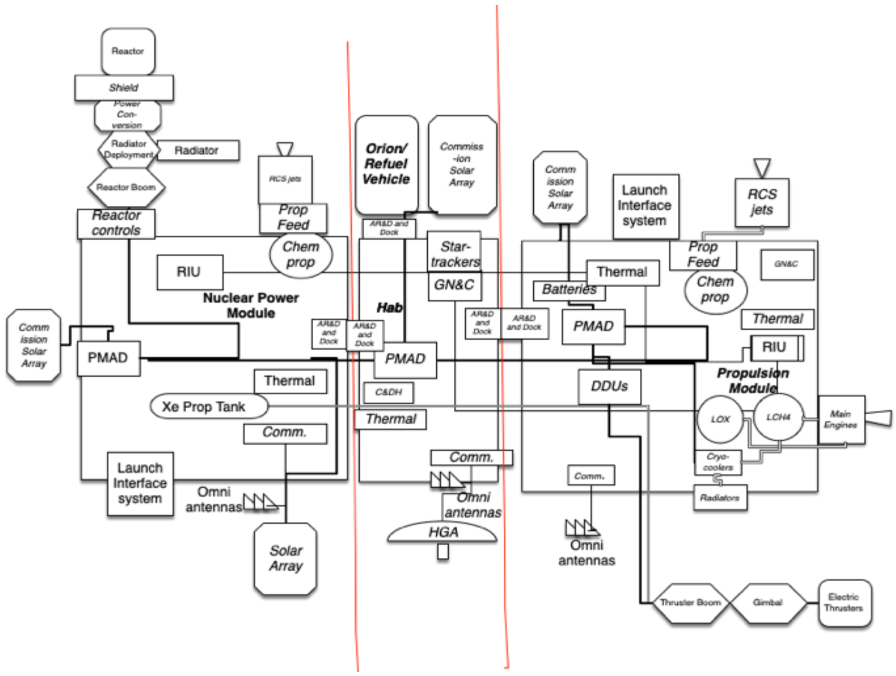

Fig. 7 Schematic diagram of the Mars Piloted NEP-Chem Vehicle with the Nuclear Power, Propulsion and Habitat Modules

B. Top-Level Design Details

1. Master Equipment List (MEL)

Table 2 and Table 3 below provide the Mars Piloted NEP-Chem MEL for the Nuclear Power and Propulsion elements. The top line sum reported for the Mars Opposition Piloted NEP-Chem Vehicle (abbreviated as Mars NEP in the MEL) includes the totals for both elements in each table. The master MEL reported below is a top-level summary of all subsystem masses. Again it is important to note that these are the launched masses and thus do NOT include all the Xenon and LOX/LCH4 propellants needed for the mission.

Table 2 Nuclear Power Element Master MEL

Description Case 1 Mars NEP Opposition CD 2019-171	QTY	Unit Mass	Basic Mass	Growth	Growth	Total Mass
		(kg)	(kg)	(%)	(kg)	(kg)
Mars NEP			102645	10%	10345	112990
Nuclear Power Module			34597	18%	6132	40729
Science			0.0	0	0.0	0.0
Attitude Determination and Control			57.6	15%	8.5	66.1
Command & Data Handling			39.0	30%	11.7	50.7
Communications and Tracking			14.4	10%	1.5	15.9
Electrical Power Subsystem			22423.7	19%	4358.2	26781.9
Thermal Control (Non-Propellant)			52.3	18%	9.4	61.7
Propulsion (Chemical Hardware)			860.2	6%	55.1	915.3
Propellant (Chemical)			3055.3	0%	0.0	3055.3
Propulsion (EP Hardware)			2375.6	15%	356.3	2731.9
Propellant (EP)			72.0	0%	0.0	72.0
Structures and Mechanisms			5647.0	24%	1331.0	6978.0

Table 3 Propulsion Element Master MEL

Description	QTY	Unit Mass	Basic Mass	Growth	Growth	Total Mass
Case 1 Mars NEP Opposition CD 2019-171						
		(kg)	(kg)	(%)	(kg)	(kg)
Mars NEP			102645	10%	10345	112990
Propulsion Module			26692	15%	4132	30824
Science			0.0	0	0.0	0.0
Attitude Determination and Control			73.3	18%	13.2	86.5
Command & Data Handling			44.3	30%	13.3	57.6
Communications and Tracking			14.4	10%	1.5	15.9
Electrical Power Subsystem			72.9	36%	26.1	99.0
Thermal Control (Non-Propellant)			2250.8	18%	405.2	2656.0
Propulsion (Chemical Hardware)			7490.8	12%	892.6	8383.4
Propellant (Chemical)			4481.6	0%	0.0	4481.6
Propulsion (EP Hardware)			8201.6	22%	1769.8	9971.5
Propellant (EP)			23.0	0%	0.0	23.0
Structures and Mechanisms			4039.7	25%	1009.9	5049.6

2. Spacecraft Total Mass Summary

The MEL in Table 4 below captures the bottoms-up current best estimate (CBE) and growth percentage of the Mars NEP-Chem S/C elements that were calculated for each line subsystem by individual subsystem team leads. In order to meet the total required mass growth of 30%, an allocation is necessary for growth on basic dry mass at the system-level, in addition to the growth calculated on each individual subsystem. This additional system-level mass is counted as part of the inert mass to be flown along the required trajectory. Therefore, the additional system-level growth mass impacts the total propellant required for the mission design. Total masses in the MEL below only include masses launched for assembly at the NRHO and do not include additional propellant required for the full mission. As such, wet masses, with growth in the chart below do not exceed the assumed 45t payload limit for the SLS 2 LV to TLI. It should be noted that growth is not reported on the habitat as its mass was assumed for the study.

Table 4 summarizes the mission for the S/C fully loaded in the NRHO, with propellant required to depart the NRHO and the mass of the S/C upon its return from Mars (includes a 250 kg Mars sample). The S/C mass reported below for departure to Mars is simply the S/C mass fully loaded less propellant consumed in the NRHO before departure.

