Earth based Asteroid Utilization (EAU)

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I. Mission Objectives and Background

The objective of this mission is to better understand the composition of near Earth asteroids, and how they may be used for the betterment of society. The first potential asteroid considered within the given constraints of having a heliocentric orbit with a periapsis greater than $0.7AU$ and apoapsis less than $1.2AU$, was 99942 Apophis (2004 MN4). This asteroid was found using NASA JPL's *Small-Body Database Lookup*, when searching with the previously mentioned constraints and adding the search parameters of max inclination less than 5° and a diameter greater than $0.1km$. After examining 99942 Apophis, it was found that the composition of the asteroid was LL chrondite. Expanding the search increasing the max inclination 10° provided with more potential targets of which 341843 (2008 EV_5) was chosen. Upon further researching the asteroid, it was found that its composition is similar to carbonaceous chondrite meteorites. Groups in this category of meteorite contain high concentrations of water, between (3% to 22%).

With this in mind, the planned mission of Earth based Asteroid Utilization (EAU) plans to extract the water from the asteroid sample and perform electrolysis on this water for use as propellant. This will be performed by *HYDROS™-C Water Propulsion System* for use as propellant and to better test the potential of using asteroid mining as a source of propellant. Analyzing the sample can help better understand the composition of similar asteroids and potential future uses. Furthermore, the potential of extracting water from asteroids can be a potential solution for the problem of water consumption in future extended manned space missions for both human consumption and for agricultural purposes. Targeting 2008 EV_5 was further reaffirmed as a good decision when considering that it was proposed as the target for NASA's Asteroid Redirect Mission which included a sample-return. Another proposed target for the ARM mission was the recently visited Bennu. A final benefit of choosing 2008 EV_5 is the larger size and therefore easier target.

II. Assumptions

Provided is a brief list of assumptions which will be further elaborated on in the rest of the memo.

- Atlas V gives the spacecraft $5400\frac{m}{s}$ (spacecraft is given a further $6600\frac{m}{s}$, and therefore still meets requirements).
- United Launch Alliance will help provide the launch vehicle interface as stated in section 5 of the Atlas V user manual.
- The spacecraft will be obstructed for no longer than 4 hours.
- Spacecraft framework weighs $40kg$.
- Thermal system weight assumed to be $10kg$.

III. Technical Specifications

A. Avionics

For the flight computer, argotec FERMI was chosen for its low power consumption while still high

- 256 Mbyte SDRAM
- 16 GB ECC-corrected mass memory
- Dual core
- $P = 5W$
- Radiation resistant 100 krad

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The memory operating unit on the spacecraft is a Mercury RH3480 Solid-State Data Recorder.

- $V = 4.5 - 5.5V$
- $P = 7W$ at idle and $25W$ at full performance
- $T = -40°C - +72°C$
- $m = 0.75kg$

This was chosen for the radiation tolerant design, low voltage, high range of operating and storage temperature. The high storage will be useful for storing images and videos taken with the on board camera. Images and videos of the impact crater after sample collection will be used to better understand the cohesion of asteroids. With its voltage range, it allows for the memory operating unit to comfortably operate at max performance. Lastly, its light weight is an additional benefit.

BAE Systems 3U-160 CompactPCI® will be used for the Peripheral IO. Many of the reasons behind the decision are similar to that of the memory operating unit, being radiation tolerant, low power, high operating temperature range, and low weight. The specific values for the aforementioned parameters are as follows:

- $V = 3.3V$
- $P = 4 - 5W$
- $T = -55° - +70°$
- $m = 0.5kg$.
- radiation resistance of >100 krad

Additionally the software features of C Compiler, assembler, linker, and simulator available for embedded microcontroller, allow for wide range of different and widely used programming languages.
A block diagram of the data distribution can be found in the appendix, figure 3.

B. Guidance, Navigation, and Control Systems
On board attitude sensors will consist of two Arcsec Sagitta star trackers.

- $P = 1.3W$
- Track up to 32 stars
- Update rate of $5Hz$
- $95x50x45mm$

Along with the track of 32 stars each, these star trackers discard most false star instances, has a good update rate, draws little power, and is small in size. Having two on board allows for backup and comparisons to be made, and is doable thanks to their lightweight and small size.

Attitude actuators will consist of three NewSpace Systems Libra 80 reaction wheels.

