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Preliminary Estimation of Launch Mass for a Direct Transfer Earth-Uranus Mission

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Abstract: A mission to Uranus facilitates a thorough examination of Uranus, its rings, satellites, and other planetary objects. To travel from Earth to any other planet, a variety of approaches can be used. Despite the standard gravity assist methods to reach the ice-giant Uranus, a direct transfer mission is also feasible. This work provides an overview of the preliminary estimation of the launch mass for a direct transfer mission. The departure year 2022-2030 is considered for this study. The payload and spacecraft subsystems for the proposed mission were selected based on past interplanetary missions. The estimated mass of the scientific instrument of this work was found to be 147.5 Kg. The mass of the payload is 11% of the spacecraft's dry mass. The delta-V for the various departure years is obtained using Lambert's problem for the different time frames. Delta-V and the launch mass are calculated for a range of 15.5-8.5 years of time-of-flight. The upper limit of the time-of-flight is selected based on a Hohmann transfer. Launch mass decreases from 15.5-13.5 years of time-of-flight and then increases to a maximum value at 8.5 years. For time-of-flight of 13.5 and 12.5 years, the delta-V and the launch mass obtained are optimum, 6.7 km/s and 1700 kg, respectively. In the period selected, these minimum values are observed for the departure year 2022. The data obtained reveal the feasibility of such a mission in the near future.

Keywords: Uranus, Ice-Giants, Preliminary Mission Design, Launch Mass, Spacecraft Subsystems, Delta-V

1. Introduction

Several missions have been carried out since the advent of the space era to study and comprehend our galaxy. Numerous manned and unmanned missions were successfully launched to explore various terrestrial bodies. Voyager 2, which was the first spacecraft to date to pass over Saturn and Uranus, is one such unmanned mission by NASA. However, Voyager 2 was never intended merely for Uranus but has completed a flyby. It never did a thorough study of Uranus and its structures. As a result, a mission to Uranus became vital.

Hughes has listed different orbiter missions to Uranus and Neptune for periods stretching between 2024 and 2038 [1]. A variety of alternative approach trajectories are developed to see if any alterations are needed to avoid rings of the planets and to establish probe entrance circumstances.

QUEST Uranus orbiter mission concept was investigated to see the viability of a mission with a reduced budget and mass framework [2]. Some of the objectives of this mission were to identify Uranus' mass configuration, cause of the magnetic field, ambiance, satellites, and rings. To achieve this, it used an Atlas V-551 20 kW power disposable launch vehicle, two NEXT solar-electric propulsion thrusters as well as a bi-propellant chemical propulsion system for maneuvers.

A thorough hunt of interplanetary routes and an initial study of low power trajectories for Uranus, Neptune, and Pluto was conducted [3]. To propel the launch vehicle, Kilo-power Electric Propulsion was used in this investigation for the mission prospects from 2025 to 2045.

Landau et al. employed the existing set of Solar Electric

Propulsion to Uranus and Neptune by examining a large design space of gravity-assist strings for pathways [4].

The current development in onboard sensors for infrared remote detection of planets and moons was discussed by Stankevich [5]. Innovative equipment and sensors are developed as technology advances. As a result, more sophisticated devices such as long-wave infrared spectral range extension, uncooled microbolometer indicators of infrared radiation, and scopes to change the field of view replace conventional ones.

QUEST mission strived to fill the void left by decades of ice giant operations by using Uranus as a laboratory [6]. It was proposed to relate Uranus' magnetic, thermal, and thermal-compositional shape compared to more enormous planets.

D'Andrea et al. conducted a mission analysis as part of the MORPHEUS feasibility study, to assemble data on the Uranus system [7]. But the chosen trajectory design caused a sensible overall cost.

Arridge et al. described the science case and technological hurdles for an orbital trip to Uranus to study its composition and atmospheric physics [8]. This work was based on a government report presented to the ESA's science topics in 2013.

The JIMO spacecraft's nuclear electric propulsion system was investigated that allowed direct missions to Jupiter, Saturn, and possibly Uranus, but not to Neptune or Pluto [9].

An extensive study of noble gases undetected by any other method was borne out to determine the abundances of He, Ne, Ar, Kr, Xe, and their isotopes in well-mixed profusions [10].

Gravity-assist techniques are used extensively for a majority of interplanetary missions. The foremost practical restriction on using a gravity assist maneuver is that planets and other massive masses are rarely in positions that would allow travel to a specific location. In this paper, the preliminary mission design is established by calculating the dry mass, propellant mass, and total launch mass of the spacecraft. Due to their being in winter darkness (as was the situation during the 1986 Voyager 2 flyby), flights to Uranus after the late 2030s will not be able to scan the never-before-seen Northern Hemispheres of the satellites [11]. Hence, the departure years 2022 – 2030 are considered for different times of flight. Considering all the mission complexities, challenges, limitations, and possibilities, the authors have chosen a direct transfer mission to Uranus. Based on past interplanetary missions, the scientific instruments, and subsystems are selected for this mission.

2. Materials and Methods

Research focusing on Uranus would help us to unravel the mystery regarding this exoplanet. The spacecraft design for a Uranus mission is critical for designing an orbiter with the payload for trajectory analysis as well as studying the Uranus exploration. It should be modeled optimally to reduce the overall load on the spacecraft with a detailed analysis of all its subsystems.

2.1. Payload Design: Scientific Instruments for Uranus Exploration

The instruments proposed for our mission are reflective of the scientific goals and objectives. The specifications of each scientific instrument are provided in table 1.

2.1.1. Uranian Plasma Wave Detector (UPWD)

UPWD will be used to calculate the rays of plasma emitted from the Uranus magnetosphere and the relation between solar winds and plasma wave phenomena to be found at different frequencies. UPWD is developed from Cassini modifications are done to shift frequencies according to Uranus [12]. It is expected to detect waves between 58.8 kHz to 900 kHz.