Table 4 Summary of System Level Mass by Design Element

Mars NEP Opposition	Nuclear Power Module	Propulsion Module	Habitat	TOTAL
Main Subsystems	Basic Mass (kg)	Basic Mass (kg)	Basic Mass (kg)	Total Basic Mass(kg)
Science	0	0	40915	40915
Attitude Determination and Control	58	73	0	131
Command & Data Handling	39	44	0	83
Communications and Tracking	14	14	0	29
Electrical Power Subsystem	22424	73	107	22603
Thermal Control (Non-Propellant)	52	2251	0	2303
Propulsion (Chemical Hardware)	860	7491	334	8685
Propellant (Chemical)	3055	4482	0	7537
Propulsion (EP Hardware)	2376	8202	0	10577
Propellant (EP)	72	23	0	95
Structures and Mechanisms	5647	4040	0	9687
Element Total	*34597*	*26692*	*41356*	*102645*
Element Dry Mass (no prop,consum)	31470	22188	41356	95013
Element Propellant	3127	4505	0	7632
Element Mass Growth Allowance (Aggregate)	6132	4132	82	10345
Additional System Level Growth (For 30% tot)	3309	2525	50	5884
Total Dry Mass w/ 30% Growth	**40911**	**28844**	**41488**	**111242**
Total Wet Mass w/ 30% Growth	**44038**	**33349**	**41488**	**118874**

wet mass < 45 t SLS payload limit

note: 30% system growth is **not applied** to habitat

TOTAL MASS SUMMARY FROM MISSION SHEET:	
Total Vehicle Mass After Adding Prop	419648
Total Mass for Mars Departure	410436
Earth Return Mass	123222

Total Mass Returned to NRHO 121508.3

A more detailed summary of the total mass of the DST departing the NRHO, after being refueled is shown in Table 5 below.

Table 5 Summary of the total mass of the DST departing the NRHO, after refueling

	NP Module	Prop Module	Habitat	Total
Dry Mass w/Growth	**40910**	**28843**	**41336**	**111089**
+ RCS Pressurant	53	53	0	105
+ Main Chem Residuals	0	2637	0	2637
+ RCS Prop Residuals	339	226	0	566
+ Xe Residuals	588	588	0	1175
+ Xenon Margin	3173	3173	0	6346
+ Main Chem Margin	0	8371	0	8371
+ RCS Margin	735	735	0	1470
Inert Mass For Mars Mission	**45798**	**44625**	**41336**	**131758**
+ Usable Xenon	52884	52884	0	105767
+ Usable Main Chem	0	167427	0	167427
+ Usable RCS	7348	7348	0	14696
Total Usable Propellant	**60231**	**227659**	**0**	**287890**
Total Wet Mass For Mars Mission	**106029**	**272284**	**41336**	**419648**

C. Concept Drawing and Description

The Piloted Mars NEP Vehicle consists of three main elements: the nuclear power module, the habitat, and the propulsion module. All three of these elements will be launched separately on their own launch vehicle and assembled in space through an automated rendezvous and docking process. The habitat was treated as a black box to which minor design changes were made as it related to the design of the full Piloted Mars NEP Vehicle. Both the nuclear power and the propulsion modules will be covered here, as they were the primary elements designed within this particular Compass Team study. Fig. 8 below shows these three main elements.

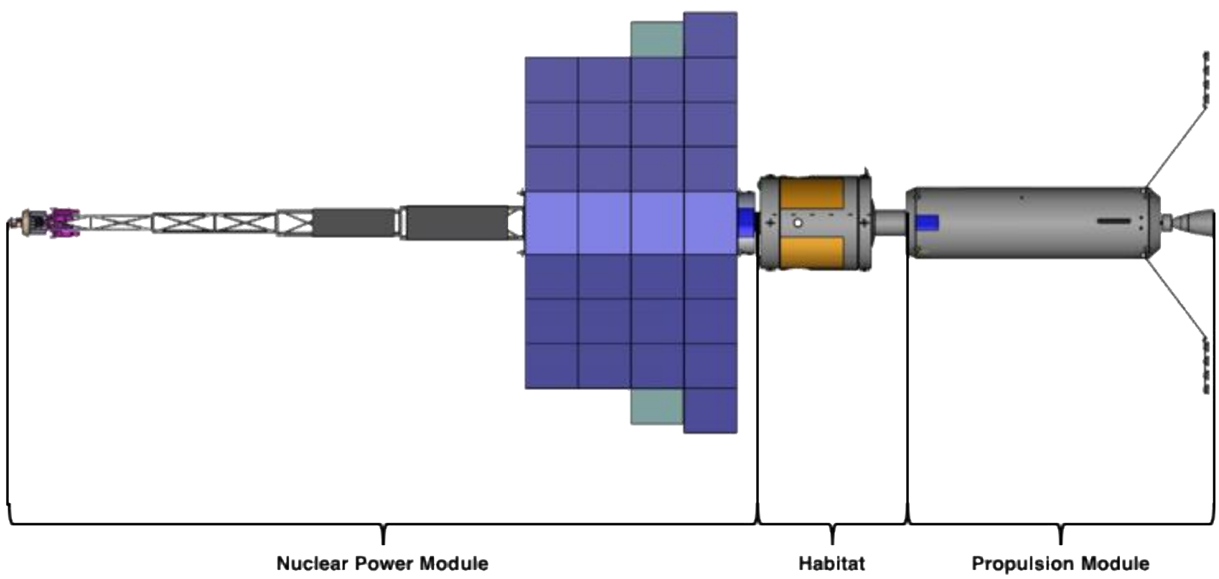

Fig. 8 The three major elements of the Piloted Mars NEP Vehicle.

Both the nuclear power and the propulsion modules are to be launched separately on an SLS utilizing the 8.4-meter diameter long payload fairing. While a payload specific Payload Attach Fitting (PAF) is required for launch, both the bus of the nuclear power module and the bottom of the bus on the propulsion module were designed to have the same diameter to allow the same PAF to be used for both launches. Fig. 9 below shows the nuclear power module stowed within this payload fairing, while Fig. 10 below shows the propulsion module inside the fairing. As mentioned previously, the maximum radiator area that could be accommodated in the SLS payload fairing was 2500 m2, which effectively determined the maximum power that could be produced by the reactor power system (1.6 MW).

