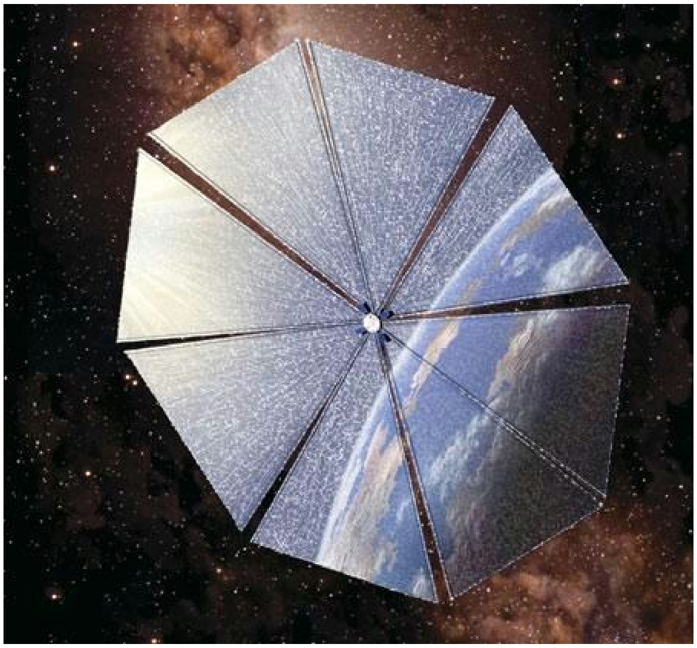


ZEPHYROS

[Odysseus 2016 Submission]

Hibah Mirza Chughtai
India

PRESENTATION AND JUSTIFICATION OF THE PROBLEM

Executive Summary

Each member of the team has observed recent reports of space launches and done a fact check on each, focusing on the aforementioned points of consideration. Via direct contact with sources or indirect through correspondence or reports, Each member feels that there is a need in the market for a system that fulfils the set targets by using some of the most inexpensive sources so that we can obtain samples directly from bodies such as the Galilean moons and other satellites of interest in a feasible time span and at a low cost, so that multiple missions for the same purpose are possible.

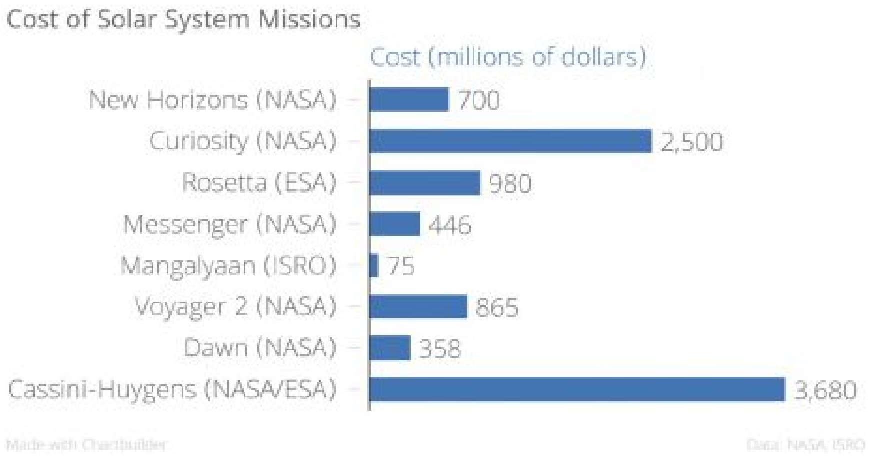
The crucial parts for a space-bound vehicle are -

1. Fuel
2. A Rocket
3. Probe/Shuttle/Orbiter/Payload

Out of the three, the current key area of concern is fuel. Most aerospace and aviation companies as well as government agencies are researching safer and more energy and cost efficient fuel in an effort to not only make space travel more economical and accessible, but also eradicate the long-standing safety issues and concerns that crop up whenever a launch goes wrong, or any related disaster occurs.

Our project's target group are government agencies and businesses in the aerospace segment aiming for trans-Jupiter missions. After a detailed discussion with Miss Heather Paul (mentor at the Asian Region Space Settlement Design Contest, an initiative by NSS board member Anita Gale) and Dr. Mason Peck, a professor at the Sibley School of Mechanical and Aerospace Engineering at Cornell University, the team learnt that there is a significant lack of cost-efficient fuel and how most experimental fuel prototypes were ruled out because of the numerous hazards they posed. The interview gave us a direct feedback from engineers directly involved in the field and a realistic idea of what the magnitude of the problem was and what specific points needed development.

An example would be for a mission going to Jupiter's moon Europa - the New Horizons missions has averaged a best of 13 months to get to Jupiter. In order to get back data and samples from Europa and have the scientists conduct detailed experimentation and analysis (in addition to those done by the sent probe), the best case scenario would be receiving samples 30 – 36 months from launch (taking in account all scientific experiments conducted by the probe. While the time span is not an issue here (the Galileo spacecraft had taken 6 years using Hohmann transfer orbits), it is the cost expended to complete a single maneuver like this – the Cassini-Huygens mission, which consisted of a lander and an orbiter, cost $3.68 bn; the New Horizons costed $700 mn (source: Forbes).



General Need For Solution –

Most current missions use LOX and LH2 as liquid bipropellants and ammonium chlorate as solid propellant, each of which has high combustibility, specific impulse, ignition temperatures and calorific values and low ignition times. However, fuels like UDMH (which are now currently used mainly in satellite launch vehicles) are extremely corrosive and become unstable with time.

Several current bipropellant mixes come under the category of hypergolic propellants – compounds which spontaneously ignite upon contact with each other. While hypergolic propellants are fairly stable and can be stored for long periods, they are extremely corrosive and, as a result, cannot be stored for too long on a spacecraft.
ROSCOSMOS (The Russian Space Agency) has been using the Proton rocket family since 1965, and continues to use its original fuel of Nitrogen Textroxide-Unsymmetrical Dimethyl Hydrazine (N2O4- UDMH) as its propellant. The fuel has an average burn time of 120 seconds and is relatively stable when compared to other hydrazine based fuels, but is extremely toxic.

"These propellants are carcinogenic, toxic, and highly reactive," said Mike Gruntman, a professor of Aerospace Engineering at the Univeristy of Southern California, commenting on the cause and subsequent contamination from the Proton-M explosion that took place in August 2013 (Source: The Verge). A 2008 analysis conducted by the NCBI on UDMH's environmental impact states that it is "persistent in the soil environment ... and remains in the environment for a significantly longer time than originally anticipated."

Following are the enivornmental impacts of 1,1 dimethylhydrazine (UDMH) (Source: USEPA) –

Acute Effects:

- Acute inhalation exposure of humans to 1,1-dimethylhydrazine has been observed to result in nose and throat irritation, mild conjunctivitis, nausea, and vomiting.
- 1,1-Dimethylhydrazine is highly corrosive and irritating to the skin, eyes, and mucous membranes, and neurological symptoms were observed in a man burned by 1,1-dimethylhydrazine.
- Central nervous system (CNS) stimulation and convulsions have been reported in animals acutely exposed to 1,1-dimethylhydrazine by ingestion.
- Acute animal exposure tests in rats, mice, hamsters, rabbits, and guinea pigs, have demonstrated 1,1-dimethylhydrazine to have high acute toxicity from inhalation, oral, and dermal exposures.

Chronic Effects (Non-cancer) -

- Liver damage in humans may occur from chronic (long-term) exposure to 1,1-dimethylhydrazine.
- Hemolytic anemia and CNS effects, such as convulsive seizures, have been observed in animals chronically exposed to 1,1-dimethylhydrazine by inhalation
- Respiratory and kidney effects have been observed in chronically exposed animals.

Cancer Risks -

- Rats exposed to 1,1-dimethylhydrazine by inhalation developed skin, lung, pancreas, pituitary, and liver tumors.
- IARC has classified 1,1-dimethylhydrazine as a Group 2B, the chemical is possibly carcinogenic to humans.

Apart from health issues, UDMH has also resulted in several disasters, the most famous being the Nedelin catastrophe, and recent ones being Virgin Galactic's SpaceShipTwo Explosion.

The SpaceShipTwo explosion that occurred on 2014, crashing in the Mojave Desert. The spacecraft exploded during its test run due to the experimental plastic- based fuel they were testing, killing one pilot and severely injuring the other.

LOX has been in use as an oxidiser since the mid-1960's, when NASA had begun flying Centaur and Saturn upper stages with it as the fuel. The liquid is stable, but mixtures with other propellants are shock sensitive and require cautious storage.

LH2 has been use since the 1950's and has been the main fuel in use for the past 20 years, but has certain drawbacks –

- Highly cryogenic (- – 423o F)
- Very low density, requiring larger amounts. This increases payload weight, which raises costs for storage, transportation and launch of the space vehicle.
- Improper storage of LH2 can result in catastrophes

Following is a list of space launch related disasters due to fuel-related problems -

- (December 6, 1957: Vanguard TV3) The United States' first attempt to launch a satellite into orbit was also its first failure. Two seconds after leaving the launch pad at Cape Canaveral, this rocket lost thrust and sank back down, rupturing and exploding its fuel tanks. It had reached a height of about four feet.

- A Proton-M rocket, carrying a Nimiq 6 communication satellite is raised to the launch pad at the Russian-leased Kazakhstan's Baikonur cosmodrome, on May 14, 2012. After the Russian Proton-M rocket lost control, tipped over and crashed in a fiery ball over the Kazakh desert on Tuesday, plumes of black and orange smoke could be seen hovering ominously over the crash site. The smoke was partly due to rocket fuel that escaped during the craft's plunge back to Earth. The plume comprised of unsymmetrical dimethylhydrazine and nitrogen tetroxide, in addition to kerosene. Because the rocket failure occurred only 17 seconds after liftoff, the fuel had little time to burn off, and the orange cloud seen hovering over the crash site was the distinctive color of nitrogen tetroxide. Its proximity to the ground made it one of the more dangerous Proton rocket explosions because of the hydrazine, which has an ammonia-like odor and is dangerously unstable unless handled in a solution.

- In 1975, three Apollo Soyuz astronauts were hospitalized for five days and treated for lung irritation after nitrogen tetroxide leaked from the steering jets into the cabin during reentry. Release of nitrogen tetroxide at a chemical plant in Louisiana in 1995 resulted in the evacuation of 3,000 people and 81 hospital admissions, but no deaths.
- The Nedelin catastrophe which occurred in 1960 is one of the most severe accidents to occur due to fuel. The second stage engines acidentally ignited, detonating the first-stage fuel tanks directly below, which destroyed the missile in an enormous explosion at the Baikonur Cosmodrome in the then USSR. Over a 100 personnel died in the incident, the highest fatality of any space-related disaster in human history.
- One of the more recent accidents has been that of the Virgin Galactic's SpaceShipTwo, which exploded during a test run midair, killing one pilot and severely injuring the other. The cause of the explosion was the instability of the new UDMH-based fuel the company was testing, which was not only a hypergolic fuel with a very short burn time but was also highly unstable.

Description & Estimation of Market Size

The market for space systems primarily has government agencies as its consumers followed by private organizations like Boeing, BAE, Northrop Grumman and SpaceX, with defense being the greatest share, followed by civil and commercial investments by space agencies. Quoted here is a paper by the OECD on the current state of the space economy. The table given below shows the investment in space by various countries –