- $I = 212.5x10^{-3}\frac{kg}{m^2}$
- $T = -30°C - +55°C$.
- $P = 188W$ (peak)
- $m = 9.4kg$
- radiation resistance of 30 krad

These were chosen for their rotor moment of inertia value, individual total weight, and peak power consumption. As with the other on board components, these reaction wheels have radiation resistance and reasonable thermal requirements within acceptable range.

For the propulsion system, two different types of thrusters will be used. First, four Aerojet Rocketdyne MR-107S will be attached to the spacecraft for higher thrust maneuvers.

- $F = 85 - 360N$
- $I_{sp} = 225 - 236s$
- $V = 28Vdc$
  - at $P = 34.8W$
  - at $T = +20°C$
- $m = 1.01kg$

These chemical engines provide high thrust, and good specific impulse for their low weight. Furthermore, four MR-107S thrusters were used on OSIRIS-REx, and therefore it was chosen for its flight history on similar missions. A second propulsion system was also added to the spacecraft, a Busek BHT-1500 hall effect thruster.

- $I_{sp} = 1710s$
- $P = 1000W - 2700W$
- $m_{total} = 6.6kg$

This was added to the vehicle to greatly add to the specific impulse of the spacecraft, as a high Δv will be required for the completion of the mission. The planned propellant of this hall effect is Xenon, with Iodine and Krypton also being available for this model of thruster. It boasts an impressive specific impulse, good power range, at a low total weight for cathode and thruster. More detail on the planned early orbit of the spacecraft will follow, but it should be mentioned that the hall effect thruster allows for inexpensive maintenance burns to occur. Together, both propulsion systems add to a total I_{sp} of $2610s$, assuming the lower end of the given ranges.

C. Communication Systems

General Dynamics Mission Systems Small Deep Space Transponder will be used as the onboard communication device. Developed by General Dynamics and NASA JPL, the terminal uses the planned X-band. Its design fits well for deep space mission purposes, is radiation tolerant and its MIL-STD-1553 interfaces for standard and low power means it fits within standardized communication protocol.

The L3 Harris High Compaction Radio Antenna was chosen for its small initial volume and large deployed antenna diameter. The X-band was chosen to be the primary band of communication to meet the required minimum distance of $20,000,000km$. The X-band is the most commonly used band for deep space applications. The uplink and downlink rates were deemed suitable for such a mission.

- $D = 5m$
- $R_{uplink} = 256kbps$
- $R_{downlink} = 64kbps$
- $f = 12GHz$

An in depth link budget is provided in the appendix of this memo.

D. Structures

For the thermal control system, Advanced Cooling Technologies Hik^{TM} and heat pipe assemblies will be used. The high conductivity plates allow for heat to be dispersed and transport heat around the spacecraft as required. The reputation of the manufacturer, and its history in spaceflight were why this system was chosen. It will maintain the temperature of the spacecraft between $-10°C$ and $45°C$ as dictated by the components. The spacecraft will be able to withstand the potential IR backload range of $11\frac{W}{m^2}$ to $88\frac{W}{m^2}$. Surface coating of $5mm$ of AZ-93 white thermal control paint will be used. This has a strong flight history as the radiators on the ISS use it, and has emissivity value of 0.91 ± 0.02 and absoptivity value of 0.15 ± 0.02

E. Payload

A copy of NASA's TAGSAM (Touch-and-Go Sample Acquisition Mechanism) robotic arm which was used on OSIRIS-REx will be used to collect the sample. This was chosen for its high sample mass of $121.6g$collected from Bennu and if similar amounts are collected, this would provide a potential $25g$ of water. Weighing $10.2kg$, it provides a lightweight solution to gathering the material.

Once the sample is gathered it will be transferred to a centrifuge to separate the heavier material from water. The centrifuge which will perform this is the Eppendorf™ 5810R centrifuge.

- $T = -9°C - +40°C$
- $V = 120V$
- $P = 1650W$
- $\omega = 14,000$
- $m = 99kg$

This centrifuge although heavy with high power requirements can be supported by the power and propulsion systems, designed with these requirements in mind. Moreover, the stay time at 2008 EV_5 will allow for many systems to go minimal power while the centrifuge draws most of the power generation. As mentioned in the *Mission Objectives and Background*, a HYDROS™-C will be onboard to perform the electrolysis and test its propulsion capabilities.

- $P = 5 - 25W$
- $m = 2.7kg$

Lastly, a camera will be placed on board to photograph and video the moments before and after impact with a 2336 x 1752 resolution. To do this an Imperx B2340 will be placed on the spacecraft.