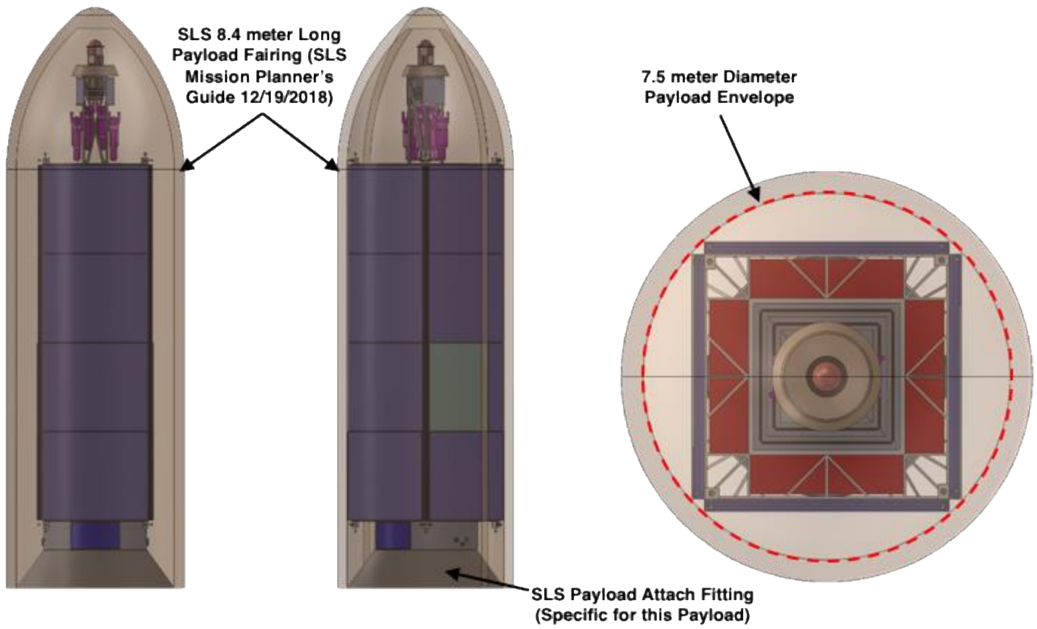

Fig. 9 Nuclear power module stowed within the SLS 8.4 meter long payload fairing.

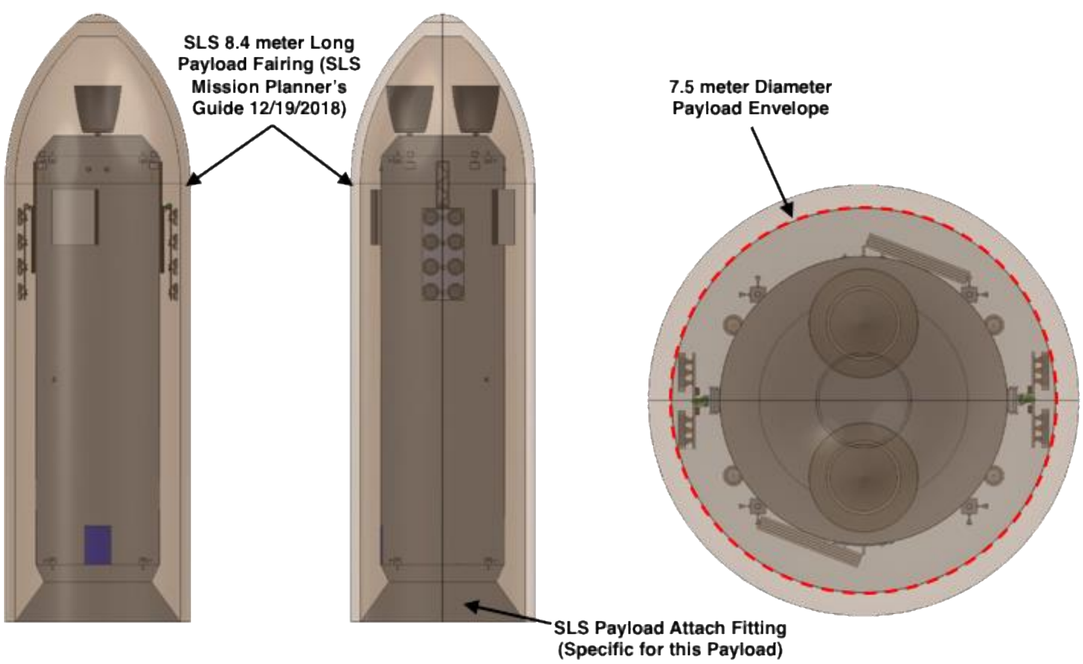

Fig. 10 Propulsion module stowed within the SLS 8.4 meter long payload fairing.

Fig. 11 below shows these radiators, along with other major components of the nuclear power module.

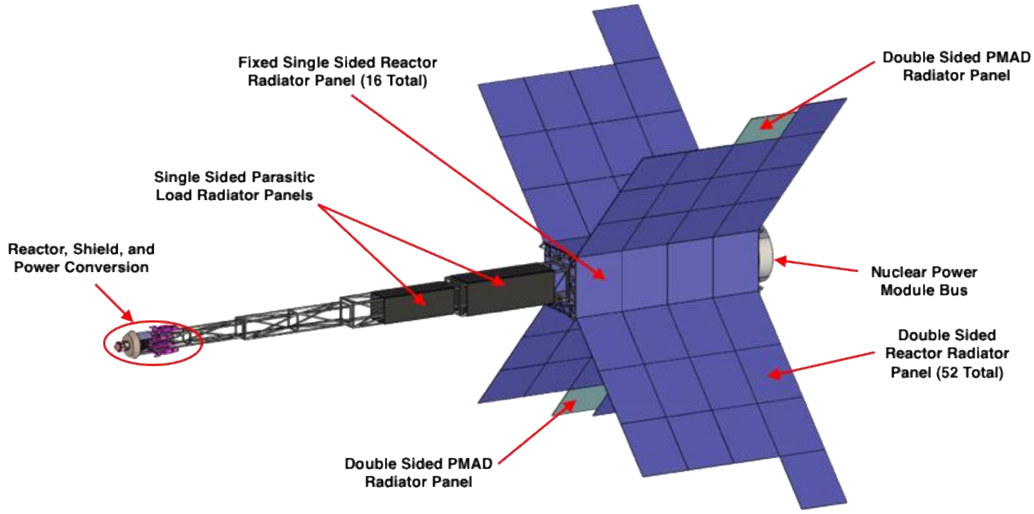

Fig. 11 Major external components of the nuclear power module

Fig. 12 below shows a transparent view of the full Piloted Mars NEP Vehicle.