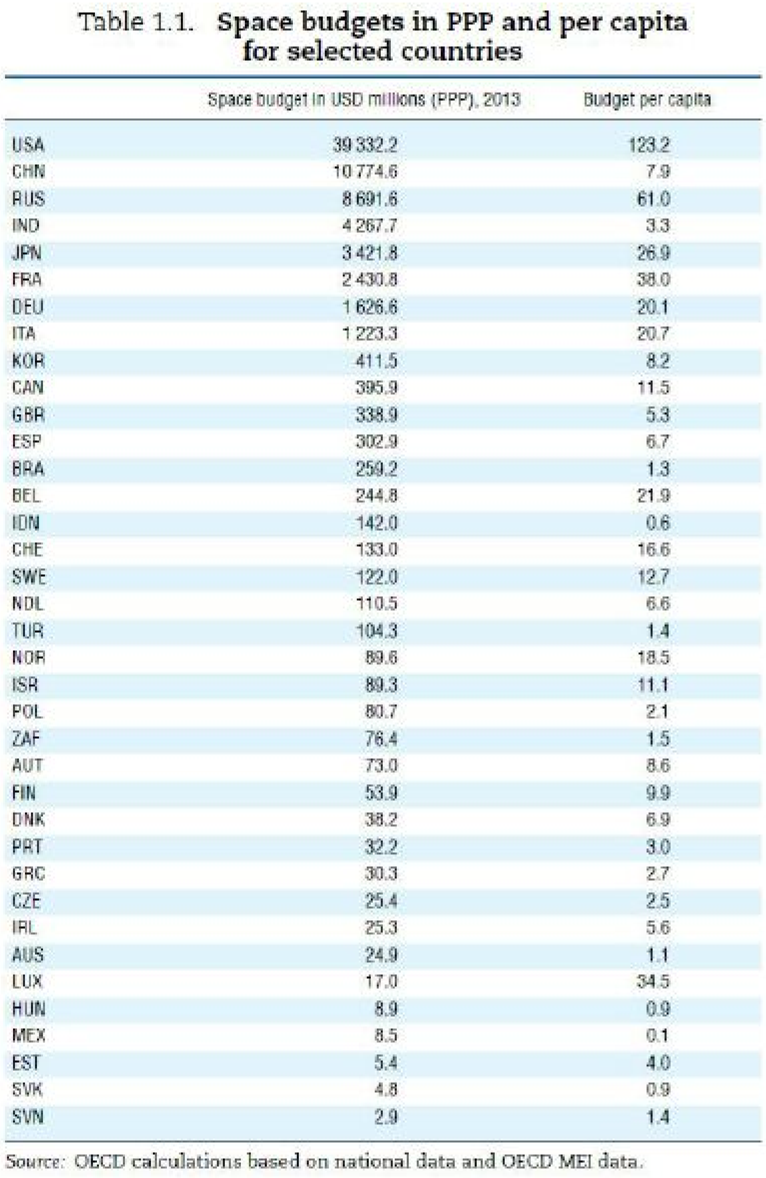


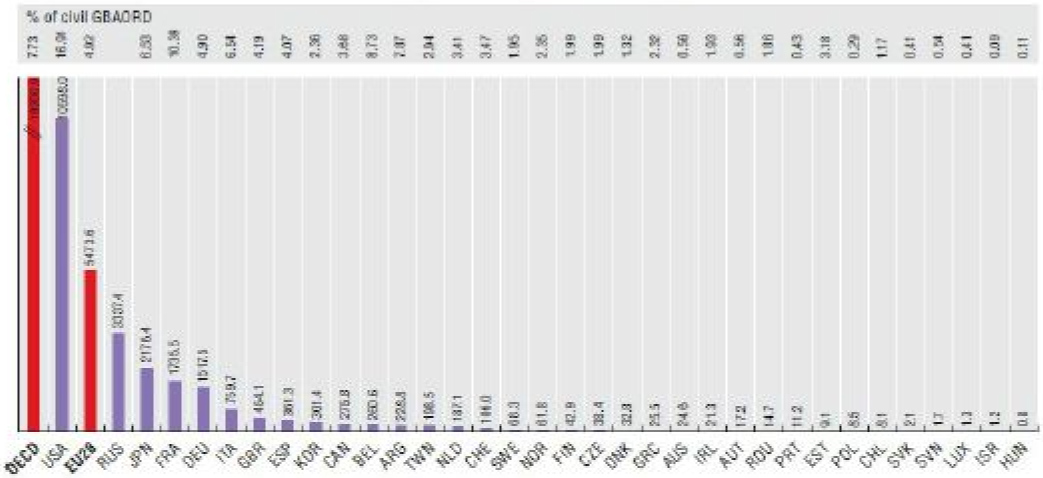
Table 1.1. **Space budgets in PPP and per capita for selected countries**

	Space budget in USD millions (PPP), 2013	Budget per capita
USA	39 332.2	123.2
CHN	10 774.6	7.9
RUS	8 691.6	61.0
IND	4 267.7	3.3
JPN	3 421.8	26.9
FRA	2 430.8	38.0
DEU	1 626.6	20.1
ITA	1 223.3	20.7
KOR	411.5	8.2
CAN	395.9	11.5
GBR	338.9	5.3
ESP	302.9	6.7
BRA	259.2	1.3
BEL	244.8	21.9
IDN	142.0	0.6
CHE	133.0	16.6
SWE	122.0	12.7
NDL	110.5	6.6
TUR	104.3	1.4
NOR	89.6	18.5
ISR	89.3	11.1
POL	80.7	2.1
ZAF	76.4	1.5
AUT	73.0	8.6
FIN	53.9	9.9
DNK	38.2	6.9
PRT	32.2	3.0
GRC	30.3	2.7
CZE	25.4	2.5
IRL	25.3	5.6
AUS	24.9	1.1
LUX	17.0	34.5
HUN	8.9	0.9
MEX	8.5	0.1
EST	5.4	4.0
SVK	4.8	0.9
SVN	2.9	1.4

Source: OECD calculations based on national data and OECD MEI data.

From the table given on the previous page, it can be reasonably inferred that the countries with the highest investment are mostly developed nations, with the USA, Russia, India, China, Japan and the European Union being the forerunners. Of all the countries in the top, India is the only developing country which makes the list. Even then, it has the lowest per capita investment in the space sector. Further insight into this issue is given by the following bar graph which shows the percentage of GDP invested by various countries.

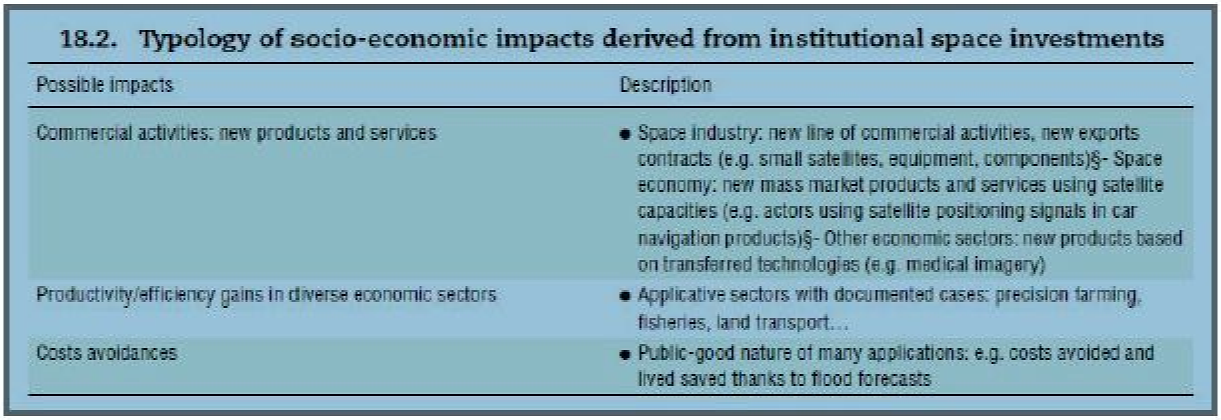
2.2. Civil space budgets in GBAORD, 2013



Source: *OECD Main Science and Technology Indicators Database.*

StatLink http://dx.doi.org/10.1787/888933141703

The above graph shows that most EU nations, USA, Russia and Japan are the only countries that invest a significant amount of their GDP into space research. All these countries are developed nations, and have most infrastructure and healthcare facilities in sufficient supply for their citizens. They have a stable economy, low rates of poverty and an overall higher level of development than other nations, which allows them to invest more in "secondary concerns" such as space. Socio-economic impacts of aerospace are also some of the few reasons why countries invest in it, as shown in the table below -



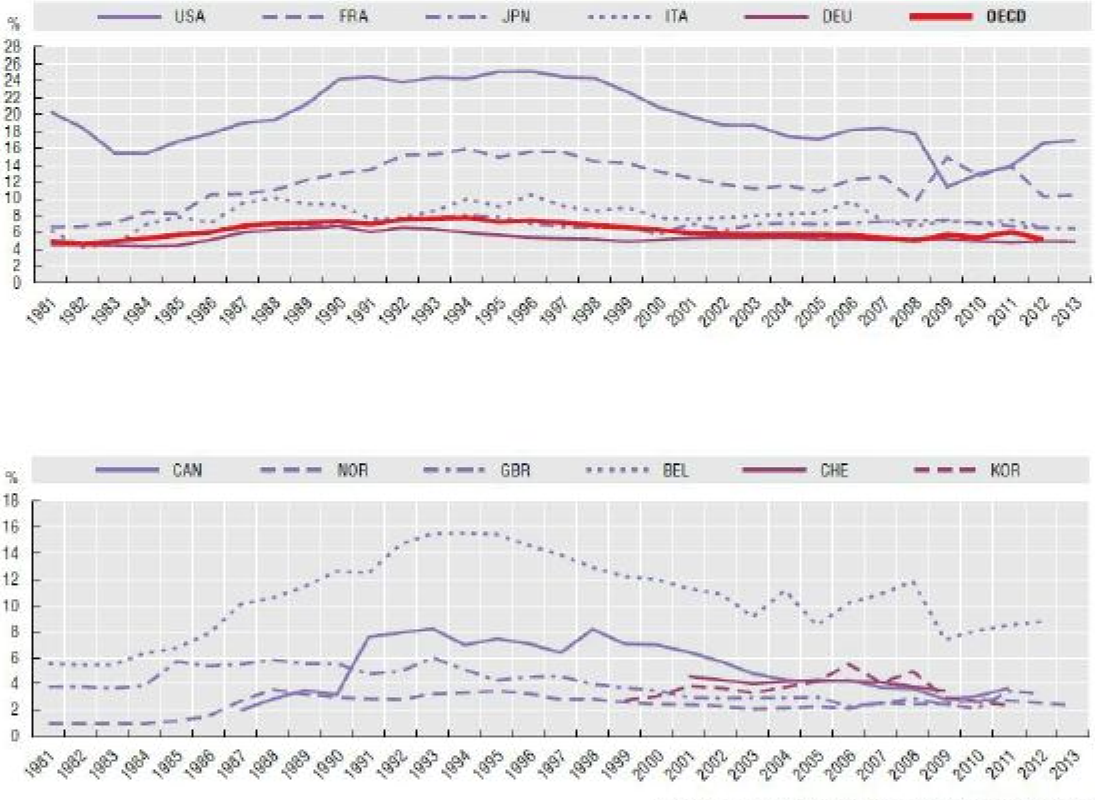
18.2. Typology of socio-economic impacts derived from institutional space investments

Possible impacts	Description
Commercial activities: new products and services	• Space industry: new line of commercial activities, new exports contracts (e.g. small satellites, equipment, components)§- Space economy: new mass market products and services using satellite capacities (e.g. actors using satellite positioning signals in car navigation products)§- Other economic sectors: new products based on transferred technologies (e.g. medical imagery)
Productivity/efficiency gains in diverse economic sectors	• Applicative sectors with documented cases: precision farming, fisheries, land transport…
Costs avoidances	• Public-good nature of many applications: e.g. costs avoided and lived saved thanks to flood forecasts

The line graph on the following page shows the change in civil space budgets over a 32 year period (1981-
2013), with 2 North American, 8 European and 2 Asian countries taken for comparison. It tracks how the changing global economic trends have affected investment by nations in the field of space.

2.1. Evolutions of civil space budgets in government budget appropriations or outlays for R&D (GBAORD) for selected countries, 1981-2013

As a % of GBAORD (or latest available year)



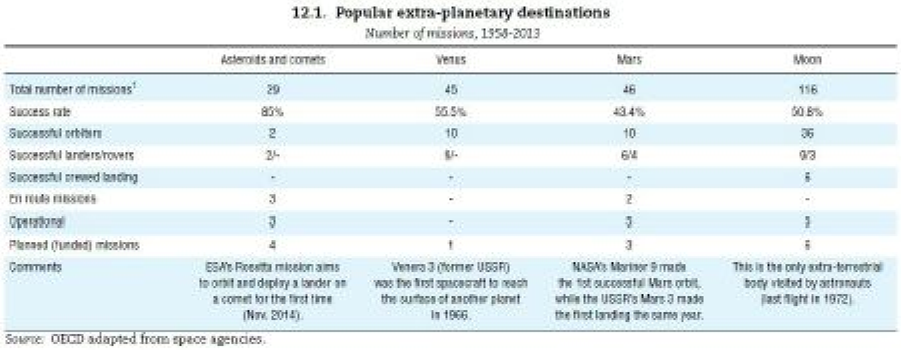
StatLink http://dx.doi.org/10.1787/888933141684

As shown by the graph above, an upward trend followed for the first half of the study, with investment in space reaching its peak by 1995. The following years showed a decline, with a sharp drop following the 2008 global recession. This shows that investment in space is closely linked to the global fiscal condition.

Also seen in the graph is the overall lower investment by smaller nations such as KOR, CHE and JPN. (Note
- smaller nations do not necessarily mean small economies, taking in account KOR and JPN).

Changes in the fiscal situation of a country drastically affect their investment in space, with most developed nations diverting most of their GDP towards healthcare, while developing countries opting to focus their GDP on health care, education and infrastructure rather than space. Hence cheaper and fuel efficient options for space missions become a key focus area.

The table on the following page shows the interest in extra-planetary missions and non-commercial space investment by evaluating the total number of missions and the uses they have been deployed for (i.e. rovers/landers, probes/orbiters and their success rate, as well as future plans for interplanetary missions. It is important to have this data since it gives an accurate idea of what the current market demand for the product, since the solution aims at long distance-deep space, inter-planetary and extra-planetary missions. By having information on the current trends in space exploration, we will have an accurate idea of the market size, demand and market requirements, which the product must meet.