2.1.2. Uranus Infrared Spectrometer (UIS)

UIS will work as a remote-sensing thermometer. Every object that does not have absolute zero temperature emits light in different waves and of different strengths and by analyzing this its temperature can be studied. UIS will measure the infrared thermal radiation from our target. This might shed light on how Uranus' violent seasons affect its rings and atmosphere. UIS will capture the light, split it into its component's wavelength, and then measure the strength of every wavelength to measure the temperature and the material composition of the target object [13]. It will work on the same principle of raindrop splitting light into rainbow the only difference will be the light will be in the infrared red spectrum.

2.1.3. Uranian Aurora Experiment (UAE)

UAE detects the electrons around the orbiter [14]. It has 3 detectors to detect and measure the ions at different places and of different strengths.

2.1.4. Uranus Camera (UraCam)

UraCam is a visible light camera mounted atop the orbiter structure. UraCam is used to provide imaging for public outreach. Apart from this, it is utilized to examine the moons, rings, and pictures of moons. UraCam produces 4 channel images at 100 megabits per channel.

2.1.5. Microwave Radiometer (MR)

MR works on the principle of measuring temperature through brightness, which is again dependent on both the pressure and temperature profile. The study will be done by observing the different levels of opacity at different places. Using the available knowledge regarding Uranus atmospheric gases absorption, models can be created. The pressure above and below the clouds can be measured at the predicted location. Based on this the gases present in abundance can be predicted.

2.1.6. Antenna (RadAnt)

RadAnt is used to transmit and receive radio signals to and from the earth [15]. The amplitude-phase received is used to obtain electron density and other data like atmospheric temperature, pressure, density, and other gravitational parameters. It works with minimal additional mass, power, and cost. And the orbiter doesn’t have to take the processing burden. Data can be acquired at 250 km and 0.3 bar to 2 bars.

2.1.7. Uranus Ultraviolet Imaging Spectrograph (UUIS)

UUIS creates pictures by observing ultraviolet light. The instrument splits the light into different components of colors that could help us to study the material the light is passing through. UUIS will be a box of 2 telescopes that could see UV lights. After collecting the light, the instruments will process the pattern and give us an insight into the chemical composition of matter.

2.1.8. Uranus Dust Analyser (UDA)

It detects dust particles one-hundredth of a micrometer wide and as small as one-millionth of a millimeter. The particle enters the UDA, and the instruments determine the particle's charge, speed, and size. For dust composition, the particle is broken down into smaller parts. Dust wanders in the Uranian system. Not all come from its system, some come from beyond the solar system. Some particles were part of the rings and moons. UDA will study this, helping us understand where the particle comes from or how they interact with the Uranian system and the magnetosphere. The high-rate detectors will study the rings and chemical composition of their particles.

2.1.9. Visible Infrared Mapping Spectrometer (VIMAS)

VIMAS will collect light from the visible spectrum and ISO from the higher wavelength infrared zone. It will be used to study the temperature of the Uranian system. The VIMAS will observe the light reflected and emitted from different materials. It then disintegrates the light into hundreds of separate colors and thus gives us results regarding determining its origin. Light changes after moving through dust and these changes will be captured by our UUIS and wavelengths that are present or absent. This instrument consists of two cameras-one for visible and another for infrared.

2.1.10. Uranus Magnetometer (URAM)

Uranus Magnetometer is packed with the heritage from Cassini including scalar/vector magnetometer and fluxgate magnetometer. URAM measures the magnitude, and the direction of the magnetic field, as our orbiter orbits around the planets. The range of URAM is dynamic and modifiable to obtain data at appropriate resolution during various phases of the insertion and orbits. The maximum observed magnetic field strength by Voyager 2 flyby was 413nT. Thus, URAM will operate at a field strength of +/- 25000nT at the highest dynamic range. URAM will work at 100 vectors/second, producing 1.6 Kbits/secs. All data will be transmitted to Earth unprocessed. The baseline instrument is the Juno magnetometer where the Juno mission could achieve a degree 10 harmonic model after nine orbits. The objective is to collect data to understand Uranus magnetic field.

2.2. Spacecraft Subsystem and Its Design

The spacecraft subsystem is categorized into seven main types. They are the power system, propulsion system, heat management, Attitude Control System (ACS), command and data handling, structure and configuration, and communication system [16-18]. Each subsystem plays a vital role in the design and operation of the spacecraft and so must be comprehensively studied.

Table 1. *Scientific instrument specifications.*

Scientific Instrument	Specifications
Uranian Plasma Wave Detector (UPWD)	Mass – 4.34 kg, Peak operating power – 6 W
Uranus Infrared Spectrometer (UIS)	Mass – 31.14 kg, Peak operating power -29.81 W, Data rate – 6 kb/secs
Uranian Aurora Experiment (UAE)	UAE -D – to measure electrons from 0.1 to 100 keV, UAE – I – for detecting electrons ions from 0.5 eV to 50 keV, Mass – 2.8 kg
Uranus Camera (UraCam)	Mass – 5 kg, Peak operating power – 5.6 W, Data per orbit – 1800
Microwave Radiometer (MR)	Mass – 28.54 kg, Power – 28.73 W
Antenna (RadAnt)	Mass – 14 kg
Uranus ultraviolet imaging spectrograph (UUIS)	Mass – 12.40 kg, operating power – 10.08 W
Dust Analyser (UDA)	Mass – 13.36 kg, Peak operating power –15.38 W
Visible Infrared Mapping Spectrometer (VIMAS)	Mass – 28.78 kg, Peak operating power – 27.20 W
Uranus Magnetometer (URAM)	Mass – 7.12 kg, Peak operating power – 14 W
Total payload mass	147.5 Kg

2.2.1. Power System

The power system provides the energy which acts as the life of the spacecraft. Spacecraft can perform better until they have power during the mission and the remaining errors will be configured by the ground station. This system has block units namely power generation, power reservoirs, power control systems, and power loads. This subsystem energizes the functionality of the power-based components, it could be communication, data handling, etc.