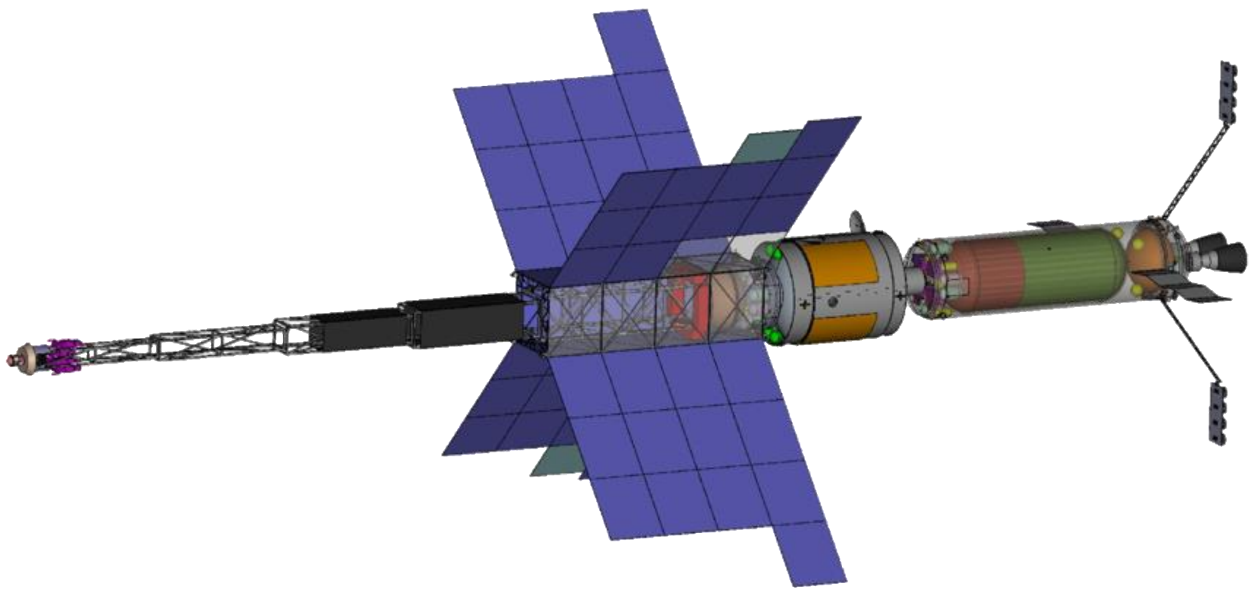

Fig. 12 Transparent view of the full Piloted Mars NEP Vehicle

IV. Subsystem Breakdown

This section provides a brief summary of the major subsystems.

A. Habitat Elements

The in-space habitation module mass was calculated and assumed to be known for the duration of the study. To obtain the mass assumed for the habitat used in the Mars Opposition Piloted Vehicle study, the 22.3t operational mass of the empty habitat, was included with approximately 18.6t of logistics mass. These assumptions resulted in a total habitat mass of 40.9t, which was tracked in the habitat subsystem MEL.

B. Communications

The Habitat and Core Modules provide the main communications system for the NEP mission (Fig. 13 Communications System Schematic below). Overview of the X-band Direct to Earth links and the S-band proximity links Communications Coverage shown in Fig. 14 diagram.

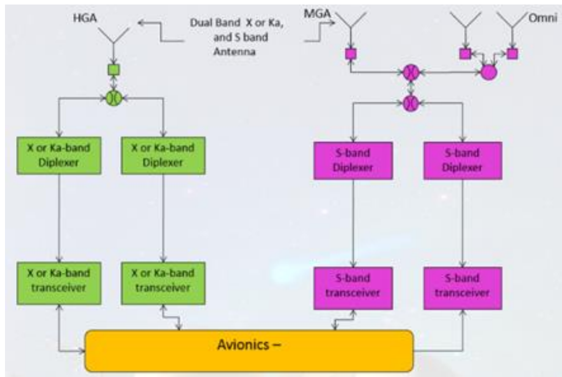

Fig. 13 Communications System Schematic.

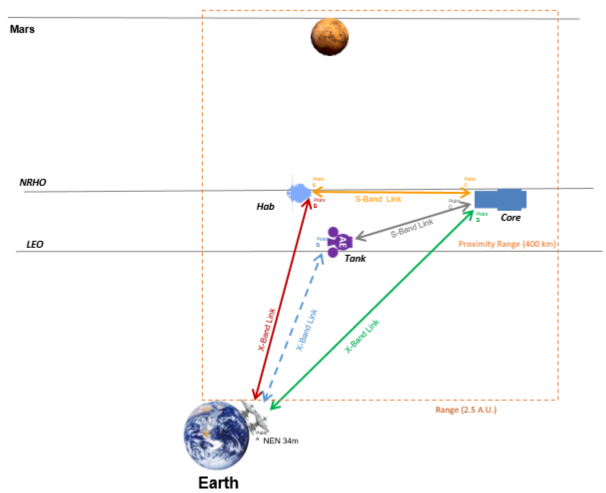

Fig. 14 Communications Coverage and System Schematic.

NEP Communications system will primarily function for NRHO operations and a backup for Mars mission. The System will have two separate double string redundant band X-band Direct-to-Earth (DTE) for Communication Tracking capability and S-band for in space Proximity communication Link.

C. Command and Data Handling (C&DH)

The C&DH avionics packages are designed around the AiTech SP0 Single Board Computer (SBC) and adhere to single-fault tolerant requirements. Each computer is responsible for the C&DH of all subsystems including most actuator controllers, and each package contains a set of standard/analog IO interface cards and motor drivers. The propulsion module includes a Gimbal Control Electronics unit to drive the gimbals for the electric propulsion engine decks. Each unit attached to the compact Peripheral Component Interconnect (cPCI) backplane adheres to the 3U avionics card size standard, and the cPCI handles all DC-to-DC power conversion required by the avionics package.