12.1. Popular extra-planetary destinations

Number of missions, 1958-2013

	Asteroids and comets	Venus	Mars	Moon
Total number of missions[1]	29	45	46	116
Success rate	85%	55.5%	43.4%	50.8%
Successful orbiters	2	10	10	36
Successful landers/rovers	2/-	8/-	6/4	9/3
Successful crewed landing	-	-	-	6
En route missions	3	-	2	-
Operational	3	-	3	3
Planned (funded) missions	4	1	3	6
Comments	ESA's Rosetta mission aims to orbit and deploy a lander on a comet for the first time (Nov. 2014).	Venera 3 (former USSR) was the first spacecraft to reach the surface of another planet in 1966.	NASA's Mariner 9 made the 1st successful Mars orbit, while the USSR's Mars 3 made the first landing the same year.	This is the only extra-terrestrial body visited by astronauts (last flight in 1972).

Source: OECD adapted from space agencies.

Problem Statement –

The key considerations for space missions are –

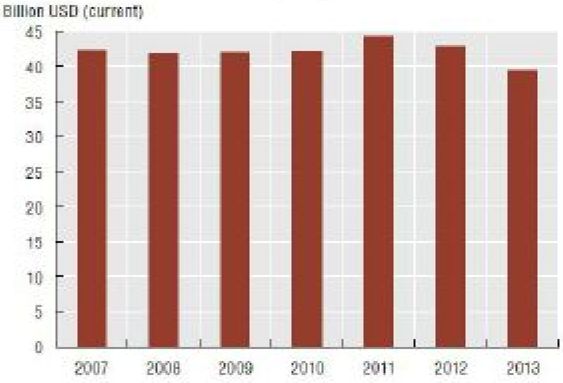
- Economic viability
- Reduction of space-junk and usage of hazardous compounds/substances as fuel/parts

Economic Viability –

Economic viability is a deciding factor for any space-related endeavor, and global economic conditions heavily affect the chances of that mission/project's consideration. Given below are graphs that show recent economic trends in major space-investing countries, and how these trends have effected their spending on aerospace.

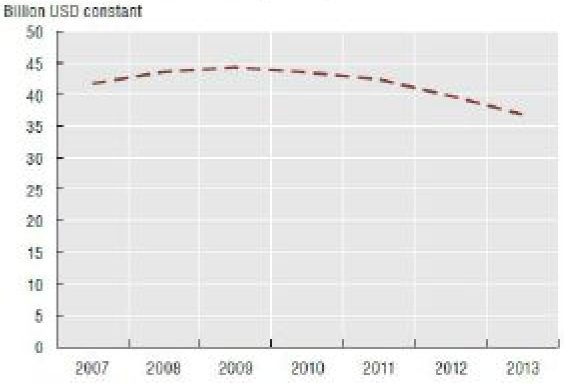
36.1. US space budget estimates

In billion USD (current), 2007-13



36.2. US inflation-adjusted space budget

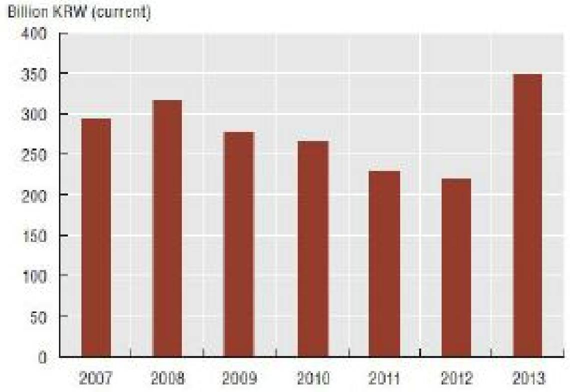
In billion USD (constant), 2007-13



Source: OECD calculations based on relevant space-related budgets from NASA, Department of Defense, Department of Commerce (NOAA), Department of Transportation (Federal Aviation Administration), Department of the Interior (US Geological Survey), and OECD consumer prices (all items), extracted from MEI database, June 2014.

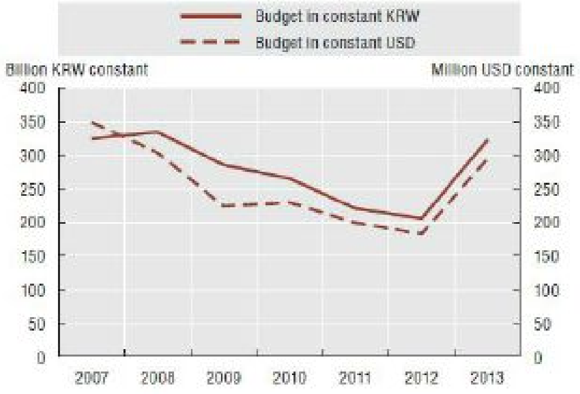
32.1. Korea's space budget

In billion KRW, 2007-13



32.2. Korea's inflation-adjusted space budget

In constant billion KRW and million USD, 2007-13



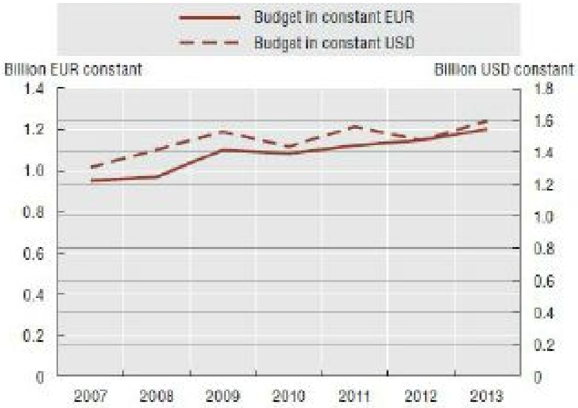
Source: Korean Ministry of Education, Science and Technology, 2014.

29.1. Germany's space budget

In million EUR (current), 2007-13

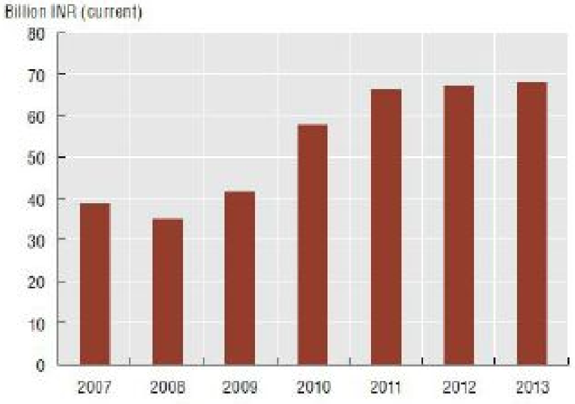
29.2. Germany's inflation-adjusted space budget

In billion EUR and USD (constant), 2007-13



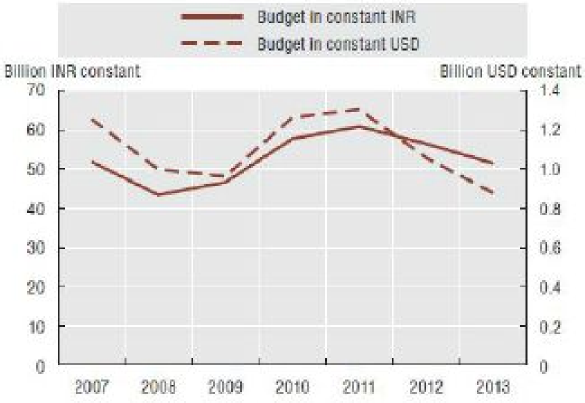
30.1. Indian space budget

In INR billion (current), 2007-13



30.2. Indian inflation-adjusted space budget

Constant INR and USD billion, 2007-13



The above given graphs all show a drop in from the 2008-2009 period, followed by a steady decrease in the case of the US and India.

As NASA's core technology program for all space transportation, the Advanced Space Transportation Program at the Marshall Center is pushing technologies that will dramatically increase the safety and reliability and reduce the cost of space transportation. Today, it costs $10,000 to put a pound of payload in Earth orbit. NASA's goal is to reduce the cost of getting to space to hundreds of dollars per pound within 25 years and tens of dollars per pound within 40 years.

The high cost of space transportation coupled with unreliability and dangers of space contamination through hazardous fuels is a virtual padlock on the final frontier. But, imagine the possibilities when space transportation becomes safe and affordable for ordinary people. Whether it's living and working in space, exploring new worlds or just leaving the planet for vacation, the opportunities for business and pleasure on the space frontier are endless.

This project is inspired by the dream of everyday life in space and named **"Zephyros - Greek God of West Wind"** which also symbolizes "Solar Sail movement in the solar system"

The solution being put forward is a combination of a self-repairing solar sail and a photoelectrolytic engine for safer and cheaper propulsion for space exploration. The engine utilises water as its primary fuel. The sail is powered by the Sun. The key change made to the sail is a self-repair feature, which involves a layer of vacuum between the surface and the repair polymer for the sail. Upon damage, the vacuum seal will break, with the polymer instantaneously repairing the damaged portion.

The self-repair feature is not currently seen in solar sails but is a big plus point. Auto-repair allows the sail to repair itself in case of a breach, by detecting breakage in the vacuum seal. When the seal is broken, the repair CP-1, stored in liquid form, will flow into the holes - the resin solidifying and sealing the breach keeping the sail fully operational.

ANALYSIS OF PRIOR SOLUTION ATTEMPTS

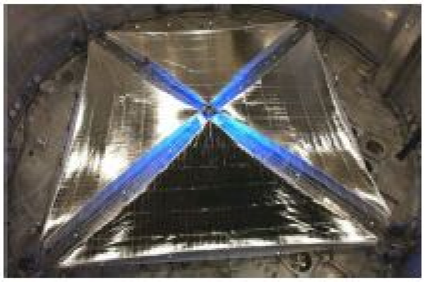
Benchmarking of Competitive Products

Part 1 – The Solar Sail

Three different types of sail material were analyzed and benchmarked against each other in order to establish a baseline for material performance. The following materials were chosen for comparison –

1. DuPont™Kapton HN
2. TAP Plastics' Mylar Film
3. Nexolve Materials' LaRC CP-1 Polyimide

DuPont™Kapton HN



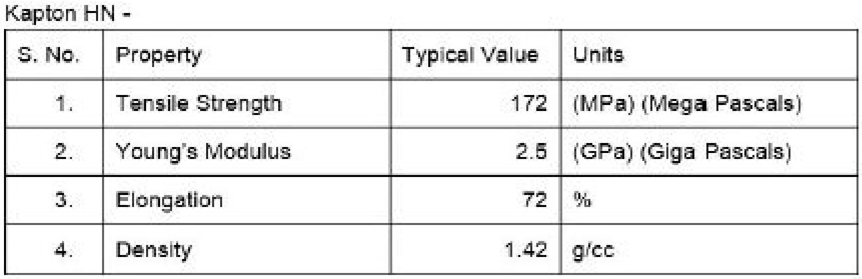
LaRC CP-1 Polyimide – Manufactured by ATK for NASA's ST9 mission.

TAP Plastics' Mylar

Objective Benchmarking Data –

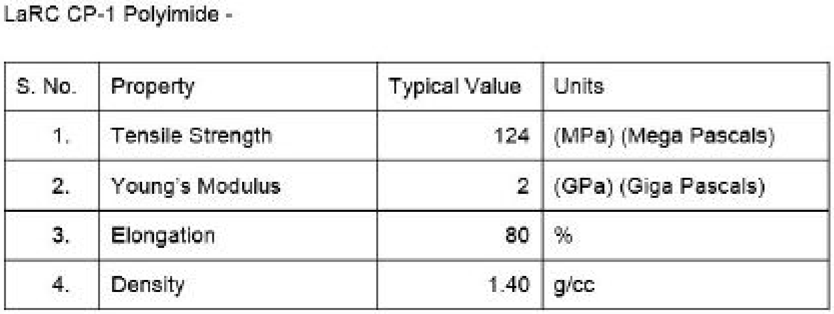
The given materials were then rigorously tested on the following criteria –
1. Tensile Strength
2. Young's Modulus
3. Density
4. Elongation

The tensile strength of the material is the capacity to which it can elongate without breaking or the maximum stress a material can endure before it is pulled or broken apart by it. The Young's Modulus is the ratio of the stress to the strain a material can endure before it reaches its breaking point. Density is the mass per unit volume of a substance and is a critical aspect to determine the material used in solar sail building. The elongation of the material can be defined as the measure of the ductility of a materials – in other words it is the amount of strain it can experience before failure in tensile testing.