- $P = 7W$
- $m = 0.1kg$
- $T = -40°C - +85°C$

In all, the spacecraft will be able to better determine the viability of mining asteroids for use in propulsion. The amount of water collected will also be used to better comprehend the makeup of asteroids such as 341843 2008 EV_5, and their potential use in long duration human spaceflight. The cohesive properties of the asteroid would also be studied to understand any complications which may arise during asteroid mining.

F. Ground Infrastructure

The ground stations and mission operations systems is discussed in the *Operational Plan* section of this memo.

G. Power Generation, Storage, Management and Distribution

With all major components discussed, the power management was designed to be able to withstand all required power with acceptable margin. For power generation, two wings of three Sparkwing solar panels will be used. Folding mechanisms were included in the weight calculations.

- $P = 2160W$
- $A = 7.26m^2$
- $m_{total} = 27.588kg$
- 6 panels total

For power storage, two Eaglepicher technologies COTS 1022 will be used. The manufacturer has experience in space flight, having similar models be used on ORION. This will allow the various systems to operate on low power mode in the event of the solar panels being obstructed for roughly 4 hours.

- $V_{individual} = 22 - 38V$
- $Q_{individual} = 37.8Ah$
- $E_{systemtotal} = 2872.8Wh$
- $T = -10°C - 45°C$

The power conditioning unit will the Airbus Pearl power conditioning and distribution unit. With its lightweight, small volume, it provides a good solution for power management.

- $V = 22 - 38V$
- $P_{SolarArrayPower} = 760W$
- $P_{DistributedPower} = 1500W$
- $m = 2.5kg$
- $395x125x65mm$

A block diagram of the power distribution can be found in the appendix, figure 2.

H. Overview

- $m_{launch} = 551.4kg$ (fuel discussed in *Operational Plan*)
  - $m_{propellant} = 238.4kg$
  - $m_{xenon} = 28.5kg$
- $m_{dry} = 284.5kg$
- $I_{sp_{HallEffect}} = 1710s$
- $I_{sp_{Chemical}} = 900s$
- $P_{Generation} = 2160kW$
- $P_{SurvivalConsumption} = 650W$
- $E_{Storage} = 2872.8Wh$
- $R_{uplink} = 256kbps$
- $R_{downlink} = 64kbps$
- $f_{Comms} = 12GHz$

IV. Orbit and Launch Details

NASA JPL's *Accessible NEAs database*, *Small-Body Mission-Design Tool*, and *ASmall-Body Database Lookup* were all used in the design process of this mission. Proposed is two variants from the *Accessible NEAs database*, with the spacecraft trajectory figure being generated using the *Small-Body Mission-Design Tool*, and *Small-Body Database Lookup* being used originally to decide on 2008 EV_5 as the target asteroid.
Mission key factors:

- Shortest Mission Length Available
  - Launch Date of 09/30/2039
  - 57 days outbound flight time
  - 8 days stay time
  - 33 days inbound flight time
  - $\Delta v = 11,771\frac{m}{s}$ required
- Longest Mission Length Available
  - Launch Date of 06/24/2024
  - 137 days outbound flight time
  - 56 days stay time
  - 169 days inbound flight time
  - $\Delta v = 6,291\frac{m}{s}$ required

The launch vehicle for this mission will be the Atlas V. This was the rocket which brought OSIRIS-REx to space, and provided the spacecraft with approximately $5400\frac{m}{s}$ of Δv. This vehicle was chosen for its strong reputation, similar mission history, and its capability to provide high Δv to the spacecraft. Additionally, United Launch Alliance works with the organization to provide the launch vehicle interface.

The shortest mission will require a total Δv of $11,771\frac{m}{s}$ to complete the shortest duration trip. This requires launching on September 30th, 2039, and will last a total of 98 days. A longer duration mission was considered, however most of those would require launching within the next year, and so the decision was made to go for a shorter duration trip at the expense of higher Δv costs, but grants the mission more time to be optimized with a launch planned so far in the future. Figure 1 shows what this spacecraft will look like.