Each SP0 SBC operates with 8 GB of storage, which will contain the Real Time Operating System (RTOS), C&DH specific flight software (not modelled) and any emergency backup storage required. Overall, mass memory storage of this system will be maintained on the HAB (not modelled). All components are radiation tolerant up to 100k TID.

D. Guidance, Navigation and Control (GN&C)

Both the Nuclear Power module and the Propulsion Module are required to insert themselves into the NRHO and as such, they have identical GNC components in terms of what sensors and effectors are typically found on free flying spacecraft. Each module contains:

- 2 IMUs
  - Three single axis rate gyros to measure vehicle body rates
  - Three single axis accelerometers to measure vehicle body accelerations
- 4 Star Trackers
  - Determine inertial attitude
  - 2 data processing units each with 2 optical heads
- 8 sun sensor assemblies
  - Coarse attitude determination
  - Knowledge of direction to sun for safe mode

E. Electrical Power System (EPS)

The Electrical Power System (EPS) is designed to provide electrical power to the loads on the spacecraft from launch through operation of the spacecraft. To encompass the various operational modes of the spacecraft and the associated load demand in each phase, eight (8) operational power modes were defined for the spacecraft throughout the mission. Table 6 provides a summary of the various power modes and the associated durations.

Table 6 NEP-Chem Mission Power Modes

Power Mode 1	Power Mode 2	Power Mode 3	Power Mode 4	Power Mode 5	Power Mode 6	Power Mode 7	Power Mode 8
Launch to NRHO	Assembly at NRHO	NRHO to LDHEO and Crew Embark	NEP Thrusting	Deep Space Coast	Venus Flyby	Chemical Burn in Mars Orbit	Capture in LDHEO
140 days	up to 2 years	90 days	~ 410 days	~ 220 days	~1 day	1 hour each	1 hr

Once fully assembled, the NEP-Chem spacecraft has an abundance of electrical power available due to the nuclear reactor. However, startup of the nuclear reactor is not done until the spacecraft assembly is complete after power mode 2. Therefore, power modes 1 and 2 require another source of power generation until the vehicle assembly is completed and the nuclear reactor is turned on. To accomplish this, both the nuclear and propulsion stages include commissioning solar arrays and batteries. The Compass Team assumed that the habitat stage has its own power generation capability for power modes 1 and 2. Starting in power mode 3, the nuclear reactor provides power to the vehicle, and the commissioning solar arrays and batteries are no longer used.

The primary power system consists of a nuclear reactor connected to Brayton power conversion system and its associated heat rejection. Three-phase power produced by the Brayton cycle is converted to DC and then distributed to the NEP and other loads throughout the system.

1. Nuclear Fission Reactor

Fig. 15 below shows examples of the design space for heat transfer and power conversion in nuclear fission systems. The preliminary system analysis performed for this assessment focused on a liquid-metal-cooled reactor with HeXe Brayton conversion at three different reactor outlet temperatures: 1200 K, 1500 K, and 1800 K. The Brayton system was presumed to supply 480 Vac to the EP bus that was rectified to 650 Vdc for the Hall thrusters.

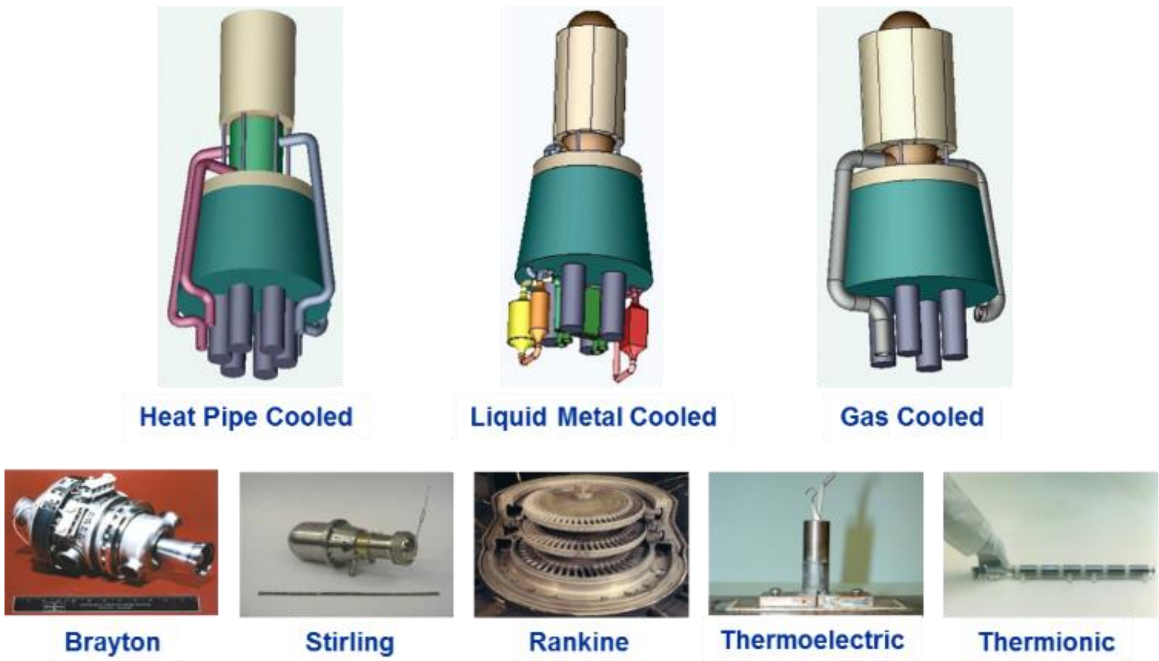

Fig. 15 Heat Transfer and Power Conversion Options

The nuclear fission reactor is the major element of the EPS. NASA has pursued nuclear fission power many times in its history due to the attractive features of the technology, namely its high power density and independence from solar irradiance. Nuclear electric propulsion systems have been studied over a wide range of power levels and use cases. The most recent concept study performed in 2012 targeted 2.5 MWe using a reactor core based on highly enriched uranium nitride (UN) fuel studied under the SP-100 program, with pumped liquid metal heat transfer to a helium xenon (HeXe) Brayton power conversion system. Since this study, significant progress in the field of space nuclear fission technology was achieved through the Kilopower project. This small reactor design with a 1-10 Kilowatt Electric (kWe) power range used a uranium molybdenum (UMo) core with passive sodium heat pipes providing the heat transfer to a Stirling cycle power conversion system. Though the simplified, lower power design of Kilopower allowed for a successful ground nuclear prototype test in 2018 that increased the agency's interest in space fission systems once again, that technology is only a small step toward the multi-MWe systems discussed in this study.