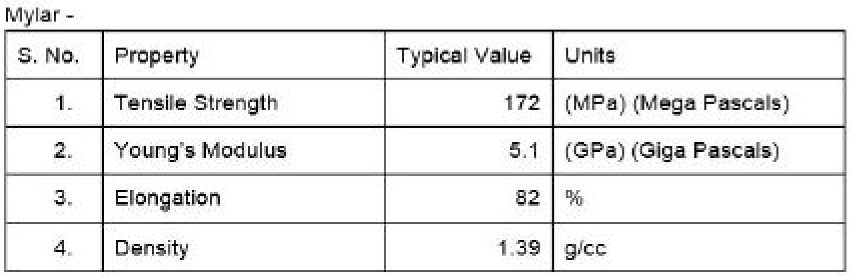
Kapton HN -

S. No.	Property	Typical Value	Units
1.	Tensile Strength	172	(MPa) (Mega Pascals)
2.	Young's Modulus	2.5	(GPa) (Giga Pascals)
3.	Elongation	72	%
4.	Density	1.42	g/cc



LaRC CP-1 Polyimide -

S. No.	Property	Typical Value	Units
1.	Tensile Strength	124	(MPa) (Mega Pascals)
2.	Young's Modulus	2	(GPa) (Giga Pascals)
3.	Elongation	80	%
4.	Density	1.40	g/cc



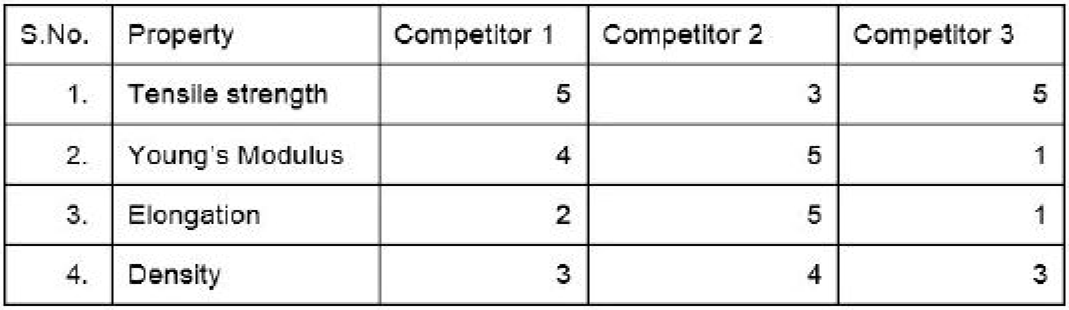
Mylar -

S. No.	Property	Typical Value	Units
1.	Tensile Strength	172	(MPa) (Mega Pascals)
2.	Young's Modulus	5.1	(GPa) (Giga Pascals)
3.	Elongation	82	%
4.	Density	1.39	g/cc

(All values are taken at 23oC)

Table II.A.IV ranks the three competitors on a scale of 1 to 5, with one being the least optimal to five being the most optimal value for each property used for benchmarking.
A material with a higher tensile strength is given a higher ranking because of its ability to endure higher stress as compared to other materials. A material with a lower Young's Modulus is given a higher ranking because of its ability to withstand higher elongation before breaking. A material with lower density is given a higher rating due to its lower weight for the same surface area employed. A material with higher elongation percentage is given a higher rating as it can withstand higher stress until it reaches its breaking point.
Lower Young's Modulus and higher tensile strength are important to the durability and longevity of the space sail and are a crucial component in determining sail material, while lower density would ensure a higher surface area for lower weight.

Comparison rankings on a scale of 1 to 5



S.No.	Property	Competitor 1	Competitor 2	Competitor 3
1.	Tensile strength	5	3	5
2.	Young's Modulus	4	5	1
3.	Elongation	2	5	1
4.	Density	3	4	3

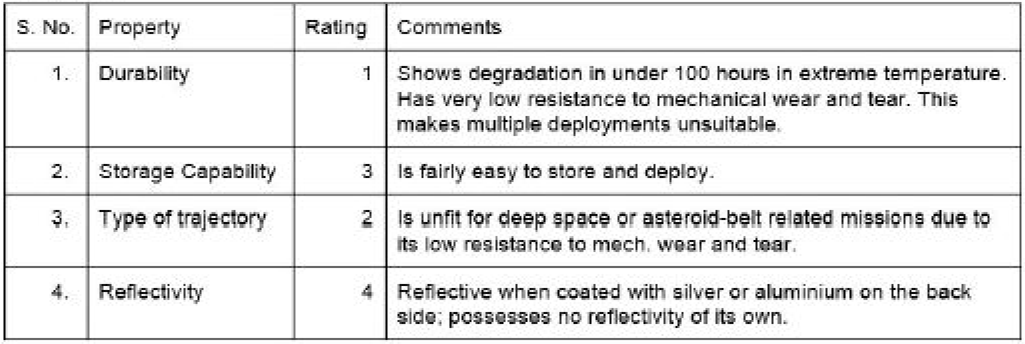
Subjective Benchmarking Data

Subjective benchmarking data ranks the parameters according to user-defined requirements, i.e. it compares how optimal the values of each given property are for the given problem.
For example, a low Young's Modulus is preferable due to amount of stress endurance, but an extremely low Young's Modulus may hamper the sail's functioning.
Also, the cost of the material, its availability and feasibility in given usage conditions is also important.

For an example, a material with superior tensile strength may not have a low enough ductile-to-brittle transition temperature, rendering it useless for deep space conditions.

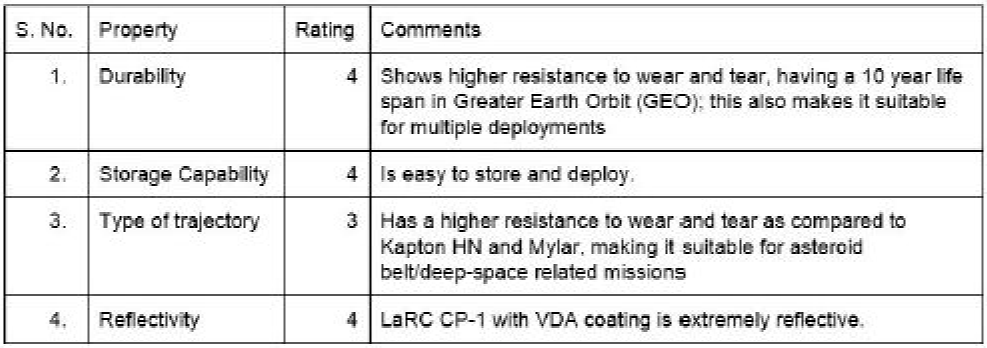
The parameters used for the subjective benchmarking data are as follows:
1. Durability
2. Storage Capability
3. Type of Trajectory
4. Payload

Kapton HN -



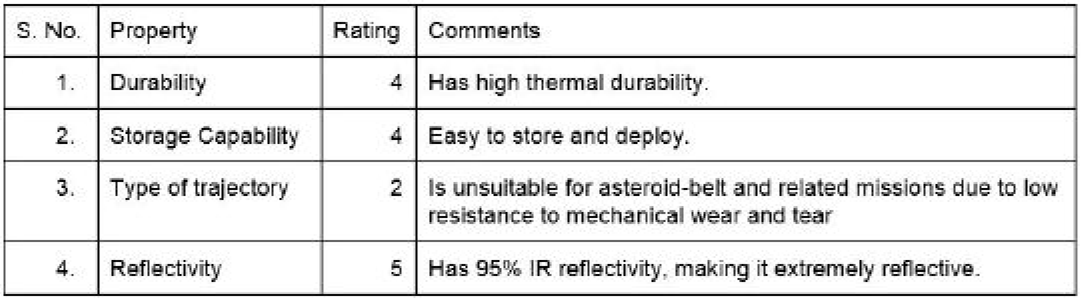
S. No.	Property	Rating	Comments
1.	Durability	1	Shows degradation in under 100 hours in extreme temperature. Has very low resistance to mechanical wear and tear. This makes multiple deployments unsuitable.
2.	Storage Capability	3	Is fairly easy to store and deploy.
3.	Type of trajectory	2	Is unfit for deep space or asteroid-belt related missions due to its low resistance to mech. wear and tear.
4.	Reflectivity	4	Reflective when coated with silver or aluminium on the back side; possesses no reflectivity of its own.

LaRC CP1 Polyimide



S. No.	Property	Rating	Comments
1.	Durability	4	Shows higher resistance to wear and tear, having a 10 year life span in Greater Earth Orbit (GEO); this also makes it suitable for multiple deployments
2.	Storage Capability	4	Is easy to store and deploy.
3.	Type of trajectory	3	Has a higher resistance to wear and tear as compared to Kapton HN and Mylar, making it suitable for asteroid belt/deep-space related missions
4.	Reflectivity	4	LaRC CP-1 with VDA coating is extremely reflective.

Mylar



S. No.	Property	Rating	Comments
1.	Durability	4	Has high thermal durability.
2.	Storage Capability	4	Easy to store and deploy.
3.	Type of trajectory	2	Is unsuitable for asteroid-belt and related missions due to low resistance to mechanical wear and tear
4.	Reflectivity	5	Has 95% IR reflectivity, making it extremely reflective.

Summary of Results

1. Tensile strength – Objectively, the sail material must have a value between 80 to 234 Megapascals (MPa). A value lower than this can cause the sail material to break during deployment or in the case of any collisions.
2. Young's Modulus – Objectively, the sail materials must have a Young's Modulus between 2 to 5. A higher Young's Modulus would render the sail too inflexible for proper utilization of the radiation pressure.
3. Elongation – Objectively, the sail material must possess an elongation percentage in the range of 15% to 92%
4. Density – Objectively, the sail material's density must lie between 1.39 to 1.54. A material with higher density will add to the weight of the structure – however, a material with lower density will be unsuitable for sail making material.
5. Durability – Subjectively, the sail must possess high thermal durability for deep space missions and a low degradation rate with a high mean time between failures (MTBF) in order for it to have the longest time functioning without any repair.
6. Storage Capability – Subjectively, the material must be easy to store and deploy, with no hazards posed by thermal or radiation-related changes in environment.
7. Type of trajectory – The sail material must be able to sustain and efficiently navigate all trajectories and space conditions such as asteroid belts and icy objects regions like the Oort Cloud.

8. Reflectivity – must possess high reflectivity to ensure maximum thrust provided by radiation.

Patent Search

US6565044B1 – Combination solar sail and electrodynamic tether propulsion system (Details have been directly taken from above-mentioned patent).
Inventor(s) – Charles L. Johnson, Gregory L. Matloff

Background of the Invention –
The present invention relates generally to systems and methods for traveling through space. More particularly, this invention pertains to a system and method for traveling through space using solar sails and electrodynamic tethers.
Solar sail propulsion systems typically include a large, flat, thin, reflective material supported by a lightweight deployable structure. These systems generate propulsion by transferring the momentum of solar photons reflected by the solar sail to the sail. In other words, propulsion is generated when sunlight strikes and is reflected off of the solar sail. Solar sail propulsion systems have several advantages.

First, such systems do not require the use of propellants to generate propulsion. As a result, a spacecraft can be made lighter or, alternatively, the spacecraft payload carrying capacity can be increased. Second, systems using solar sails to generate propulsion can reach speeds that are much faster than speeds reached by conventional propellant-type propulsion systems. As a result, solar sail propulsion systems can reach planets far away from the Earth in a much shorter time period. Solar sails rely on the sun, and the sunlight that it produces, for propulsion and cannot generate sufficient thrust when the solar sail moves too far away from the sun, typically approximately 5
Astronomical Units (AU). As a result, solar sails cannot be used to steer a spacecraft into orbit or to perform orbital maneuvers around a planet, such as Jupiter or Saturn, which is very far away from the sun.

Summary Of The Invention –
Accordingly, one object of the present invention is to provide an improved solar sail propulsion system that provides sufficient propulsion to steer a spacecraft into orbit and to perform orbital maneuvers around planets that are so far away from the sun that the solar sail is no longer effective.

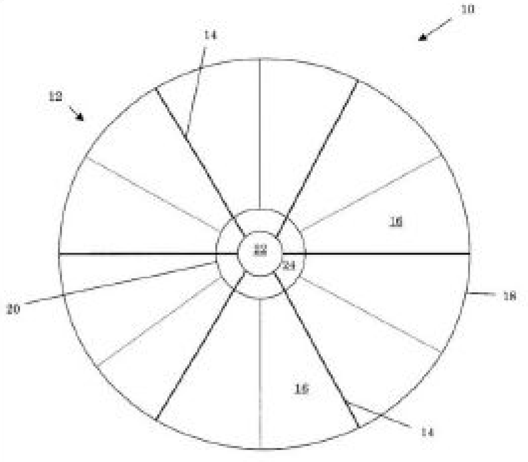


Fig. 1

Brief Description of the Drawings -
FIG. 1 is a front view of one embodiment of the combination solar sail and electrodynamic tether propulsion system of the present invention.

Description of Preferred Embodiments –

FIG. 1: one embodiment of the combination solar sail and electrodynamic tether propulsion system 10 of the present invention includes a spin-stabilized solar sail 12 and multiple electrodynamic tethers 14 embedded in the sail 12. The solar sail 12 includes six (6) pie-shaped gore segments 16, each segment 16 having an outer edge 18 and an inner edge 20. The segments 16 are connected together along the outer edges 18 using connecting tethers (not shown) to form a circular shape and the inner edges 20 are connected to a sail cylinder 22 using connecting tethers (not shown) to form a central opening 24.

The electrodynamic tethers 14 are connected on one end to the sail cylinder 22 and extend radially outward to the outer edges 18. In one embodiment, the solar sail 12 has a diameter of approximately 410 meters and the electrodynamic tethers 14 are approximately 200 meters long.