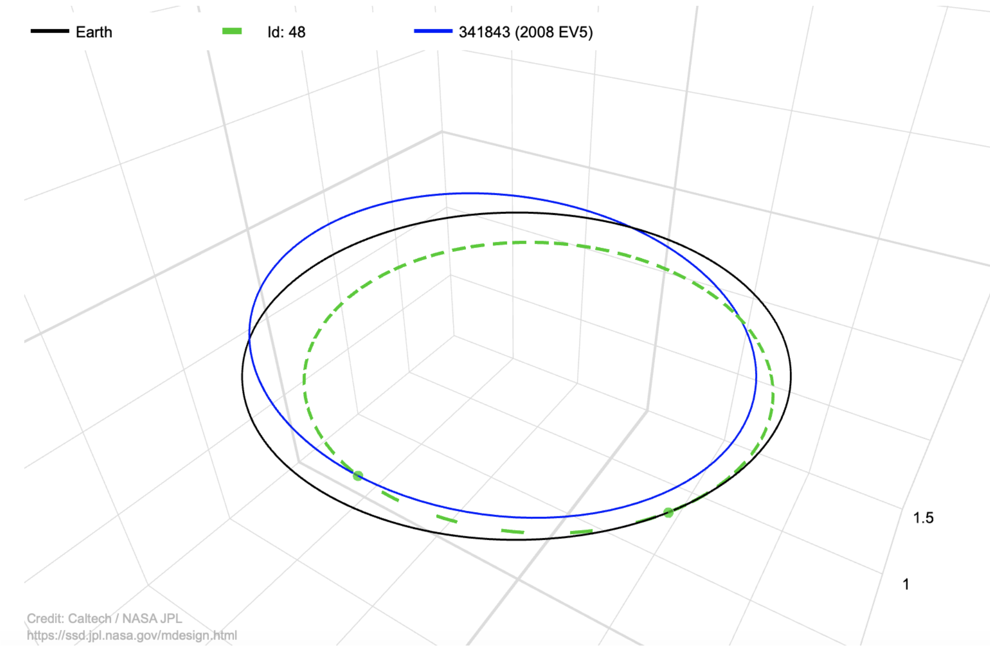

Fig. 1 Spacecraft trajectory (green), in reference to Earth (black), and 2008 EV$_5$ (blue)

The second proposed launch date is a theoretical date as it could not be done in time. The 2039 launch window has possible visits that provided similar performances with cheaper Δv costs. This mission would last 362 days in total, and the longer stay time of 56 days would allow for multiple sample collections given an adjustment in the TAGSAM arm to be able to contain multiple nitrogen gas injections. This would also allow for greater water mass to be used in the HYDROS™-C propulsion system.

This launch date can be changed weather dependent, and it is why the highest Δv mission was chosen as a target. If the launch window changes for whatever reason, the following month still allows launches to complete the mission at the expense of a longer mission duration. The design of the spacecraft allows for the completion potential changes to the flight plan. The spacecraft can enter a distant retrograde orbit, before completing its transfer maneuver to intersect the asteroid's orbit. This orbit family was chosen for its low stability index, requiring few maintenance burns to maintain its position. Any required maintenance burns can be completed with the on board hall effect thruster BHT-1500.

Finally, a chart showing the different launch dates compared to the Δv requirements provided by NASA JPL's *Accessible NEAs database* can be found in the appendix.

V. Operational Plan

Assuming the Atlas V rocket will give the spacecraft $5400\frac{m}{s}$ of Δv once in space, a further $6600\frac{m}{s}$ of Δv is recommended to provide adequate margin. To meet the high Δv requirements with a dry weight of $m_{dry} = 284.5kg$, and proposed $I_{sp} = 900$ as a conservative assumption for the thrusters, $238.4kg$ of propellant for the chemical engines is suggested. This will generate $5000\frac{m}{s}$ of Δv, assuming this propulsion system is used first and must carry the Xenon propellant weight. Furthermore, $28.5kg$ of Xenon will be added to provide the hall effect thruster to propel the spacecraft at dry weight $m_{dry} = 284.5kg$, to generate the remaining $1600\frac{m}{s}$ of Δv required to meet the target $6600\frac{m}{s}$ of Δv. Importantly, the $6600\frac{m}{s}$ of onboard Δv can complete the longer proposed mission without the assumed $5400\frac{m}{s}$ of Δv given by the Atlas V. Propellant mass calculations were done using the rocket equation $\Delta v = I_{sp_{HallEffect}} * g_0 * ln(\frac{m_{dry}+m_{xenon}}{m_{dry}}) + I_{sp_{chemical}} * g_0 * ln(\frac{m_{dry}+m_{xenon}+m_{fuel}}{m_{dry}+m_{xenon}})$. Solving for m_{xenon} in the first half of the equation to get the final $1600\frac{m}{s}$, gives the equation $1600\frac{m}{s} = I_{sp_{HallEffect}} * g_0 * ln(\frac{m_{dry}+m_{xenon}}{m_{dry}})$. With a figure for m_{xenon}, the fuel required for the initial $5000\frac{m}{s}$ generated by the chemical thrusters can be found,

$5000\frac{m}{s} = I_{sp_{chemical}} * g_0 * ln(\frac{m_{dry}+m_{xenon}+m_{fuel}}{m_{dry}+m_{xenon}})$. A full mass budget table can be found in the appendix of this memo, figure 4. This will allow the spacecraft to meet the required Δv with margin for error, using the chemical thrusters first.