The major design considerations for sizing the power of a nuclear electric system are the reactor core design, the heat transfer method, and the power conversion system. Core design, while complex from a nuclear engineering standpoint, can be viewed simply as determining the temperature provided by the heat source. The UMo Kilopower fuel is designed to operate at 1100 K, while the 2012 NEP concept is designed for 1500 K operation. As with any thermodynamic cycle, the heat source temperature plays an important role in system efficiency and therefore specific mass. Higher temperature reactors are desirable from a mass perspective, but are less mature and more difficult to test and therefore lower on the TRL scale. The success in testing the relatively lower temperature Kilopower design over the higher temperature SP-100 and Prometheus designs demonstrates this fact. For higher power systems, however, the high temperature reactor becomes enabling to minimize the system mass and radiator area required to achieve a viable mission design.

Heat transfer plays a major role in system design and reliability. The three major primary heat transfer methods for cooling space reactors are heat pipes, pumped liquid metal, and pumped gas. Though all of these methods involve some sort of fluid motion, the mechanisms differ significantly. Heat pipes work on a passive two-phase evaporation/condensation cycle that requires no external power, while liquid metal or gas cooling requires drive pumps or compressors to actively circulate the fluid. The benefit of active cooling over passive heat pipes is flexibility in design and higher thermal throughput. Typical liquid metals used in pumped loops are lithium, sodium, potassium, or a mixture of sodium and potassium (NaK). Gas-cooled systems have the option of directly coupling to a Brayton engine, increasing the efficiency of the heat transfer subsystem. However, this leads to a single shared gas space for the reactor and power conversion system, which may have impacts on the system reliability.

The typical options for power conversion include the Stirling, Brayton, and Rankine thermodynamic cycles, as well as thermoelectric and thermionic devices. Each option presents different characteristics on thermal efficiency and power throughput, and therefore on the system mass. On the low end of the efficiency scale, thermoelectrics have a long history of use in radioisotope power systems. However, the lower efficiency is a challenge for high power fission systems due to the larger reactor, radiation shield, and waste heat radiator. The Stirling cycle has high efficiency but does not scale well to higher power, while HeXe Brayton systems fair better at higher power but the lower heat rejection temperature results in a larger radiator. A supercritical CO_2 Brayton system may perform better than the HeXe system but work on that technology has been principally focused on terrestrial and marine power applications rather than space. A potassium Rankine cycle has the potential for high efficiency and heat rejection temperature, but the two-phase system design is a challenge and the TRL is low.

The heat rejection subsystem is the largest and most visible element of the NEP vehicle. The vacuum of space requires radiative heat rejection, which is dependent on bulky radiators. In fact, the limiting design factor for the fission system in this study was the stowed radiator volume that could be accommodated in a single launch vehicle. Preliminary radiator stowage concepts have indicated a maximum radiator area of approximately 2500 m^2 for the 8.4 m SLS fairing. The benefits of the higher temperature reactor designs discussed above are increased system efficiency and higher heat rejection temperature, which both act to increase the power output for a given radiator area.

Another challenge for fission power systems is the need to shield the mixed neutron and gamma radiation field produced by the fission reactor. The amount of radiation is directly correlated to the thermal power and operating life of the reactor, which adds an additional motivation to increase the thermal efficiency of the nuclear system. The need for shielding is driven by both electronic and materials tolerance as well as human dose limits for crewed missions. Shielding design is relatively straightforward with little change between the various fission power systems studies performed in the past. Low-atomic-number materials like hydrogen, beryllium, lithium, boron, etc. provide efficient shielding for the neutron flux, while high-atomic-number materials like tungsten or depleted uranium provide efficient shielding for the gamma dose. For this study, the shielding components chosen were lithium hydride for neutron shielding and tungsten for gamma shielding.

The PMAD subsystem includes the power transmission cabling, power electronics, and system controls required for the reliable operation of the reactor power system. A key design decision is the transmission voltage, since that voltage determines the mass of the power cabling connecting the Brayton converters to the spacecraft bus and the EP thrusters. Higher voltage is preferred, but that introduces challenges on electronics availability and potential concerns for corona arcing. This study assumed a channelized architecture with separate cables and power conditioning and controls for each Brayton engine. The Brayton alternators provide 3-phase AC power at 480 Vac and 1.2 kHz that is supplied to a rectifier/filter conversion stage that delivers 650 Vdc to the EP thrusters. A full-power pulse-width modulated parasitic load radiator is connected to each Brayton converter to provide speed control and dissipate excess electric power not required by the NEP vehicle.

Taking into account the considerations discussed above and the history of nuclear electric propulsion designs, this study produced a detailed design for the 1200 K reactor concept. The 1200 K Li-cooled reactor produces1.6 MWe, utilizing a lower-temperature version of the SP-100 reactor and superalloy Brayton power conversion. Table 7 below shows aspects of the design.

Table 7 Reactor Design Trade Study

Parameter	1200 K Reactor Design
Thermal Power	5.8 MWt
Electric Power	1.6 MWe
System Mass	22200 kg
Specific Power	68.5 We/kg
Core	Li-cooled refractory UN
Power Conversion	4x 403 kWe superalloy HeXe Brayton convertors
Heat Rejection Loop	4x SS318 NaK cooling loops, 385 - 550 K
Radiator	550 K, H2O heat pipes, 2500 m2
TRL	3 - 4

The commissioning solar arrays on the propulsion and nuclear stages provide power after launch through spacecraft assembly in LDHEO. The commissioning solar arrays were designed using state-of-the-art triple-junction (TJ) solar cells with a beginning-of-life (BOL) efficiency of 30%. The commissioning energy storage provides power during eclipses and solar array off-pointed scenarios. Energy storage is provided on both the nuclear and propulsion stages for power modes 1 and 2. Both the nuclear and propulsion stages use a lithium-ion battery. Power Management and Distribution (PMAD)

2. Propulsion System Design

The spacecraft propulsion system consists of three distinct systems located on the two spacecraft modules, bipropellant RCS, an EP system, and a high thrust cryogenic system. Both modules have independent bipropellant RCS, with the two systems being very similar, and are designed to operate in an integrated fashion once the power and propulsion modules are joined. The Hall thruster EP system is primarily located on the propulsion module; with a single xenon tank and high pressure feed system located on the power module. Finally, there is a cryogenic LO2/LCH4 based high thrust system located exclusively on the propulsion module.