Strengths –
The connecting tethers and sail cylinder ensure quick and effortless deployment and offer a steady skeletal structure. The pie-like structure of the sail also aids in retraction and deployment.

Weaknesses –
The physical structure and shape of the sail doesn't offer any other advantages other than the ones listed above. The electrodynamic thrusters are irrelevant to the solution as the system does not operate solely on solar sail propulsion alone.

US 4614319 A – Solar sail
Inventor(s) – Kim E. Drexler
Disclosure of the Invention –

The present invention is a solar sail propulsion system comprising: a sail for intercepting light pressure for producing thrust; a tension truss having two ends attached at one end to the sail for transferring the thrust from the sail and for preventing deformation of the sail under light pressure; and a payload attached to the other end of the tension truss. The solar sail is a thin metal film. The tension truss attached to the other end of the tension truss. The solar sail is a thin metal film. The tension truss includes a plurality of attachment sections for connecting shroud lines to the top of the tension truss, and a plurality of shroud lines attached to the attachment section at one of their ends and to the payload at the other end. A plurality of reels are attached to the shroud lines for controllably varying the length of the lines. A plurality of reflective panels are attached to the sail for controlling the orientation of the system.

The system further includes light-weight compression sections for preventing the collapse of the tension truss in the absence of light pressure acting on the sail. The system still further includes two-dimensional stress relief members for reducing stress across the sail. The sail is a disc attached to a truss built of tension structures with catenary-edged panels installed in apertures in the disc structures. The sail is made up of several layers of different materials.

The present invention is also a method of making a thin film comprising: coating a relatively thick surface with a volatile substance, depositing a film on the volatile substance, and volatilizing the substance to a vapor whereby the film is released from the surface. This method is performed in a microgravity environment. In this method the thick surface is a belt or drum which may be textured. The film is deposited by vapor deposition, sputtering, or chemical deposition and may comprise more than one layer. The volatilizable substance is a sublimable solid or an evaporable liquid. This method further includes the terminal step of recovering the vapor for reuse.

Sail construction –
The strategy for near-term sail construction is to make and assemble as much of the sail as possible on earth. Thus, while the delicate films of the sail must be made in space, all other compenents are made on earth. The sail construction system consists of the following elements: a scaffolding (to control the structure's deployment), the film fabrication device (to be described hereinafter), a panel assembly device, and a "crane" for conveying panels to the installation sites.

The sail's structure consists of a regular grid of tension members, springs, and dampers, and a less regular three-dimensional network of rigging. This is a very complex object to assemble in space. Fortunately, even the structure for a sail much larger than described herein can be deposited in the Shuttle payload bay in deployable form.
The structure will be deployed by pulling on cords attached to certain nodes. Deployment may be controlled by a friction brake in the hubs of the reels. By setting the brakes properly, positive tension must be applied for deployment and certain members may be made to deploy before others. Further control of the deployment sequence, if needed, may be introduced by a mechanism which prevents some elements from beginning to deploy until selected adjacent elements have finished deploying. If detailed external intervention is deemed desirable, brakes could be rigged to release when a wire on the housing is severed by laser pulse.

Solar Sail Dynamics and Control –
There are essentially two modes for operation and control of the solar sail. In the first mode, the tilting of panels produces control forces. The panels 200 may be tilted as shown in FIG. 4. Small electric motors reel or unreel lines 101 and 204 on command, so as to move the panel corner

206 up or down while maintaining proper tension on it. Each panel has a mass of some 0.3 to 1.1 kilograms.

This first mode is conceived of as a semi-passive control mode for interplanetary cruising (where only slow changes of attitude are needed). It is of importance to consider the stability of a passive sail set at various angles to the sun. In the ideal sail approximation (planar, perfectly reflecting), thrust will be normal to the sail and act through its center of area, that is, along the axis of symmetry. In an absorbing sail, its thrust is divided into purely reflective and purely absorptive components. The former produces no torque, while the latter produces a torque. To counter this torque, light pressure must be increased on the far side of the sail from the sun relative to that on the near side. Making the sail concave toward the payload accomplishes this purpose.

In the second mode of sail configuration, the payload mass is assumed to be large compared to the sail mass, and the sail is considered as a separate object linked to it by actively controlled shroud lines 202 and 204. In the second mode, the tilting of the panels 200 controls the spin rate. However, in this mode precession is effected by varying the tension exerted by the shrouds 202 and 204 on different parts of the sail. This is accomplished by reeling and unreeling the shrouds in a coordinated fashion as the sail turns.

For the sail discussed above, and the probable range of sail performances, this arrangement implies precession rates of 13 to 26 rad/100 minutes, when the sail is flat with respect to the sun. This provides a generous margin in turn rate, even from maneuvers in low earth orbits. This active control permits damping of nutation. This is important, since nutation would otherwise be initiated by rapid changes in precession rate.

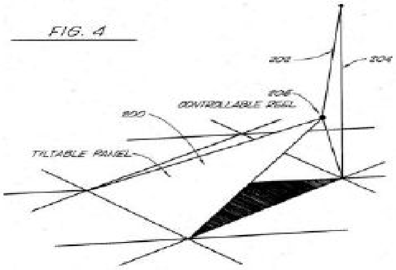


Figure 4: Front view of the sail about its axis of symmetry. The sketch depicts the controllable reel and the tilting of the sail's panels that enable it to ensure maximum reflectance of sunlight to obtain maximum radiation pressure as thrust.

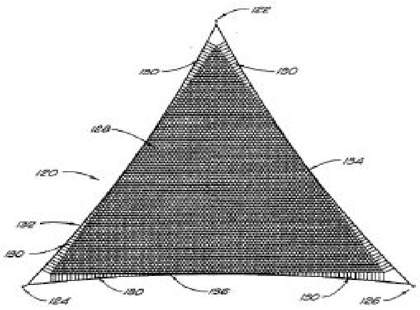


Figure 1: Front view of the side elevation of the triangular reflecting panel

For missions involving both interplanetary cruise and circumplanetary maneuvering, a vehicle able to operate in both modes is desirable. The first mode has a decisive advantage near planets (because of its maneuverability), but cannot enter a passive cruise mode. The greater distance between the payload and sail in this mode precludes balancing the torque on the sail resulting from absorbed light with a reasonable amount of concavity, as is done in the first mode.

Strengths –
The invention provides a method for producing multi-layered aluminium sails in space and puts forward a unqiue design of a sail comprising of multiple triangular panels which can be individually adjusted to gain maximum sunlight for receiving maximum radiation pressure. The two modes of control and operation of the sail endure smooth movement and receiving maximum thrust from radiation in both interplanetary and deep-space sailing.

Weaknesses –
The triangular panels of the sail are formed by slow vapor deposition in space, a process that is not only time consuming but also infeasible for a solar sail design that has a self-repair function, which involves a structure filled with the repair resin, separated from the main sail by a thin layer of vacuum. With a design that involves formation of sail by vapor deposition, the repair resin idea is structurally incompatible.

Lessons Learnt –
The following conclusions have been made from the benchmarking, summary analysis and patent searches conducted –

1. The sail structure must be made on Earth, i.e. vapor deposition is an incompatible feature for a sail which has a self-repair function.
2. It is recommended to use a circular design for easier deployment and retraction, because circular structure are stronger at handling stresses when compared to triangles.
3. A pie-structured circular sail (i.e. a circular sail divided in 16 pie-shaped sections) is the most optimum design.
4. Adjustable panels are a desirable feature.
5. No design speaks of an auto/self-repair feature.

Opportunities for Competitive Advantage –

The market lacks a sail design that is capable of automatic repair upon damage, which is the key point of innovation in the project. A sail with the above mentioned feature will last longer and have a higher mean time between failures (MTBF).

Benchmarking of Competitive Products (Engine)

Three different types of engines were benchmarked against each other to establish a baseline for engine performance. Each engine was assessed on the following criteria –

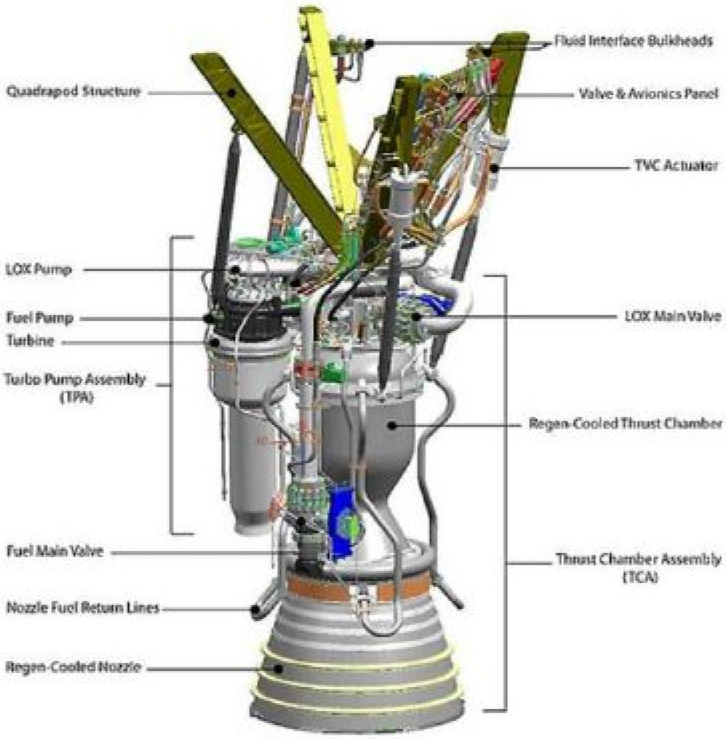
1. Propulsion System
2. Propellant
3. Specific Impulse
4. Exhaust Velocity
5. Thrust
6. Thermal Efficiency

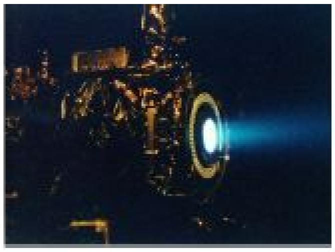
The propulsion system of the engine determines what kind of an engine it is (i.e. ion thruster, liquid fuel engines, solid-fuel engines, etc.). The propellant is the fuel employed for that particular propulsion system. The specific impulse of the engine is the impulse per unit of propellant consumed and is a measure of efficiency of jet engines.

The exhaust velocity is the velocity with which the products of the propellant combustion exit. The thrust is the total force exerted by the exhaust. The thermal efficiency of the engine is a measure of how energy efficient it is, i.e. a ratio of the output energy to the input energy.

The following engines typed were taken for consideration, with specifications given in the table below -

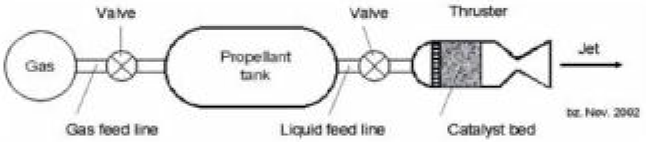
1. Dawn Mission - NSTAR (Ion Drive)
2. Liquid-Fuel Engines - Merlin 1D (SpaceX)
3. Cold gas chemical propulsion - VP-03-001
4. Solid Rocket Boosters - Pratt & Whitney's SRB
5. CubeSAT-Level Electrolysis Propulsion System - Cornell University





The Dawn-NSTAR ion thruster. Shown here is an image of a test run, with Xe (Xenon) gas being used as fuel.

The Solid Rocket Booster, developed by Pratt & Whitney. These have been in use for space launches, but their use as main engines for the shuttle itself is outdated.