For ground stations, the Swedish Space Corporation will be used as they have ground stations across the world with antenna sizes ranging from $7m$ to $15m$ in diameter. This will allow for high uplink and downlink rates as covered in the communication systems section of *Technical Specifications*. They will also provide mission operations systems as they have experience supporting a variety of science missions, and interplanetary missions.

VI. Risk Assessment and Mitigation

Beginning with the star trackers or attitude sensors, having two means if one fails, the other acts as a backup. As previously mentioned, this was also done to compare the two readings to ensure both are accurate. The onboard camera can also help with guidance if an event where both star trackers fail. Having three reaction wheels act as attitude actuators was done both for redundancy, but also for ease of maneuvers. Although only two are required to point the spacecraft, a third was added to act as a failsafe. The large size of the reaction wheels should be able to properly counteract any induced spin from the centrifuge. Although the spinning component is small, its high angular velocity was a concern and as such proper reaction wheels were chosen over control moment gyroscopes.

The high voltage required for the centrifuge will be met using a dc to ac converter. This should provide adequate voltage to the instrument.

Laser communication was not chosen due to the need for a high degree of accuracy. Atmospheric attenuation and atmospheric effects were determined to be too great of a risk and so the decision was made to stick with radio communication. Furthermore, it has little flight history to ensure that no issues will occur with communication.

VII. Regulatory Requirements

One choice made during the design of this mission was to use radio over laser communications. This was done in part due to the aforementioned accuracy issues but also because of Article III of the Outer Space Treaty, Space Weapons and the Law by the US Naval War College. It speaks of the potential of lasers being classified as weapons in outer space. Guidelines for the Long-term Sustainability of Outer Space Activities of the Committee on the Peaceful uses of Space worry about accidental illumination of passing objects in near-Earth space. The design of the mission is conscious of shielded regions of space such as areas on the Moon to not interfere with with radio astronomy work.

Will get clearance from the Federal Aviation Administration (FAA) and from their division of commercial space launches through its Office of Commercial Space Transportation. The launch will also follow standard practice of launching eastward from Florida for added Δv and to launch over an ocean so that if any event occurs, minimal human lives will be at risk. Insurance coverage will be purchased to mitigate the risks and ensure that compensation for any damage or loss resulting from the launch is covered.

VIII. Conclusion

In all, the mission meets the given constraints of being capable of operating in a heliocentric orbit between $0.7AU$ and $1.2AU$, the spacecraft is equipped with over $5000\frac{m}{s}$ of Δv, can communicate at distances over $20,000,000km$, and collects scientific data from the 2008 EV_5.

Further optimization and design of some specific components in house could be done to help increase performance, lower cost, and lower present risks in the mission. One method of analysis that could be used is multi objective optimization where different models on each subsystem are created to better search for different potential products and narrow search parameters. By generating multiple different designs, mission trajectories and spacecraft can be scored and will allow the design team to better understand where the spacecraft is lacking and any potential improvements overseen during the development process. This way, smaller margins on systems like power and propulsion can be achieved to lower mission costs.