Power Generators: Here triple-junction gallium arsenide photovoltaic solar array of the following dimensions: 36.4m^2 (18.2m^2 each) and a length of 8.2m are used to generate 10 kW of solar power. It is also evident that solar radiation near Uranus i.e., 19.6 AU from the sun does not produce much solar power. So, it deficits the power required to progress further. Hence, alternatively, another power generator named General Purpose Heat Source- Radioisotope Thermionic Generator (GPHS-RTG) is used to generate powerful energy each of which produces 300W of power.

To obtain the preliminary estimate of power generators a trade study was conducted using data obtained from previous missions. It was found that for a power requirement of 1.5kW, 6 EMMRTG is required, which increases the overall mass. Apart from this requirement additional batteries namely Ni-Cd, and Li-ion are included. Li-ion battery is used to store the energy due to its low mass and storage capacity of 1000 W. Accordingly 3 units are used to reserve the energy, alternatively power generators help the vehicle to power up and propel forward.

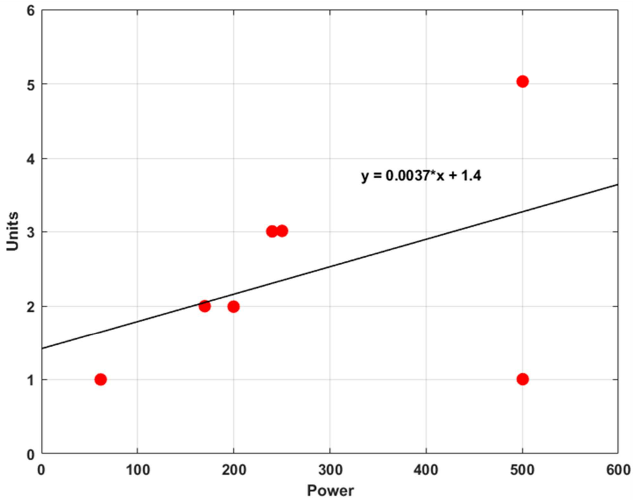

Figure 1. *Trade studies of RTG.*

2.2.2. Propulsion

To propel forward, the orbiter employs distinct phases-departure phase, orbit transfer phase, and arrival phase accordingly the propulsion is to be chosen. Electrostatic ion propulsion is chosen due to following reasons.

i. Higher specific impulse (>3000 seconds).
ii. More reliable.
iii. Quick and fast.
iv. Safer.

Ion propulsion makes use of ions to produce thrust concerning the momentum conservation principle i.e., ionize the propellant through the electron head-on collision. High energy electron i.e., the negative charge collides with the available propellant atoms causing it to release an electron from the propellant atom and finally result in a positively charged ion. The gas generated now has a combination of the positive and negative electrons in an equal amount which results in the production of electric charge i.e., thrust in addition to this a component named neutralizer makes the thrust electrically neutral.

Propellant Selection: In the past year, there is a wide variety of noble gas propellants used due to their non-radioactive element and the availability of noble gas namely Xenon, neon, and krypton. Xenon is used for its better performance range also higher atomic number element should be used as it will not affect the efficiency of the thruster. Its maximum specific impulse at 2.3 KW is 3120 seconds. The assumed specific impulse is to be 3000 seconds to carry extra fuel onboard the spacecraft for other purposes.

2.2.3. Heat Management

It is required for the mission, payload components to maintain their temperature for the optimal working of it under all circumstances where there is an optimal temperature range of the components. These temperature increases can be from solar radiation, or through the earth. So accordingly, components must be designed to eject the heat here the components named radiator, louvers, heat pipes, and insulation are used to protect from temperature increase. Also due to the increase in temperature, there is a chance of structural dimension shifts.

Due to exposure to solar radiation the internal temperature of the system gets rises, and also due to the internal heat generation by the components of the vehicle the temperature gets again increases so to maintain the temperature of the alternative evolution of the technology is been done where the material is such as aluminum foil sheet are used multiple layers such that solar radiation doesn't penetrate much into the payload system also it is highly required for the internal components such as the batteries, propulsion, C and DH systems to maintain the temperature. It is because of the optimal limitation of the working temperature of components.

The battery has an optimal temperature range of 5°C-20°C and a propulsion unit to be maintained at the optimal temperature range of 20°C-40°C. Here two Heat pipes are used to eliminate the heat produced from the battery, magnetometer towards the radiator heat pipes is placed with ammonia as working fluid. Radiators are used to radiate out the heat generated in the system.

Similarly, louvers are used according to the number of radiators. Here louvers are the shutter of the radiator which works on expansion of the bimetallic strip and the shutter releases when the temperature rises in the radiator to emit the heat to the external surrounding. Here aluminum gold foils are used in 10 numbers of thickness 0.5 in each to which the system operates in an optimal temperature range.

Hexagonal area = 23.38 m^2;
Radiator area 0.6m*0.6m=3.6m^2;
Number of radiators required = 6;
Number of louvers = 6.

2.2.4. Attitude Control System

Spacecraft attitude is denoted by three main axes namely pitch, roll, and yaw axis. The attitude control system controls the system attitude error with the inertial frame of reference. Attitude control is done by a set of components namely sensors to detect the attitude error, which is a feedback controller, thrusters, reaction wheels, etc. It is required for a spacecraft to orient according to the user inputs. Here ACS are those systems that change the orientation of the spacecraft according to the needs of the program. There are a set of stabilizers, sensors, and actuators to monitor the attitude of the spacecraft. The four types of stabilizers available are namely spin-stabilized, three-axis stabilization, dual-spin, and gravity gradient stabilization. Here three-axis stabilization is chosen due to its higher pointing accuracy of 0.0001 deg. It can adjust randomly according to the system or mission needs. It has sensors and a fast-maneuvering capacity, to adjust the equipped solar panels according to the position of the sun.