Merlin 1D -

S. No.		
1.	Propulsion System	Internal Combustion
2.	Propellant	LOX/ RP-1 (high grade kerosene)
3.	Specific Impulse	282 s
4.	Exhaust Velocity	+311 m/s (Exact figures NA)
5.	Thrust	756,000 N
6.	Thermal Efficiency	75%

NSTAR -

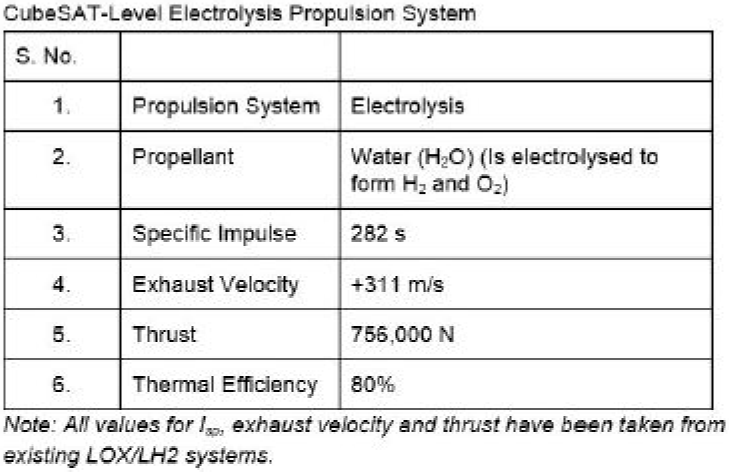
S. No.		
1.	Propulsion System	Ion
2.	Propellant	Xenon
3.	Specific Impulse	3,100 s
4.	Exhaust Velocity	30,411 m/s
5.	Thrust	9.00e-05 N
6.	Thermal efficiency	99%

VP-03-001

S No.		
1.	Propulsion System	Cold-gas propulsion
2.	Propellant	Nitrogen
3.	Specific Impulse	>70
4.	Exhaust Velocity	
5.	Thrust	0.001
6.	Thermal Efficiency	NA

Solid Rocket Boosters -

S No.		
1.	Propulsion System	Solid Rocket Propulsion
2.	Propellant	PBAN-ABCP
3.	Specific Impulse	242 s
4.	Exhaust Velocity	2630 m/s
5.	Thrust	12,000 kN
6.	Thermal Efficiency	NA



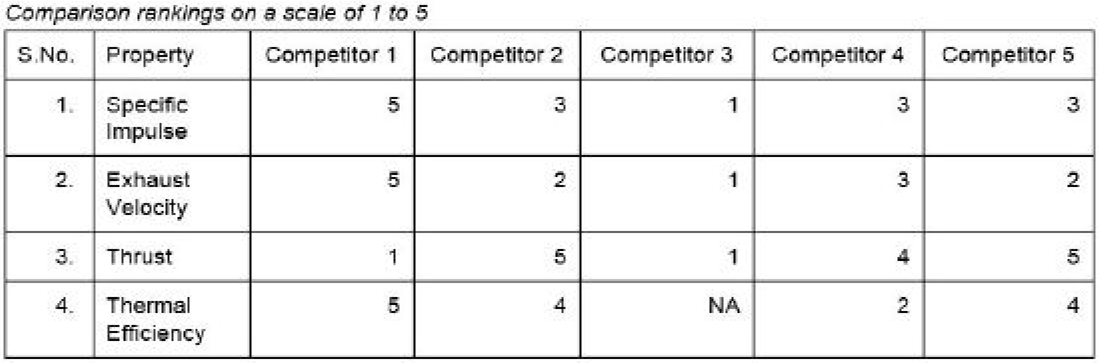
CubeSAT-Level Electrolysis Propulsion System

S. No.		
1.	Propulsion System	Electrolysis
2.	Propellant	Water (H_2O) (Is electrolysed to form H_2 and O_2)
3.	Specific Impulse	282 s
4.	Exhaust Velocity	+311 m/s
5.	Thrust	756,000 N
6.	Thermal Efficiency	80%

Note: All values for I_{sp}, exhaust velocity and thrust have been taken from existing LOX/LH2 systems.

Comparison Ranking Table –

Table II.B.IV ranks the three competitors on a scale of 1 to 5, with one being the least optimal to five being the most optimal value for each property used for benchmarking. An engine with a higher specific impulse is given a higher rating because of its ability to deliver more impulse per unit propellant. Likewise, an engine with a higher exhaust velocity is given a higher rating because of a higher velocity produced for the same amount of propellant. An engine with higher thrust is given a higher rating because of the larger force delivered for each unit of propellant burnt. An engine with a higher thermal efficiency is given a higher rating because it provides a higher percentage of input energy as output energy.



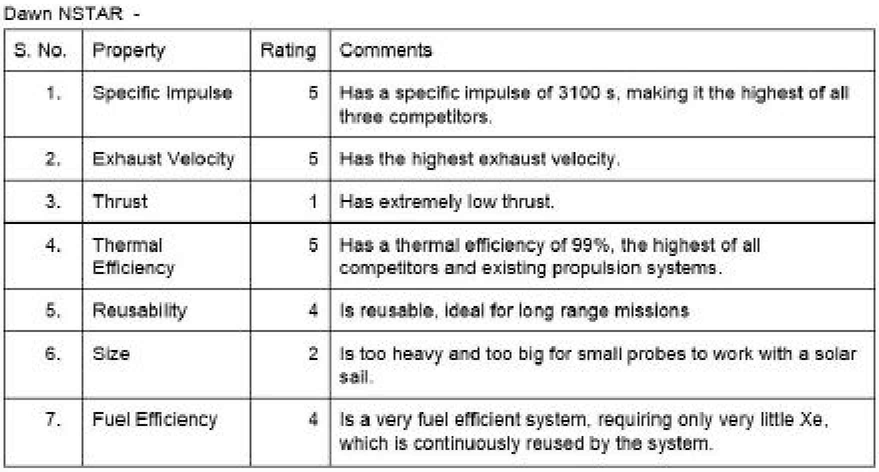
Comparison rankings on a scale of 1 to 5

S.No.	Property	Competitor 1	Competitor 2	Competitor 3	Competitor 4	Competitor 5
1.	Specific Impulse	5	3	1	3	3
2.	Exhaust Velocity	5	2	1	3	2
3.	Thrust	1	5	1	4	5
4.	Thermal Efficiency	5	4	NA	2	4

Subjective Benchmarking Data

Subjective benchmarking data ranks the parameters according to user-defined requirements, i.e. it compares how optimal the values of each given property are for the given problem. For example, a high thermal efficiency is preferable, but a non-reusable rocket is unsuitable for the given requirements. Also, the cost of the engine and size are also very important for its working with a solar sail. For example, an engine with superior thrust and thermal efficiency may be too heavy for a solar sail to propel the required distance within a feasible time frame.

The parameters used for the subjective benchmarking data are as follows:
1. Reusability
2. Size
3. Fuel Efficiency
In addition to these, parameters from the objective benchmarking data have been included as well.



Dawn NSTAR -

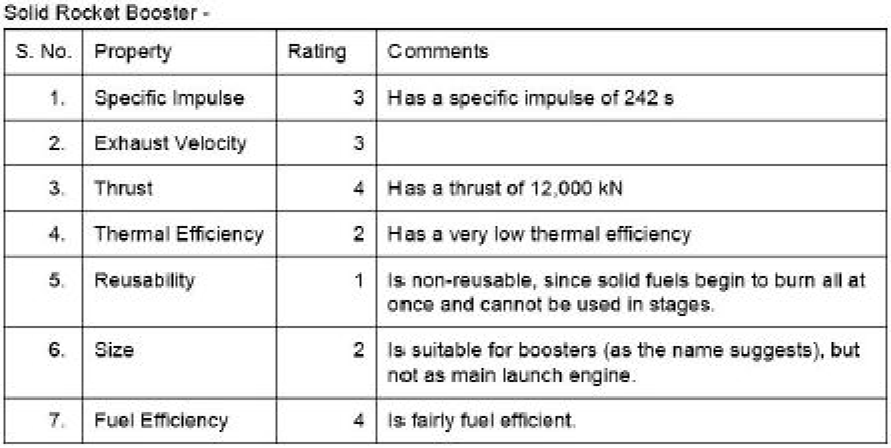
S. No.	Property	Rating	Comments
1.	Specific Impulse	5	Has a specific impulse of 3100 s, making it the highest of all three competitors.
2.	Exhaust Velocity	5	Has the highest exhaust velocity.
3.	Thrust	1	Has extremely low thrust.
4.	Thermal Efficiency	5	Has a thermal efficiency of 99%, the highest of all competitors and existing propulsion systems.
5.	Reusability	4	Is reusable, ideal for long range missions
6.	Size	2	Is too heavy and too big for small probes to work with a solar sail.
7.	Fuel Efficiency	4	Is a very fuel efficient system, requiring only very little Xe, which is continuously reused by the system.

Merlin 1D -

S. No.	Property	Rating	Comments
1.	Specific Impulse	3	Has a specific impulse of 282 s.
2.	Exhaust Velocity	2	Has an exhaust velocity of 311 s, placing it in the middle of the spectrum.
3.	Thrust	5	Has a very high thrust.
4.	Thermal Efficiency	4	Has a thermal efficiency of 75%, highest in the liquid propulsion systems.
5.	Reusability	4	Is a reusable system and ideal for medium to long range missions.
6.	Size	3	Is moderately sized, capable of use with smaller probes as well as rockets like Falcon 9 (v1.1)
7.	Fuel Efficiency	4	Is fairly fuel efficient.

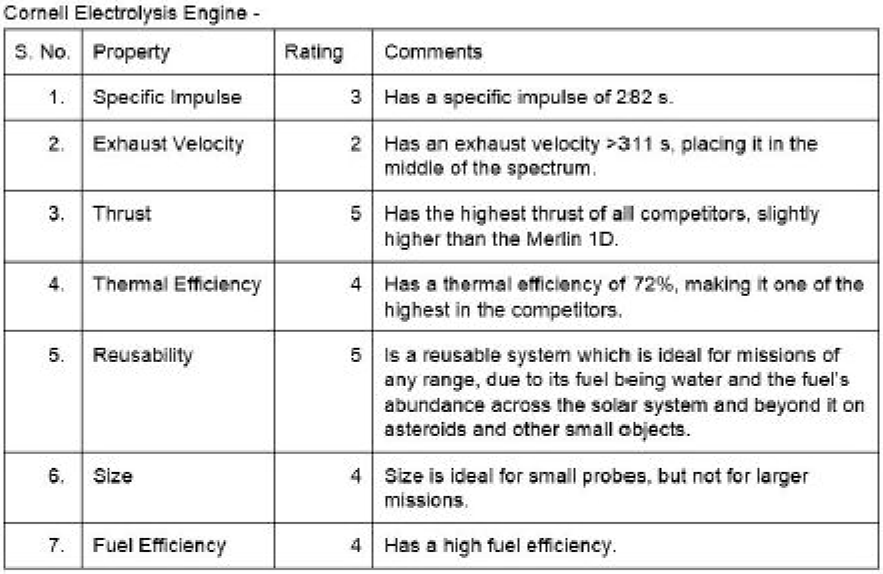
VP-03-001

S. No.	Property	Rating	Comments
1.	Specific Impulse	1	Has a very low specific impulse.
2.	Exhaust Velocity	1	Has a low exhaust velocity
3.	Thrust	1	Has extremely low thrust, making it highly unsuitable.
4.	Thermal Efficiency	NA	NA
5.	Reusability	3	Is reusable.
6.	Size	3	Is small, but does not deliver enough thrust to merit a higher rating.
7.	Fuel Efficiency	NA	NA



Solid Rocket Booster -

S. No.	Property	Rating	Comments
1.	Specific Impulse	3	Has a specific impulse of 242 s
2.	Exhaust Velocity	3	
3.	Thrust	4	Has a thrust of 12,000 kN
4.	Thermal Efficiency	2	Has a very low thermal efficiency
5.	Reusability	1	Is non-reusable, since solid fuels begin to burn all at once and cannot be used in stages.
6.	Size	2	Is suitable for boosters (as the name suggests), but not as main launch engine.
7.	Fuel Efficiency	4	Is fairly fuel efficient.



Cornell Electrolysis Engine -

S. No.	Property	Rating	Comments
1.	Specific Impulse	3	Has a specific impulse of 282 s.
2.	Exhaust Velocity	2	Has an exhaust velocity >311 s, placing it in the middle of the spectrum.
3.	Thrust	5	Has the highest thrust of all competitors, slightly higher than the Merlin 1D.
4.	Thermal Efficiency	4	Has a thermal efficiency of 72%, making it one of the highest in the competitors.
5.	Reusability	5	Is a reusable system which is ideal for missions of any range, due to its fuel being water and the fuel's abundance across the solar system and beyond it on asteroids and other small objects.
6.	Size	4	Size is ideal for small probes, but not for larger missions.
7.	Fuel Efficiency	4	Has a high fuel efficiency.

Summary of Results –

1. Specific Impulse – Objectively, the specific impulse for a propulsion must be >280 s
2. Exhaust Velocity – Objectively, the exhaust velocity must be >311 m/s.
3. Thrust – Objectively, the propulsion system must be able to deliver to deliver at least 756,000 N of thrust.
4. Thermal Efficiency – Objectively, the system must have a thermal efficiency of at least 72%.
5. Reusability – Subjectively, the system must be reusable to endure multiple firings of the engine and not be a one-use system (like Solid Rocket Boosters), i.e. it must be able to operate in stages.
6. Size – Objectively, the engine cannot be too big or too heavy in order to keep total weight of the system at a minimum.
7. Fuel Efficiency – Objectively, the fuel used must have high efficiency in order to maintain a high input/output ratio.

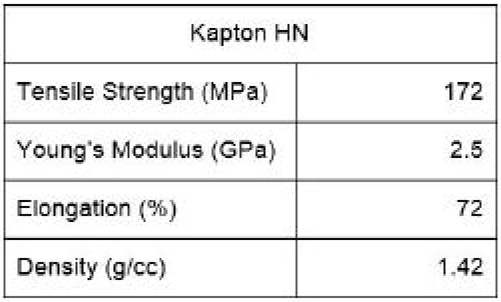
DESIGN CONCEPT GENERATION, ANALYSIS, AND SELECTION

Design Considerations –
The first step that needed to be accomplished was to determine what material and type of solar sail and engine to use.
A major part in choosing the sail type was ease of storage and deployment, as the sail would be required to be frequently retracted and deployed as it switches between interplanetary and deep space cruising. A major criteria for choosing the material was also the same, as certain materials are prone to quick degradation and wear and tear, and would hence be unsuitable for the task. Also important was the availability and cost of the said material.