Reaction Control System

The RCS consists of 48 bipropellant thrusters, with 24 thrusters located on each module mounted in eight pods containing three thrusters each. These thrusters develop a nominal 220 N (50 lb_f) thrust at 295 s of Isp. The propulsion module RCS also has four additional larger aft facing R-4D bipropellant thrusters. These four thrusters develop 490 N (110 lb_f) thrust at 315 s Isp, however 295 s was used in the mission analysis for both conservativism and simplification of mission calculations. Both of these thrusters have heritage via Orion Service Module and a nominal oxidizer to fuel (O/F) ratio of 1.65 [4] [5]

Electric Propulsion System

The EP system consists of sixteen 100 KWe class Hall thrusters utilizing xenon gas as a propellant which is stored in two high-pressure tanks. The thruster system is a 15+1 design utilizing direct drive power processing with no cross strapping among thruster strings. The thrusters are mounted to two gimballed pallets with eight thrusters each. These pallets are located at the end of two booms attached to the propulsion module. The booms provide separation of the EP thrusters from the S/C to minimize Hall thruster plume impingement [6].

The Hall thrusters are a magnetically shielded design with a center mounted cathode. They provide a nominal 2,600 s of Isp at 600 V with xenon propellant, and have a nominal thruster string efficiency of 59.4%. Each thruster string consists of a single thruster, an integrated DDU/DCIU, and a single low-pressure xenon feed system [7].

Xenon for the EP system is stored supercritically in two identical doorknob shaped COPV tanks. These tanks have a titanium alloy liner with a T-1000 composite overwrap, and have a MOP of 124.1 bar (1,800 psi). There is one tank each located on both the power and propulsion module. Each tank is attached to a high-pressure flow controller via a nominal single fault tolerant feed system containing normally closed pyrotechnic valves for isolation during launch.

The high pressure sides of the feed system connect the two tanks when the S/C is assembled, and this interconnect is where the refueling interface is located. The low-pressure sides of the feed system are also interconnected, providing redundancy to the high-pressure flow regulator assemblies as high-pressure propellant from both tanks can be routed to the regulation system on either S/C module. The regulated xenon flow is then sent down parallel redundant lines on each boom to the thruster pallets, where each thruster string has its own low-pressure flow controller.

Cryogenic Propulsion System

The high thrust cryogenic propulsion system is primarily used for large ΔV maneuvers, such as departure and orbit capture. This design uses a LO2/LCH4 based system with two large expander cycle engines based on the RL-10C-2. Since LO2 and LCH4 are liquids over very similar temperature range, the propellants are stored in a common bulkhead style tank. This tank is made primarily from 2090-T83 aluminum alloy, has a MOP of 3.45 bar (50 psi), and is supported via low thermal conductivity struts. The tank is launched empty, except for inert nitrogen gas pressurant, thus the struts do not need to withstand the launch loads associated with a full propellant tank. The tank has cryocoolers, broad area cooling (BAC), TVS, internal mixers, and MLI for thermal control and propellant conditioning. There are internal slosh baffles, PMDs, and propellant level gauging hardware as well.

F. Structures and Mechanisms

Structures provide the backbone of the overall spacecraft. The structures system provides the foundation on which all the other systems are supported. The design allows the system to stand up to the launch and operational loads while providing sufficient stiffness to avoid interference with other system operations. Mechanisms are used to setup the various systems into an operational condition, to separate from hardware that is no longer necessary for the mission, and to deploy other hardware to initiate parts of other systems. The main challenge for the design was the extendable boom, both in terms of structure and mechanisms. In a nested configuration the nested boom proved capable of carrying the reactor through the launch loads although further work is needed to stiffen the stowed boom to increase its primary frequency > 8 hz. In the deployed configuration the main load impact was the chemical thruster operation which imparted only a 1/20th of g acceleration. The boom was designed to handle this load.

Fig. 16 below illustrates the overall spacecraft with deployed power system boom, deployed radiators and thruster booms.

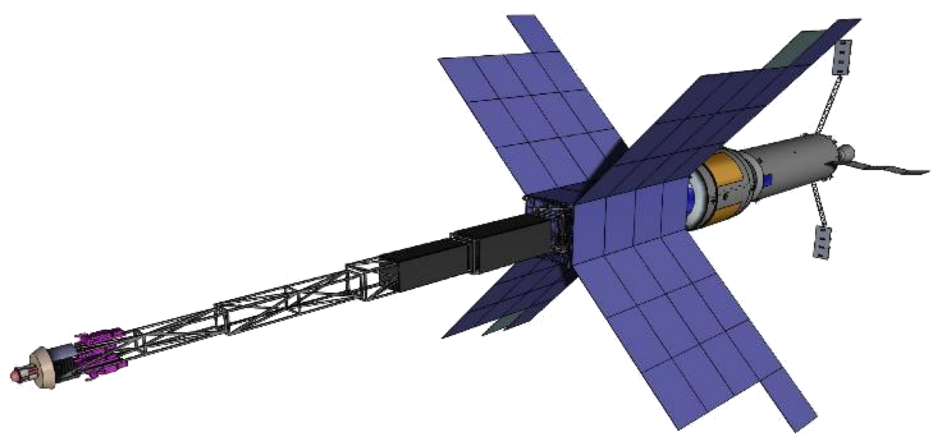

Fig. 16 Spacecraft stack with the nuclear power module, habitat, and the propulsion module.