Momentum wheels of 2 units are used.
Star scanners 1 unit.
Thrusters = 3

2.2.5. Command and Data Handling

The subsystem named command and data handling handles and stores the data. There are three sorts of data handled: science data gathered via payload, ground station data, and commands. The data handling function is to collect, convert analog to digital, format store, map, and encode data. To collect data storage devices such as solid-state tape recorders are used. Spacecraft computer specification:

Processor: 1.66GHz;

SDRAM: 2 GB DDR2;

Flash memory: 1GB;

3.125 Gbps/lane Serial Rapid IO interfaces for maximum performance and control;

Solid-state recorder: MINISTER SSR – H, storage capacity: 4GB;

Uplink Frequency: 457.6 MHz;

Downlink Frequency: 467.6 MHz.

2.2.6. Structure and Configuration

Structures are an integral part of the spacecraft it acts as the framework. Structure design should be done accordingly to withstand the payload and other components. The types of structural shapes namely polygon, cube, sphere, etc. must be chosen according to the volume capacity. If one can minimize the spacecraft structure mass, more amount of fuel can be added. For the structure shape selection, a few criteria such as Field Of View (FOV), nadir pointing, propulsion, etc. must be noted before opting for a particular shape. When it comes to the cubic structure, the volume capacity is less, and when it comes to the sphere, the volume capacity and FOV are less.

In this study, the hexagonal structure is chosen as the bus structure. It has more volume capacity to place the payload, even if, there is a disadvantage of difficulty in designing.

Structure Dimensions and Configuration: 1.732 m on each side from the origin and 3m on each side. Length=1.8m. Circular protrusion for antennae mounts of diameter =0.5 m. Aluminum 7075-T6 alloy for the bus frame and similarly for the bus structure composite Kevlar is considered.

2.2.7. Communications

Communication is the most important part for anyone to get the work done. Communication here takes place with two steps namely uplink and downlink where uplink is the data sent from the ground stations, and down station is those which is received from the spacecraft. Followed by this, to increase the range of communication, antennas are used namely HGA, LGA, and MGA antennae. Further communication is done through X, S, Ka, and Ku bands chosen according to the frequency ranges. They are used according to the phase of the trajectory.

X and S-band are used for planetary missions, and it is used accordingly during the respective phase (cruise, orbit). S-band has the limitation of a downlink rate of 62.5kbps.

Antennae: Typically, three main types of antennae are used. Low Gain, Middle Gain, and High gain antennae (LGA, MGA, HGA).

For this mission, 2-LGA is selected for communications: One LGA for near-earth communication and another one for emergency communications. Apart from this DSN ground station supports 34m and 70m parabolic antennae.

HGA = 1 unit;

LGA /MGA= 2 units;

Diameter of HGA = 4m;

Length of antennae = 3m;

Total payload mass = 148 kg.

The spacecraft's dry mass is approximately 11% of the payload mass.

Total Spacecraft dry mass = (148/11) *100 = 1345.4545 kg.

The planet has tilted 90 deg to its axis to avoid intersecting with one of the many rings that surround Uranus. If particles in these rings impact the spacecraft, it will likely result in a mission failure. Since it is a 3D trajectory, no assumptions were made for the interplanetary trajectory that the planetary orbits were circular, and they were all coplanar in the ecliptic plane. The position and velocity vectors of the planets are considered for real-time. It was assumed that the mass distribution of the planets was radially symmetric and that disturbing forces such as solar radiation pressure, electromagnetic forces, aerodynamic forces, etc. acting on the spacecraft are negligible. The values obtained for the launch mass are approximated to the highest order to eliminate the decimal values.

3. Results & Discussions

The payload mass of the scientific instruments used to explore Uranus is 148 kg. The payload mass is approximately 11% of the spacecraft's dry mass. Consequently, the spacecraft's dry mass is 1345.4545 kg.

The spacecraft uses NSTAR 2.3 KW Ion Thruster engine with xenon as its propellant that has a maximum specific impulse of 3120 seconds. For power, three types of generators are used. Triple Junction Gallium Arsenide Photo Voltaic Solar Array that generates 10KW power at 1 AU.

General Purpose Heat Source Radioisotope Thermionic Generator that generates up to 300 watts of power, and three pieces of Lithium-Ion Battery that can store up to 3 KW. Structural configuration employed 22.5m*9.5m*6m dimension. Figure 2 shows the Uranus orbiter CAD model.

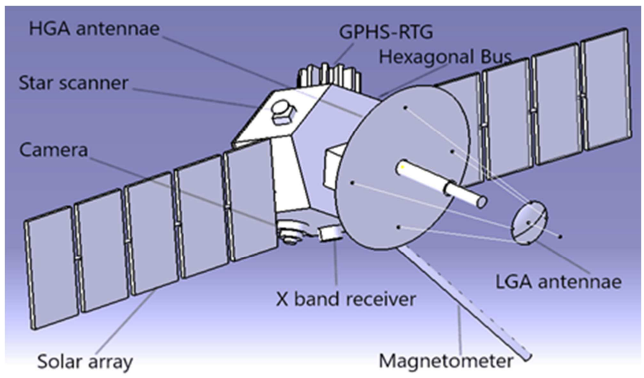

Figure 2. *Uranus orbiter CAD model.*

3.1. Delta-V for Various Time-of-Flight

To obtain the launch mass the delta-V has been found using Lambert's problem [19-21] for the departure year 2022-2030.

The direct transfer trajectory analysis was performed to obtain the optimum delta-V for the Uranus mission [22]. The upper limit of the time of flight is decided by performing a Hohmann transfer for the mission. The launch mass is calculated by the rocket equation [23] as in equation (1).

$$\frac{\Delta m}{m} = 1 - e^{\frac{\Delta V}{I_{sp} g_o}} \tag{1}$$

Where m indicates the mass of the spacecraft before the burn, go is the sea-level acceleration due to gravity and Isp is the specific impulse of the propellant used in the spacecraft.

From 15.5 years to 11.5 years of time-of-flight, delta-V increases (figures 3-10). Delta-V is almost constant for 10.5 years. For 9.5 and 8.5 years of time-of-flight delta-V decreases. The minimum delta-V obtained is 6.7 km/s for the time-of-flight of 13.5 years (figure 11).