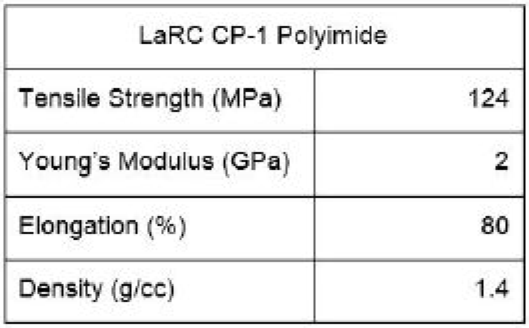
The key points in engine selection were size, weight and cost of producing the engine, the availability of fuel required to propel the engine, the fuel's impact on the environment and efficiency of the engine. The solar sail and engine models chosen below were picked after extensive research by the team members, the research being on existing candidates which could fulfil the solution's requirements.

Choosing a Sail Material matrix

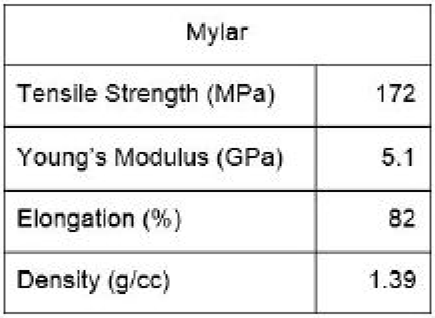
Specifications regarding "Choosing a sail material" matrix –



Kapton HN	
Tensile Strength (MPa)	172
Young's Modulus (GPa)	2.5
Elongation (%)	72
Density (g/cc)	1.42



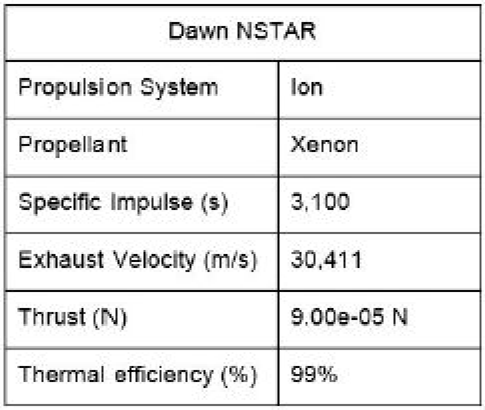
LaRC CP-1 Polyimide	
Tensile Strength (MPa)	124
Young's Modulus (GPa)	2
Elongation (%)	80
Density (g/cc)	1.4



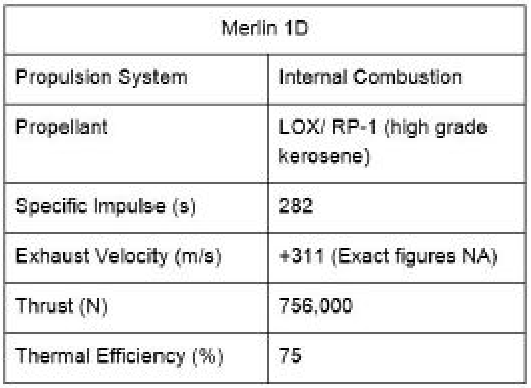
Mylar	
Tensile Strength (MPa)	172
Young's Modulus (GPa)	5.1
Elongation (%)	82
Density (g/cc)	1.39

Choosing an Engine matrix

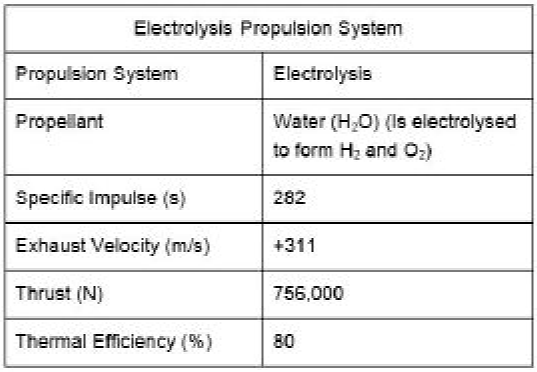
Specifications regarding the "choosing an engine" matrix –



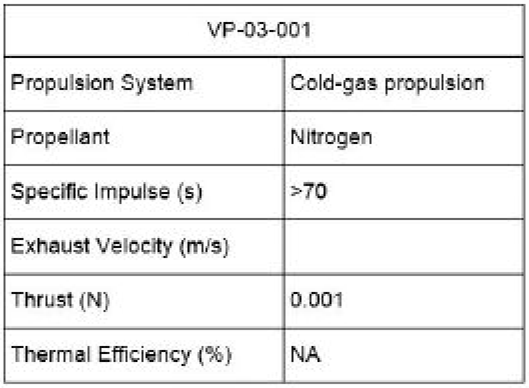
Dawn NSTAR	
Propulsion System	Ion
Propellant	Xenon
Specific Impulse (s)	3,100
Exhaust Velocity (m/s)	30,411
Thrust (N)	9.00e-05 N
Thermal efficiency (%)	99%



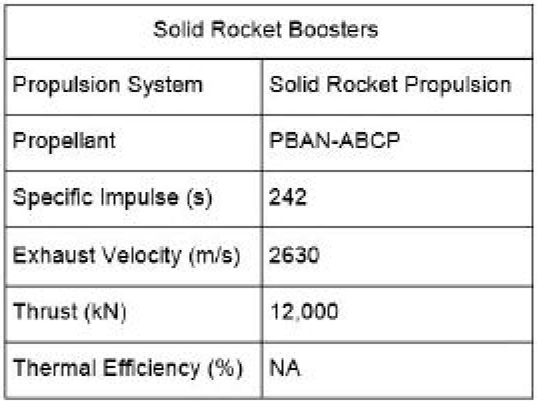
Merlin 1D	
Propulsion System	Internal Combustion
Propellant	LOX/ RP-1 (high grade kerosene)
Specific Impulse (s)	282
Exhaust Velocity (m/s)	+311 (Exact figures NA)
Thrust (N)	756,000
Thermal Efficiency (%)	75



Electrolysis Propulsion System	
Propulsion System	Electrolysis
Propellant	Water (H_2O) (Is electrolysed to form H_2 and O_2)
Specific Impulse (s)	282
Exhaust Velocity (m/s)	+311
Thrust (N)	756,000
Thermal Efficiency (%)	80



VP-03-001	
Propulsion System	Cold-gas propulsion
Propellant	Nitrogen
Specific Impulse (s)	>70
Exhaust Velocity (m/s)	
Thrust (N)	0.001
Thermal Efficiency (%)	NA



Solid Rocket Boosters	
Propulsion System	Solid Rocket Propulsion
Propellant	PBAN-ABCP
Specific Impulse (s)	242
Exhaust Velocity (m/s)	2630
Thrust (kN)	12,000
Thermal Efficiency (%)	NA

Solution Matrix –

Brainstorm solutions with brief descriptions –
1. Self-Repairing Feature – The sail will possess an auto-repair system that will immediately seal the damaged portion with the same material the sail is made of.
2. Circular sail with pie shaped portions for easier storage and deployment.
3. A lorimerlite structure encased between two layers of the sail (the front and the back), with the repair resin flowing through the cells of the structure.
4. The lorimerlite structure will have a very thin vacuum layer (~0.1 micron) between it and the resin to act as an indicator – when the sail gets ruptured, the vacuum and the structure will be breached, which will prompt the resin's flow into the hole. The team decided to name it **SKIN**,

because much like the healing mechanism of human skin after it gets a cut or graze, the repair resin (like the platelets and other blood cells) seals the hole/breach/damage (like the cut/rupture on the skin) and like how new skin slowly grows on the wound, merging with the original skin, the resin solidifies becoming a part of the sail's original structure.
5. Usage of Reflective CP-1 Polyimide for maximum thrust.
6. Use of CP-1 Polyimide resin as the repair material in the honeycomb-shaped skeletal structure.
7. Trusses behind the sail for aiding in deployment
8. Photoelectrolytic engine - engine which involves photoelectrolysis of water to produce hydrogen and oxygen as bipropellants, which are then burnt as fuel. The engine's design is modeled upon that of the Cornell University's CubeSAT-level Electrolysis Propulsion system, although its dimensions have been adjusted to larger spacecraft requirements (- Rosetta, New Horizons, Mangalyaan).
9. Water in the form of ice will be used in the photolectrolytic engine, as water will instantaneously freeze in outer space with no insulation around it.
10. The engine will contain a series of heating coils which will first melt the ice to liquid water, preparing it for electrolysis.
11. The engine will also contain a Proton Exchange Membrane (PEM) electrolyser attached to the holding chamber in which water will be electrolysed to form O2 and H2.
12. The outer body of the engine will be covered with solar panels, which will power the electrolyser, engine and the spacecraft.
13. The electrolysis chamber will have a valve leading to the combustion chamber, where a spark will be passed by the sparkplug to ignite the gases.
14. The gases will then produce heat energy and water vapour as products, the heat energy providing the necessary thrust to propel the spacecraft.
15. For contingency measures, the engine will be connected to a Li-Polymer battery in case supply of energy falls short.

Decision Specifications (In order of importance) –

The decision specifications have been given in order of importance

Presentation and Justification of Solution Requirements.
DQ = Disqualifying Specification

Specification 1 (DQ): The solution, when implemented and used according to the assembly and operating instructions, will prevent serious harm to scientific equipment and specimens aboard the system and will not interfere with the primary system of the spacecraft.
Positive Aspect (+ve): Device prevents fatal injuries and malfunction as mentioned.
Negative Aspect (-ve): Device does not prevent fatal injuries and malfunctions as mentioned.

Specification 2 (DQ): The solution, when correctly implemented, does not have a tendency to cause malfunction or interference with the spacecraft's primary systems.
Positive Aspect (+ve): Device does not cause malfunction or interference in primary systems.
Negative Aspect (-ve): Device tends to cause interference in primary systems.

Specification 3 (DQ): The technology necessary to implement all aspects of the solution exists.

Positive Aspect (+ve): Present technology is able to fulfil design requirements.
Negative Aspect (-ve): Present technology is unable to meet basic design requirements.

Specification 4 (DQ): The solution is compatible with existing systems and rocket models.
Positive Aspect (+ve): Solution is compatible with all existing launch systems and launch models.
Negative Aspect (-ve): Solution is incompatible with existing launch systems and rocket models.

Specification 5 (DQ): The solution will require smaller amounts of fuel.
Positive Aspect (+ve): The solution will require low amounts of fuel for propulsion.
Negative Aspect (-ve): The solution will require large amounts of fuel.

Specification 6 (DQ): The solution will be open-sourced by federal agencies and business houses.
Positive Aspect (+ve): The solution will be available for public access.
Negative Aspect (-ve): The solution will have limited access to federal agencies and select business houses.

Specification 7: The solution concept will not overwhelmingly resemble existing designs or solutions for the same problems.
Positive Aspect (+ve): The solution will be unique, innovative and will rectify flaws and problems as stated in the problem statement.
Negative Aspect (-ve): The solution will overwhelmingly resemble existing solutions.

Specification 8: The system should be capable of refueling itself in space when it runs out of stored fuel.
Positive Aspect (+ve): The system will be able to refuel itself in space and hence will not require to carry all the required fuel for the journey aboard the spacecraft, drastically reducing payload.
Negative Aspect (-ve): The system will have to carry all the required fuel as payload because it cannot refuel in space due to lack of fuel or a refueling mechanism.

Specification 9: Each solution, for marketability and stability, will cost less than existing systems.
Positive Aspect (+ve): The solution will be cheaper than its competitors and will have an economic advantage.
Negative Aspect (-ve): The solution will be more expensive than the competitors, putting it at a disadvantage.

Specification 10: The solution will contain replaceable parts, and must be capable of repair without any human interference.
Positive Aspect (+ve): The system will be able to repair itself in space even when out of communication with the system engineers (due to any reason), ensuring every part of the spacecraft stays in operation at all times.
Negative Aspect (-ve): The system will be unable to conduct repairs on its own in space and if any part goes out of commission

Specification 11: The solution must be low-maintenance, i.e. it will not require frequent repairs and that repair tasks will not be extremely expensive.
Positive Aspect (+ve): The solution will not require frequent maintenance and repair, and that repair operations conducted will not go on to exceed the cost of the solution itself.

Negative Aspect (-ve): The solution will require frequent maintenance and repairs, with its costs exceeding the cost of a brand new system.

Specification 12: The solar sail will not interfere with the working of the photoelectrolytic engine
Positive Aspect (+ve): The solar sail and the photoelectrolytic engine will work seamlessly as one unit, providing the spacecraft with sufficient thrust.
Negative Aspect (-ve): The solar sail will interfere with the working of the photoelectrolytic engine.

Specification 13: The photoelectrolytic engine must not interfere with the working of the solar sail.
Positive Aspect (+ve): The photoelectrolytic engine will seamlessly work with the solar sail as one unit, providing the spacecraft with sufficient thrust.
Negative Aspect (-ve): The photoelectrolytic engine will interfere with the solar sail's functioning.