G. Thermal Control

The Mars NEP spacecraft utilizes a combination of a nuclear reactor powered electric propulsion system and a chemical stage to transit to Mars from Earth orbit. The thermal system has to maintain the temperature of the liquid oxygen and methane tanks below the boiling point to eliminate any boiloff of the propellant in order to ensure that the propellant can last throughout the mission. This is accomplished by using a zero-boil off system consisting of a number of cryocoolers, multi-layer insulation, and the associated system for rejecting the waste heat from the cryocoolers. The thermal system also has to reject the waste heat from the electric propulsion direct drive power units as well as the waste heat from the reactor that is used to produce the electrical power for the spacecraft and propulsion system.

The thermal system for the mission involved a number of aspects of the spacecraft and their operation. The thermal system consisted of three main categories. The power system thermal control, the habitat thermal control, and the cryogenic propellant thermal control. Fig. 17 below shows these main components.

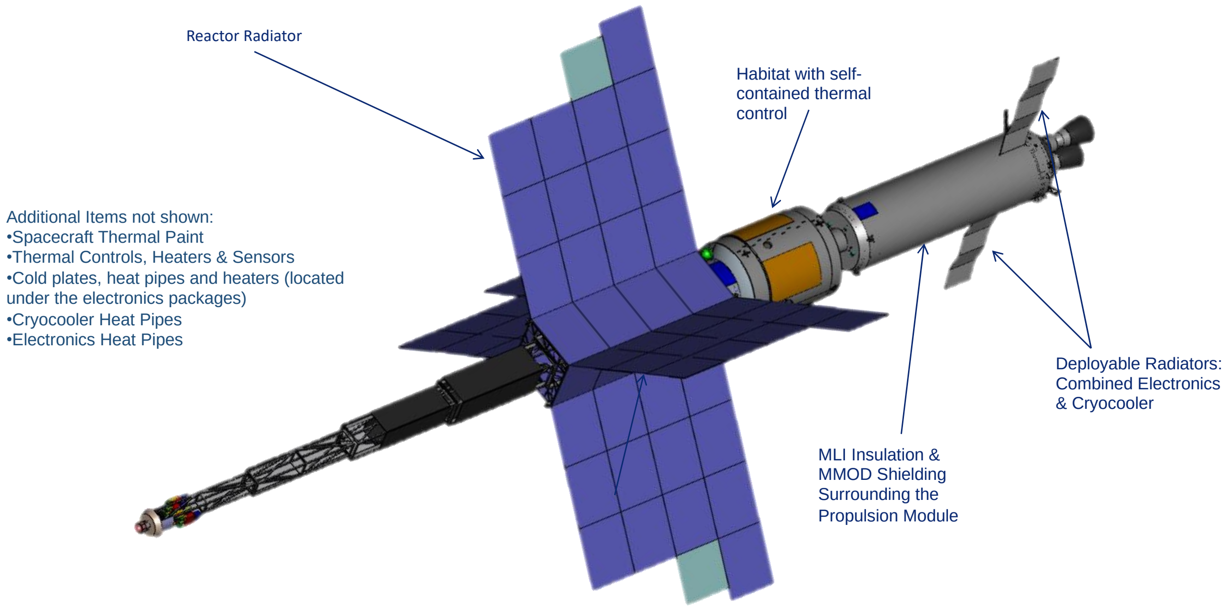

Fig. 17   Main Thermal Control Areas

V.   Risks and Technology Readiness Levels

Several risks will need to be addressed in further studies. The many deployables (Truss, Radiator, NEP thruster boom) will need to be assessed in more detail for deployment and stiffness during 0.05 g thrusting and slew events. In addition, a thorough deployment approach for both the brayton radiator coolant lines to the radiators and the high power cables to carry the >1MWe of power to the electric thrusters should be made with keeping complexity low and eliminating leaks. Ways to minimize risk by using redundancy and testing for the zero boiloff system for the chemical stage are needed. Developing a Hall thruster with throughput for multiple Mars missions should also be addressed. Finally, a more complete layout of the reactor, power conversion, radiator, and power distribution system should be performed to drive out the risks of this subsystem.

A quick assessment of the technology readiness of the primary vehicle subsystems focused on those that are <TRL 6. Not surprisingly they all focus on power and propulsion technologies.

- Power:
    - 1200K Reactor: TRL 3-4
    - Brayton power conversion: TRL 3-4
    - Large, deployable composite radiators: TRL 3-4
    - Amps Power Management and Distribution (PMAD): TRL 5

- Solar and battery technologies > TRL 6
- Propulsion:
  - Electric:
    - 100 kW Hall thrusters: TRL 4 (based on the NASA-457 100 kW test thruster)
    - Direct Drive Units: TRL 4-5
    - Xenon tanks: TRL 5
  - LOX/LCH4
    - Main engines: TRL 4-5 (but common to the Mars ascent vehicle)
    - ‘Soft’ Cryo storage: TRL 4-5 (currently under development)

VI. Lessons Learned

Developing a Mars crew transportation concept combining nuclear power, electric propulsion, and chemical propulsion led to many ‘lessons learned’ or ‘take-aways’ for future studies. They are listed here.

- As with the solar electric propulsion (SEP)-Chem concept from earlier studies, combining low and high thrust systems makes the best use of the interplanetary transfer time to minimize the chemical burn ΔVs.
- The power output and mass of the nuclear electric power system are key drivers in determining mission performance; a 1.6 MW system using relatively low-risk subsystem technologies provided sufficient performance to meet the mission goals.
- A 2600 s Hall thruster system provides nearly the optimal Isp for the mission.
- A high thrust (Isp 360 s) system was used to get the best Oberth effect in the gravity wells of Earth and Mars.
- Even higher Isps (LOX/LH2 460s or NTR 875 s) could help more as long as the mass overhead, volume, ZBO, propulsion technologies can be made available.
- Due to the high densities of the propellants (Xe, LOX, LCH4) only two SLS are needed to launch the NEP/Chem vehicle (albeit mostly unfueled).
- The largest volumetric challenge for the vehicles is the ~2500 m^2 of radiators/boom.
- Multiple commercial vehicles can be used to carry up the ~ 250t of propellants.
- Direct to NRHO launch campaign: about 30 CLV tankers over three years (~ 10 CLV per year) are needed to launch the needed propellant for each full Mars Opposition mission (less for shakedown and Conjunction missions).

VII. Acknowledgements

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