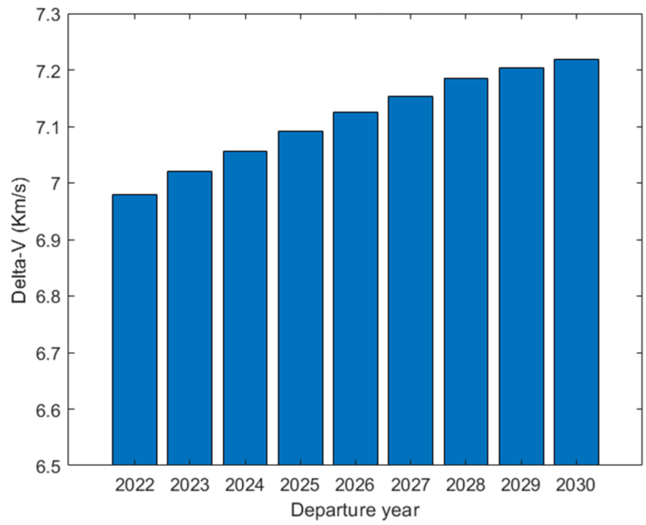

Figure 3. *Delta-V for TOF= 15.5 years.*

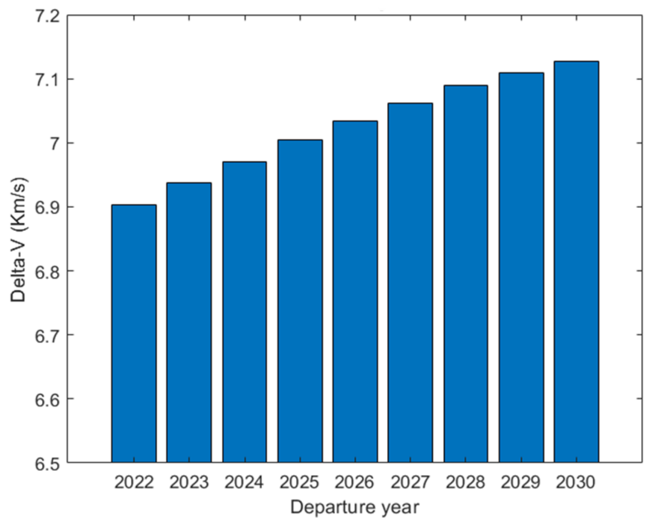

Figure 4. *Delta-V for TOF= 14.5 years.*

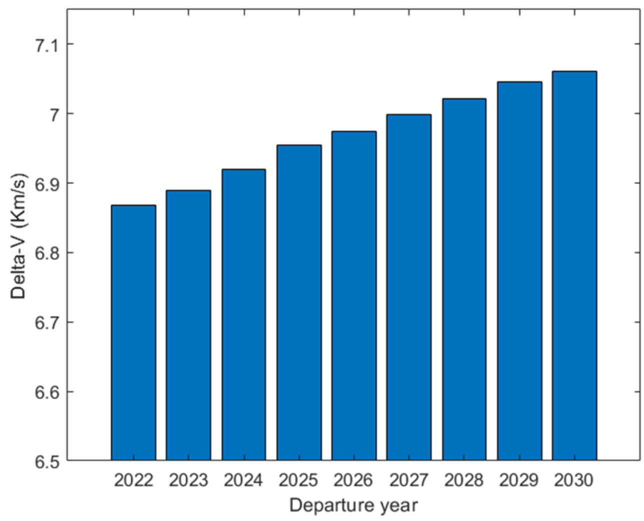

Figure 5. *Delta-V for TOF= 13.5 years.*

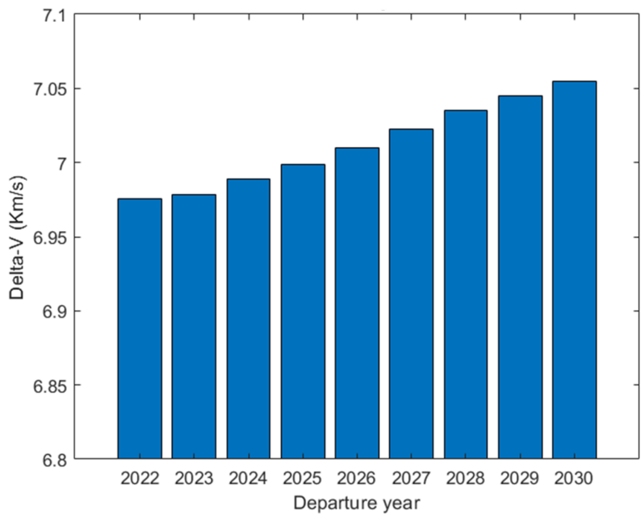

Figure 6. *Delta-V for TOF= 12.5 years.*

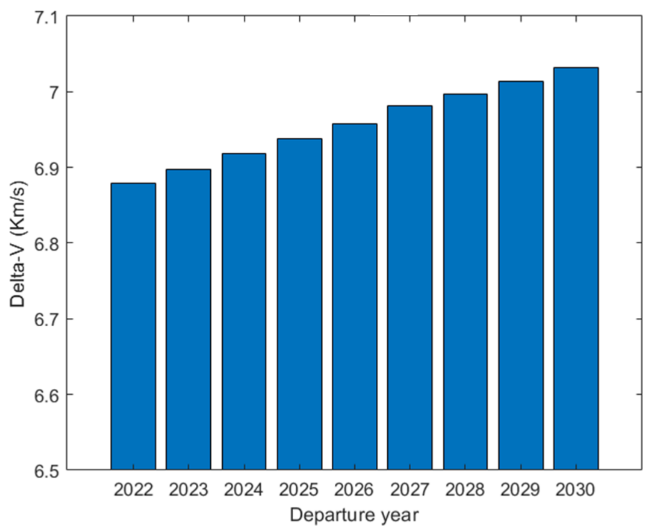

Figure 7. *Delta-V for TOF= 11.5 years.*

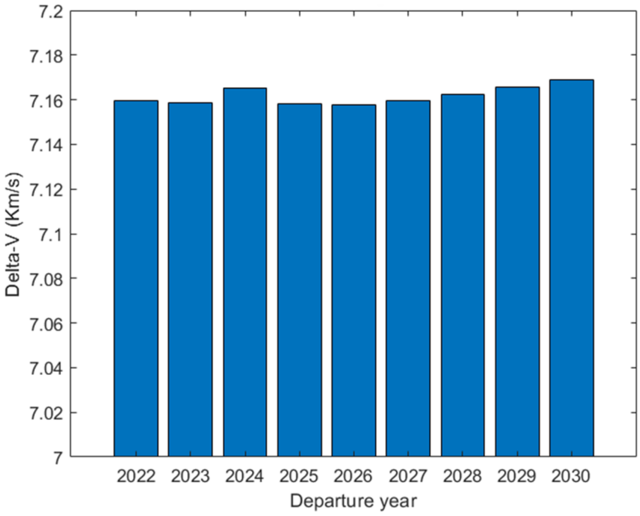

Figure 8. *Delta-V for TOF= 10.5 years.*

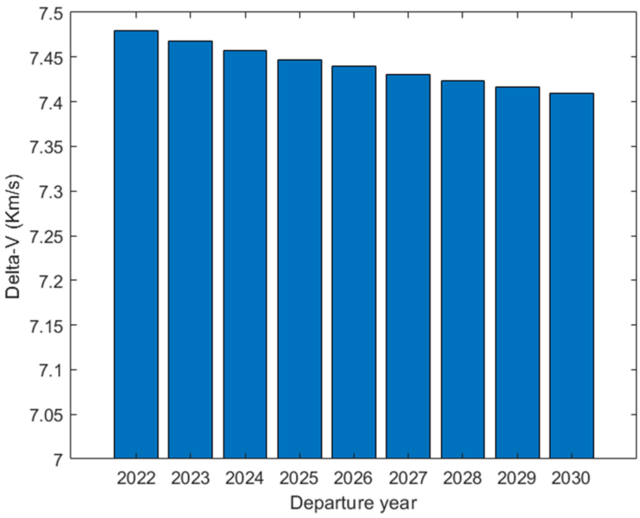

Figure 9. *Delta-V for TOF= 9.5 years.*

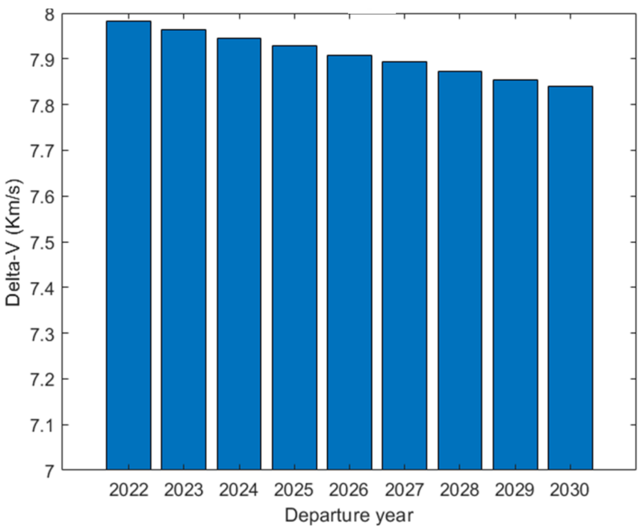

Figure 10. *Delta-V for TOF= 8.5 years.*

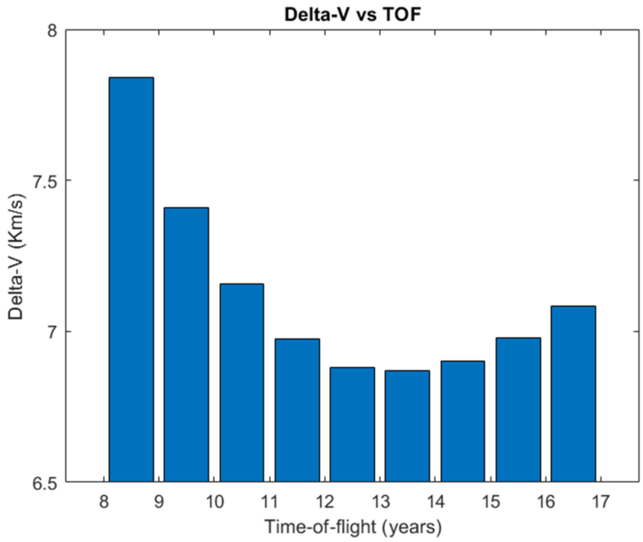

Figure 11. *Delta-V variation to TOF.*

3.2. Launch Mass for Various Time-of-Flight

The launch mass increases for the time of flight from 15.5 years to 11.5 years whereas it is constant for the time of flight of 10.5 years. The launch mass decreases for the time of flight of 9.5 years and 8.5 years. Since the launch mass of the spacecraft is less than 2000 kg, it can be launched with the present existing rockets such as GSLV, MK III, or the future most powerful rockets. Figures 12 to 19 show the variation of launch mass for the time of flight extending from 15.5 years to 8.5 years for the departure year time frame 2022 to 2030. Figure 20 shows the minimum launch mass for various time-of-flight ranges from 15.5 years to 8.5 years. It is observed to be the least for the flight times 13.5 years and 12.5 years.