IX. Appendix

A. Link Budget

PARAMETER	UPLINK (Earth - Spacecraft)	DOWNLINK (Spacecraft - Earth)	UNITS	Symbol	Reference
Speed of Light	*3.00E+08*	*3.00E+08*	*m/s*	*C=l*f*	*constant*
Frequency	*1.20E+10*	*1.20E+10*	*Hz*	*f*	*Up/Down/Crosslink Ka band (25.5-30 GHz), Earth downlink S band (2-4 GHz)*
Wavelength	2.50E-02	2.50E-02	m	λ	calculated from given
Range	*2.00E+07*	*2.00E+07*	*km*	*R*	*Uplink/Downlink - L2 max, Crosslink - MOO, Earth Downlink - L2 to Groundstation*
Boltzman's Constant	*1.38E-23*	*1.38E-23*	*W/(Hz-K)*	*k*	*constant*
Data Parameters	*Uplink*	*Downlink*	*Units*	*Symbol*	*Reference*
Bit Error Rate / Probablility of Bit Error	*1.00E-05*	*1.00E-05*	*[-]*	*BER*	*Reference: Dr. Palo and Trades*
Data Coding Scheme	*QPSK*	*QPSK*			*Input: chosen modulation (SMAD Tab.13-10)*
Required Bit Energy to Noise Ratio	9.8	9.8	dB	Eb/No	Approximated using SMAD Figure 13.9
Data Rate	*2.56E+05*	*6.40E+04*	*bps (Hz)*	*R*	*Input: based on mission / objective - C1,C2,C3*
Required Carrier to Noise Ratio	63.88239965	57.86179974	dB	C/No	C/No = Pr/No = (Eb/No)*R
Required Design Margin	*6*	*6*	*dB*		*Input: design rule (Hoffmann chap. 9.4.4)(3,6,10 db) (low,decent,comfortable)*
Minimum C/No	**69.88239965**	**63.86179974**	**dB-Hz**		Hoffmann 9.4.4 & SMAD 13-15a | Minimum C/No = C/No + Margin = Eb/No
Noise (applies to receiving elements)	*Uplink*	*Downlink*	*Units*	*Symbol*	*Reference*
Reference Temperature	*290*	*290*	*K*	*To*	*SMAD Eqn13-24 (Generally constant) Always 290 K - Dr. Palo*
Receive Antenna Efficiency	*0.4*	*0.4*	*[-]*	*ɦ*	*Constant for parabolic antenna - Brodie - for now assume 40% efficient*
Receive Antenna Physical Temperature (User)	400	286.15	K	Tphys	Actual measured or theoretical temp at that position - Worst case HOT (Kelvin)
External "scene" Noise Temperature	250	300	K	Text	Uplink: Receiver on the S/C looks at Earth which is 260K | Downlink: Receiver looks at Space which is 25K, From Palo
Antenna Noise Temperature	340	291.69	K	Tant	Tant = eff*Text+(1-eff)*Tphys - This should be btwn Tphys and Text
Receiver Cable Loss	*-1*	*-1*	*dB*	*Lc*	*SMAD Table 13-10, higher frequency increases. (reference LMR-400)*
Receiver Cable Loss	*0.7943282347*	*0.7943282347*	Linear	*Lc*	*Lc_Linear = 10^(Lc_dB/10)*
Receiver Noise Figure (based on receiver)	*3.773213792*	*2.988832814*	*dB*	*NF*	*NF = 10*log10(F)*
Receiver Noise Factor	2.384083045	1.990138408	[-]	F	F = 1 + (Tphys/Tr)
Receiver Noise Temperature	289	289	K	Tr	SMAD Table 13-10 (20 GHz) - vender specific
Receiver System Noise Temperature	879	880.69	K	Ts	Sum of noises (usually 700-750 K) Currently Ts = Text + Tant + Tr
Receiver System Noise Power	-199.1613204	-199.1529785	dBW-Hz	No	Need equation, No = k*Ts (NEED TO CONVERT TO DB)
Receiver Parameters:	*Uplink*	*Downlink*	*Units*	*Symbol*	*Reference*
Receive Antenna Diameter	*5*	*10*	*m*	*D*	*Earth - DSN 34 m | Assumption - Only Parabolic antennas - Consider Smaller Antennas*
Receive Antenna Area	19.63495408	78.53981634	m^2	A	Assumption - parabolic = pi*r^2
Receive Antenna Efficiency	*0.4*	*0.4*	*[-]*	*ɦ*	*Source - Brodie Wllace - Dr. Palo's Phd student*
Receive Antenna Gain	50.76361653	56.78421644	dBi	Gr	function of diameter G = 10log10(4*pi*Ae/λ^2)
Receive Antenna Effective Area	10.79922475	43.19689899	m^2	Ae	Ae = 0.55 * A (Parabolic reduced equation)
Receive Antenna Beamwidth	3.50E-01	1.75E-01	degrees	qr	Directivity vs Gain function qr = 70 * (λ/D)
Receive Antenna Pointing Accuracy	*0.1*	*0.1*	*degrees*	*er*	*const (solve once) - SMAD pg 130*
Receive Antenna Pointing Loss	-3	-3	dB	Lpr	SMAD 13-21 | -12(e/qr)^2 loss from angle errors (Assumption - 3 db loss)
Receiver Cable Loss (see noise)	*-1*	*-1*	*dB*	*Lc*	*Input: typical value*
Receiver Figure of Merit	21.32372778	27.33598579	dB/K	FOM	FOM = Gr - 10log10(Ts)
Propagation Parameters:	*Uplink*	*Downlink*	*Units*	*Symbol*	*Reference*
Space Loss	-260.045997	-260.045997	dB	Ls	from altitude - free space path loss (using isotropic equation currently, change this to reflect directivity)
Atmospheric Attenuation (clear air)	0	-7	dB	La	Worst case - Avg 7db Attenuation Daily
Polarization Loss			*dB*	*Lp*	*Input: typical value constant - decide polarization, right or left hand circular*
Propagation Losses	*-260.045997*	*-267.045997*	*dB*		Sum of all propagation losses = space + atmospheric attenuation