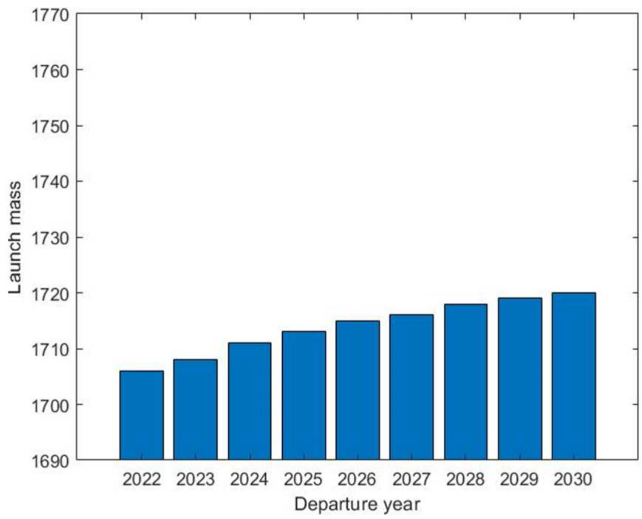

Figure 12. *Launch mass for TOF= 15.5 years.*

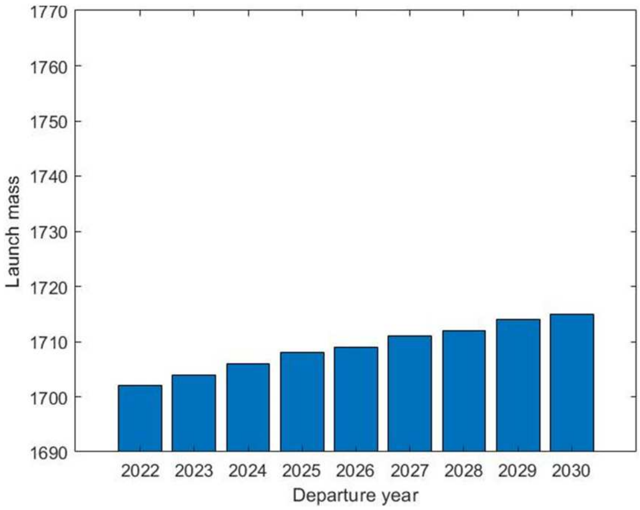

Figure 13. *Launch mass for TOF= 14.5 years.*

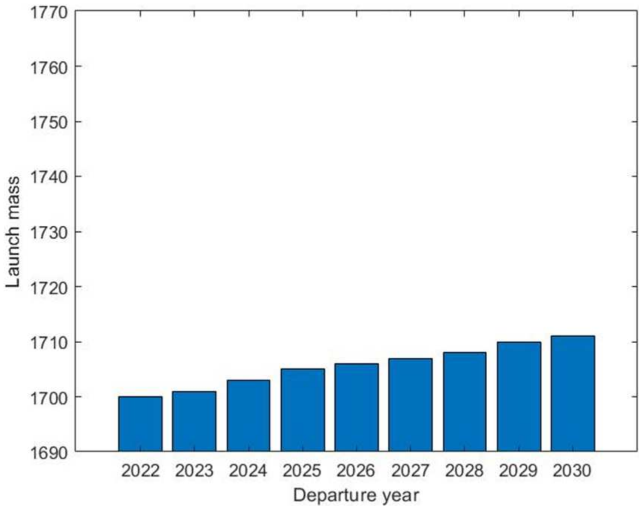

Figure 14. *Launch mass for TOF= 13.5 years.*

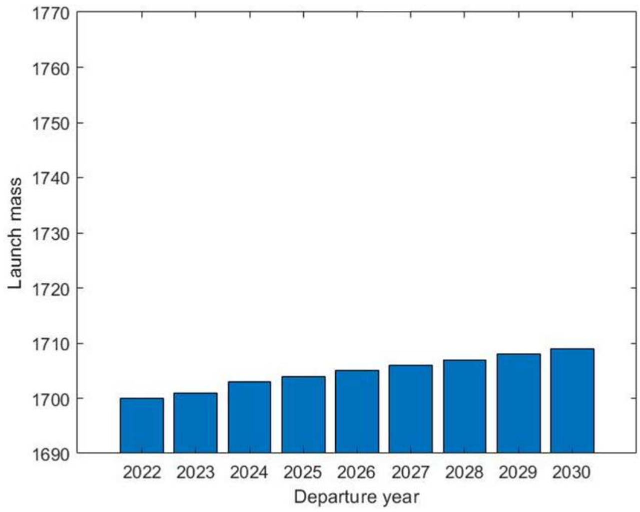

Figure 15. *Launch mass for TOF= 12.5 years.*

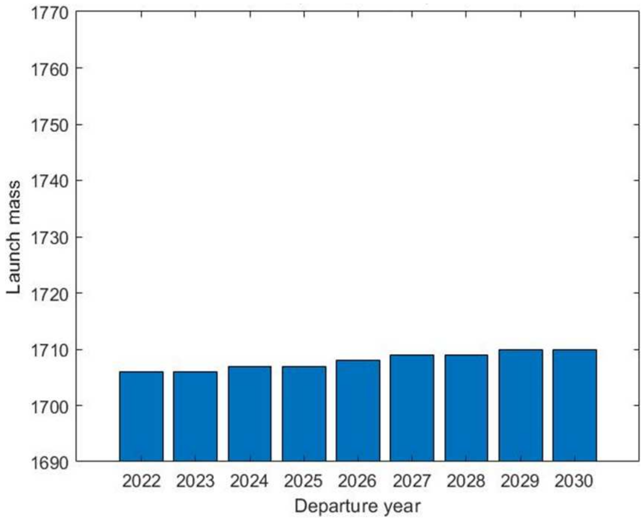

Figure 16. *Launch mass for TOF= 11.5 years.*

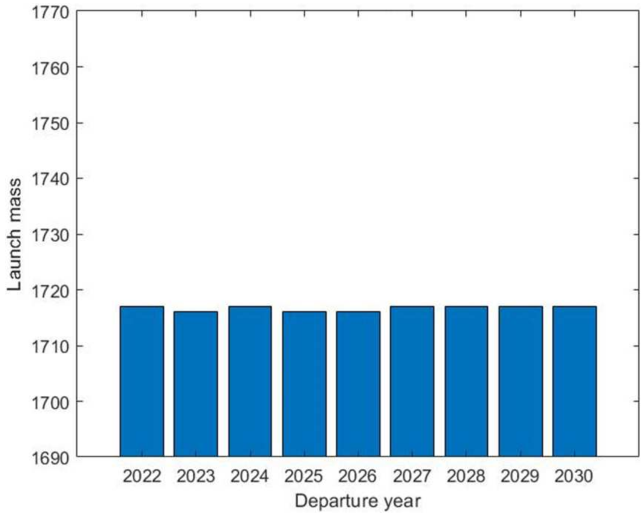

Figure 17. *Launch mass for TOF= 10.5 years.*

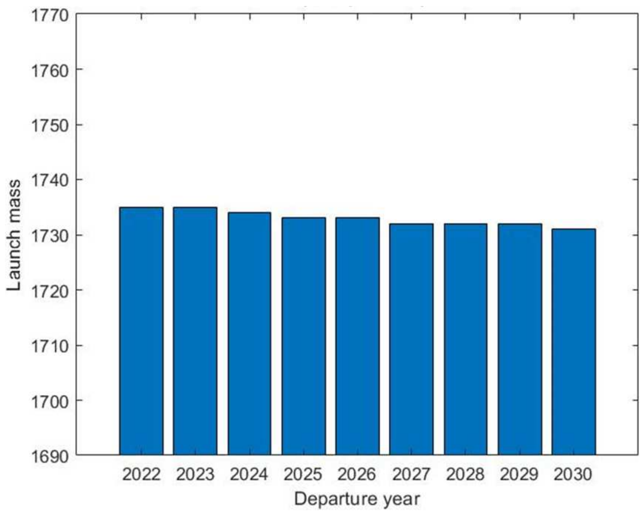

Figure 18. *Launch mass for TOF= 9.5 years.*

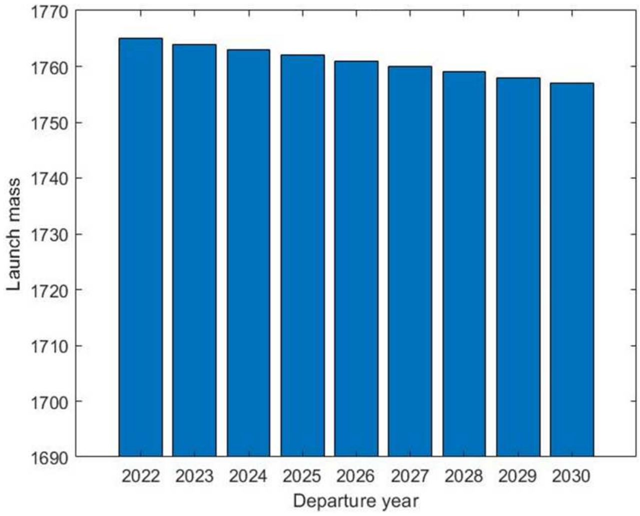

Figure 19. *Launch mass for TOF= 8.5 years.*

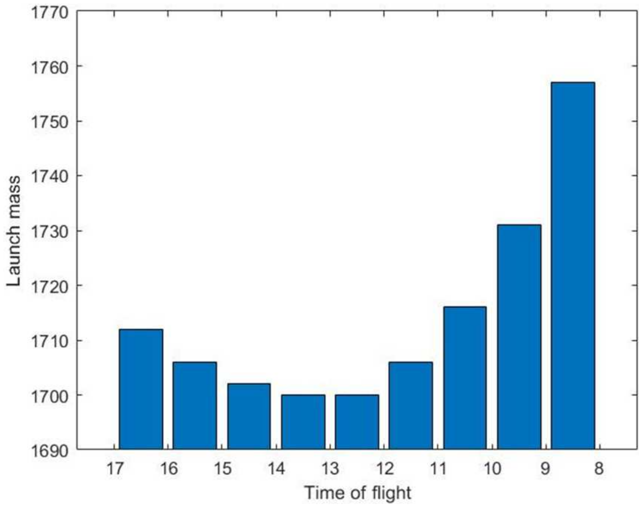

Figure 20. *Effect of time of flight on launch mass.*

For the direct transfer trajectory towards Uranus with the time of flight ranging from 15.5 – 8.5 years for the departure year 2022 – 2030, the launch mass required for a mission design has been determined. It has been noted that the propellant mass varies concerning the time of flight. The launch mass is the bare minimum, 1700 kg for the time of flight 13.5 and 12.5 years. The lowest launch mass for the time-of-flight of 14.5 years is 1702 kg. For the time-of-flight 15.5 and 11.5 years, the minimum launch mass is, 1706 kg whereas the lowest launch mass is 1715 kg for the time-of-flight of 10.5 years. 1730 kg is the minimum launch mass for 9.5 years. The value of the minimum launch mass is high, 1757 kg for the time-of-flight of 8.5 years. Apart from that, it can also be noted that for the time of flight from 15.5 years to 11.5 years, the launch mass is examined to be minimum for the departure year 2022. For the time-of-flight of 10.5 years, the least launch mass is observed for the departure year 2026. For the time of flight 9.5 years and 8.5 years, the slightest launch mass is observed for the departure year 2030.

4. Conclusion

For interplanetary missions, gravity assist techniques are frequently used. Due to their challenges and limitations in mission design an option for a direct transfer method is put forth. The focus of this paper is a direct transfer mission to Uranus. Preliminary estimation of spacecraft subsystems and payload is obtained from trade studies. The ceiling for the time-of-flight range is decided by performing a Hohmann transfer. Using Lambert's algorithm total delta-V for the mission is calculated for the various times of flight. The minimum value of delta-V obtained is 6.7 km/s for the mission. The optimum launch mass for this mission is determined for a departure year time frame of 2022-2030. For time-of-flight of 13.5 and 12.5 years, delta-V and the launch mass are observed to be the minimum. According to preliminary investigation, this mission could be a viable option for Uranus missions shortly.

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