Transmitter Parameters:	Uplink	Downlink	Units	Symbol	Reference
Transmit Antenna Diameter	10	5	m	D	reciprocity Tx is Rx and Rx is Tx: Linked to row 33 but swapped - **Design Variable**
Transmit Antenna Area	78.53981634	19.63495408	m^2	A	reciprocity Tx is Rx and Rx is Tx: Linked to row 34 but swapped - **pi*r^2**
Transmit Antenna Efficiency	0.4	0.4	[-]	h	*Reciprocity - Refer Row 39*
Transmit Antenna Effective Area	43.19689899	10.79922475	m2	Ae	Need to decide user antenna for Uplink| Currently assuming all parabolic
Transmit Antenna Gain	56.78421644	50.76361653	dBi	Gt	G = 10log10(4*pi*Ae/λ^2) (25-35 dB = Dr. Palo) possibly
Transmit Antenna Beamwidth	3.50E-01	3.50E-01	degrees	qt	Reciprocity - Refer Row 42
Transmit Antenna Pointing Accuracy	0.1	0.1	degrees	et	Assumption
Transmit Antenna Pointing Loss	-3	-3	dB	Lpt	Assumption* double check number
Transmit Line Loss	-0.5	-0.5	dB	Lt	*Input: based on chosen cable/geometry, asume same as reciever*
Tramsmit Power, Linear	1000	1.00E+03	W		*Input: chosen transmitter DSN data & trade*
Transmit Power	30	30	dBW	Pt	Converted from linear
Effective Isotropic Radiated Power	86.28421644	80.26361653	dBW	EIRP	EIRP = Pt + Gt (in dB)

Link Budget:	Uplink	Downlink	Units	Symbol	Reference
Effective Isotropic Radiated Power (63)	86.28421644	80.26361653	dBW	EIRP	linked to row 66
Pointing Losses	-3	-3	dB		sum of transmit and receive pointing losses 1/10 -> few db (can assume 3db)
Propagation Losses (49)	-261.045997	-268.045997	dB	L	linked to row 47 (should be negative)
Receive Antenna Gain (38)	50.76361653	56.78421644	dB	Gr	linked to row 36
Received Power	-125.998164	-132.998164	dBW	Pr	Pr = EIRP + Lpointing +Lprop + Gr (asuuming the losses are negative)
Receiver System Noise Power (31)	-199.1613204	-199.1529785	dBW-Hz	No	linked to row 30
Received Carrier to Noise Ratio	73.16315634	66.15481444	dB-Hz	C/No	Received power - system noise power
Minimum C/No (20)	69.88239965	63.86179974	dB-Hz	C/No	linked to row 20
Link Margin	**3.280756684**	**2.293014697**	**dB**		received C/No - Minimum Pr/No **(Link Closes if more than 6 dB)**

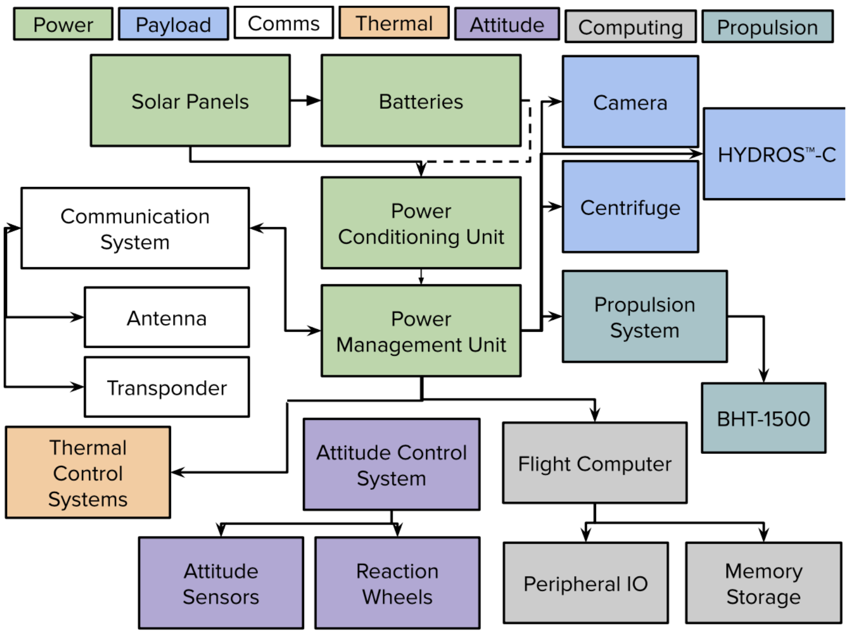

Fig. 2 Power Distribution Block Diagram

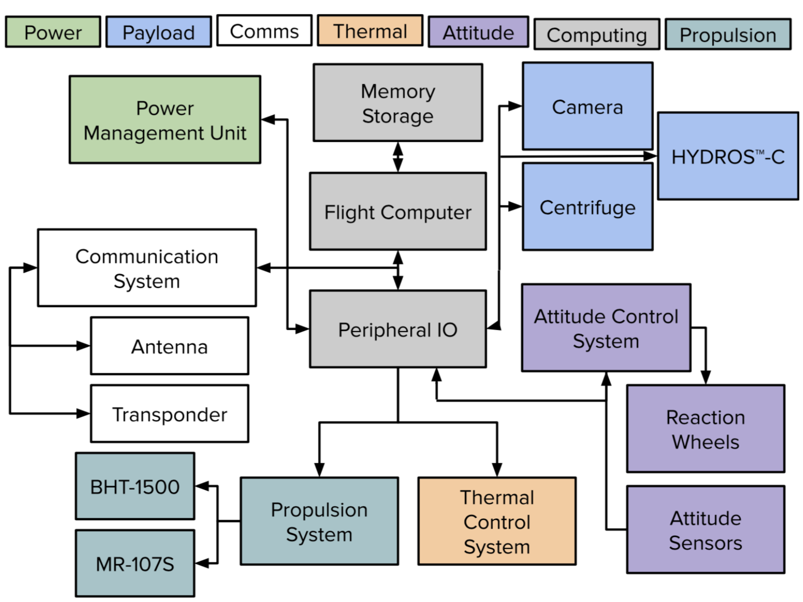

Fig. 3 Data Distribution Block Diagram

Solar Panels	27.588 kg
Batteries	13.5 kg (total)
Power Conditioning Unit	2.5 kg
Flight Computer	10 kg
Memory Storage Operating	0.75 kg
Peripheral IO	0.5 kg
Attitude Sensors	3.4 kg (total)
Attitude Actuators	28.2 kg (total)
Thrusters	4.04 kg (total)
Hall Effect Thruster	6.6 kg
Thermal Control System	10 kg
Transponder	3.2 kg
Antenna	22 kg
Camera	0.337 kg
HYDROS-C	2.7 kg
Centrifuge	99 kg
TAGSAM	10.2 kg
Framework	40 kg
Dry Total	284.5 kg
Propellant	238.4 kg
Xenon	28.5 kg
Launch Total	551.4 kg

Fig. 4 Spacecraft Mass Budget

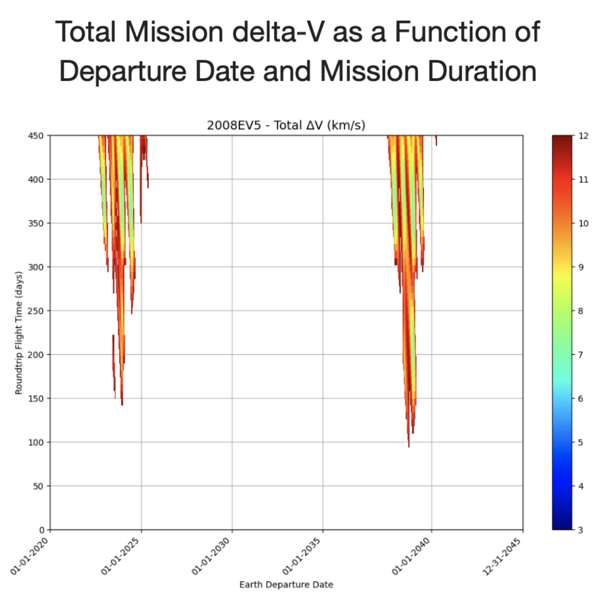

Fig. 5 NASA JPL's *Accessible NEAs database*

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