Specification 14 (DQ): The sail must not obstruct the exhaust nozzle of the engine.
Positive Aspect (+ve): The sail will not obstruct the exhaust nozzle of the engine, preventing any damage to itself and will not obstruct the flow of the propellant from the nozzle.
Negative Aspect (-ve): The sail will obstruct the exhaust nozzle of the engine and will interfere with the flow of the propellant as well as end up causing severe damage to the sail.

Specification 15: The sail must be retractable.
Positive Aspect (+ve): A retractable sail will be capable for multiple uses and not as a one-time use and throw feature.
Negative Aspect (-ve): The sail will be non-retractabe and will henceforth be only capable of a onetime use.

Specification 16: The solution can withstand temperatures as low as absolute zero.
Positive Aspect (+ve): The solution will be capable of withstanding all temperatures deep space and will be capable of functioning in all temperature conditions.
Negative Aspect (-ve): The solution will be unable to work in all temperature conditions.

Defense of Design Choice

For the solar sail, Mylar initially won the matrix, but Kapton HN and CP-1's lower Young's Modulus and availability in liquid resin form provided them meeting a key design requirement of a material available in both liquid resin as well as sheet form in order to use the sail's building material as the repair material as well.

For the engine, the Merlin 1D initially won the matrix, but the electrolysis engine's key feature of using water as fuel put the latter at an advantage, as water's availability throughout space on different asteroids makes refueling a possibility, something not possible with LOX/RP-1 propelled engines.

Material Choice of Repair Resin Matrix -

The following materials are shortlisted for the repair resin in the solar sail -
1. Teflon FEP resin
2. CP-1 Polyimide Resin
3. Aramid (Kevlar) resin

Density (g/cc):
Positive (+): 1.29 - 1.54
Negative (-): >1.54 or <1.29

Price (per kg): The resin must be reasonably priced in order to keep the solution economically feasible.
Positive (+): < $50
Negative(-): > $200

Temperature: The resin must retain its liquid form at extreme temperatures
Positive (+): Can retain liquid form at temperatures < 20K
Negative (-): Does not retain liquid form at temperatures < 20K

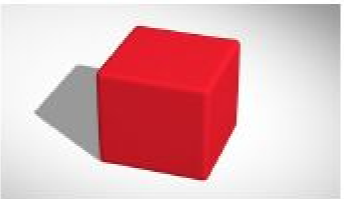
Toxicity: The material should ideally be non-corrosive and non-reactive when in storage.
Positive (+): Is non corrosive and does not react while in storage.
Negative (-): Is corrosive and/or reactive while in storage.

Compatibility: Must be compatible with the sail material, i.e. sail remains fully functional after repair.
Positive (+): Resin is compatible with sail material; sail is fully functional after repair.
Negative(-): Resin is incompatible with sail material and is unable to repair the sail, leaving the damaged portion non-functional.

Defense of design choice: **CP-1 Polyimide Resin** was the winner here because it meets all the qualification criteria of density, costs, toxicity and most importantly, compatibility, for it is the only sail material which is available in both resin as well as sheet form, removing the compatibility issue. CP-1 will also be used for the SKIN Structure which will hold the repair resin, separating it from the front and back of the sail by a thin layer of vacuum (~0.1 micron).

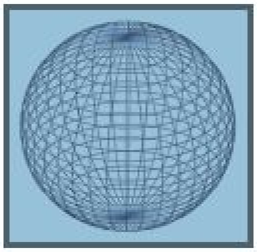
SKIN Structure Design Decision Matrix -

Several designs for the structure of SKIN were considered, each given below with its strengths and weaknesses -
1. Cube
2. Spherical/Bubble-like
3. Hexagonal/Honeycomb
4. Lorimerlite

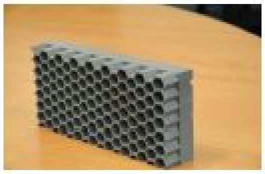


Cube -
Advantages -
Good at taking compressive forces.
Disadvantages -

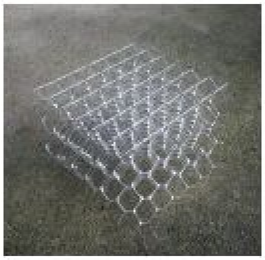
Cannot take tension forces as well as compressive forces Is prone to lateral buckling.



Spherical/Bubble-Like –
Advantages –
Strongest figure in a constant universal pressure 3D environment (liquids in particular).
Disadvantages –
In case of solids, geometry is rarely ever in circular form.



Hexagonal/Honeycomb –
Advantages –
Strongest shape in compression
A tessellation of hexagons is the strongest 2-D structure isotropically.
Disadvantages –
In 3D, hexagon/honeycomb structures are prone to buckling.



Lorimerlite –
Advantages –
Strongest figure in 3D geometry
Structure reduces the danger of lateral buckling as each beam finds the shortest unbraced path within the structure.
Loads are distributed evenly at each tetrahedral joint.

Defense for SKIN structure design decision –

The Lorimerlite structure was chosen as it is the strongest 3-D structure, being able to bear the highest amount of compressive forces from a minimal amount of material. As stated by its creator Alexander Lorimer, lorimerlite "resists compressive forces with a minimal amount of structural material. The geometry of the structure is inspired by the tetrahedral geometry of soap bubbles, which are constantly trying to optimise their form in terms of material usage".
"This lorimerlite framework fills the largest volume of spaces with the least un-braced length of a predefined number of struts, making them inherently resistant to buckling and enhancing the strength of the entire structure," and "All struts meet at an angle of 109.5 degrees, as with the vertices in soap bubble clusters.”

Defense of Engine Propellant Decision –

Water was chosen as the photo electrolytic engine's fuel as it does not require much energy when compared to the calorific values of the fuel employed (15870 kJ/kg is required to electrolyze water while H2 has a calorific value of 141790 kJ/kg).
Most existing crafts use LH2/LOX as their bipropellants, but transport and storage of these is difficult. By carrying them as water, a key hurdle is eliminated.

APPLICATION OF STEM PRINCIPLES AND PRACTICES

Part 1 – The Solar Sail

Pressure exerted by upon any surface exposed to EM radiation, and can be calculated by three methods, depending upon the way the pressure is exerted:

Radiation Pressure by absorption (1) –

$$P_{absorb} = \frac{E_f}{c}\cos^2\alpha$$

Radiation Pressure by reflection (2) –

$$P_{reflect} = \frac{2E_f}{c}\cos^2\alpha$$

Radiation Pressure by emission (3) –

$$P_{emission} = \frac{E_f}{c} = \frac{\varepsilon\sigma}{c}T^4$$

Where –
Pabsorbtion = Pressure by absorption
Preflect = Pressure by reflection
Pemission = Pressure by emission
Ef = Energy flux (rate of transfer of energy)
c = speed of light
cos□ = angle with which radiation strikes sail
□ = Stefan-Boltzmann constant
T = temperature of incident radiation
□ = Emissivity (coeff. of emission)

For the sun, an easier equation can be used to calculate radiation pressure, as given below –
1. For perfectly absorbing planar surface **(4)** –

$$P_{absorb} = \frac{W}{cR^2}\cos^2\alpha$$

2. For perfectly reflecting planar surface **(5)** –

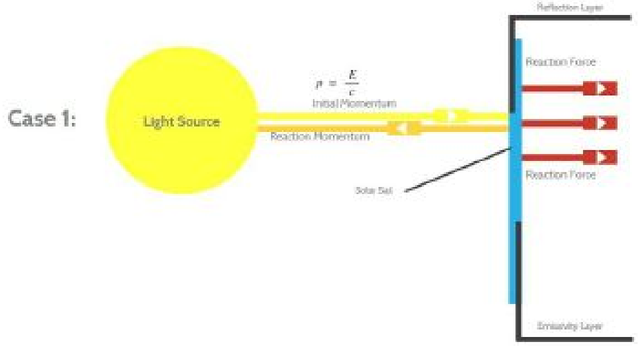
$$P_{reflect} = \frac{2W}{cR^2}\cos^2\alpha$$

Where
W = Solar Constant (energy flux in the previous equations); it has a value of 4.59 Wm–2
R = Distance of sail from the sun
cos□ = Angle of incident radiation

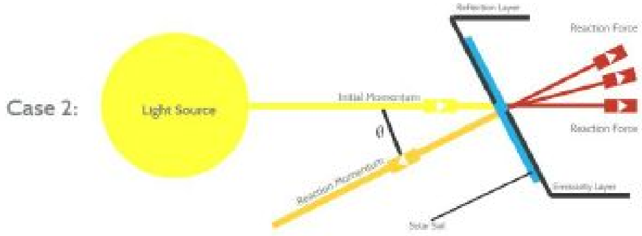
For a sail made of a reflective material, **(2) and (5)** were primarily used in the calculations.

Solar sails are extremely dependent on the intensity of light – the further you get away from the light source, the less efficient the sails become because less photons hit the sail. When there are no photons hitting the sail, the spacecraft will still travel through space, but without acceleration, it will decrease in speed, becoming extremely slow because space is not 100% vacuum. For instance, Jupiter has about 1/24 the amount of light intensity of Earth, so a spacecraft by Jupiter would have to be 24x larger to produce a similar force of a solar sail close to Earth. Solar sails have a maximum speed which is 10% the speed of light, which equates to 18,600 miles per second or 67,100,000 mph. Solar powered spacecrafts are able to travel faster than conventional rocket fueled spacecrafts due to constant light pressure being applied to the sail, propelling it forward.

A clearer illustration of the sail's workings is given below –



Case 1: When the reflecting surface (i.e. solar sail) is perpendicular to incident radiation.



Case 2: When the reflecting surface (i.e. solar sail) isn't perpendicular to incident radiation

Forces and Momentum

Initial Momentum

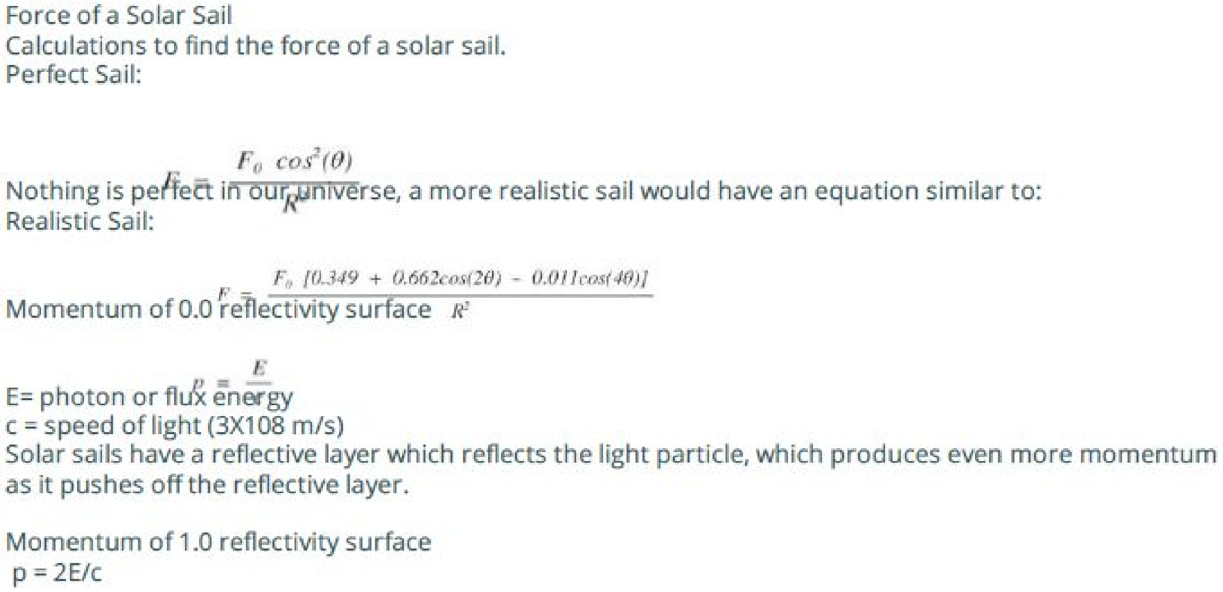
The initial momentum is the momentum of protons from a light source. This momentum applies a force into the solar sail.

Reaction Momentum

Reaction momentum is the reaction from the initial momentum from the light source. The reaction momentum is always perpendicular to the solar sail's surface, as seen in case 2 diagram.

Reaction Force

The reaction force is thermal energy which is due to the solar sail being heated up. The emissivity layer emits thermal heat, which creates a reaction force which is a negative force on the spacecraft trajectory.



Force of a Solar Sail
Calculations to find the force of a solar sail.
Perfect Sail:

$$F = \frac{F_0 \cos^2(\theta)}{R}$$

Nothing is perfect in our universe, a more realistic sail would have an equation similar to:
Realistic Sail:

$$F = \frac{F_0 [0.349 + 0.662\cos(2\theta) - 0.011\cos(4\theta)]}{R^2}$$

Momentum of 0.0 reflectivity surface

$$p = \frac{E}{}$$

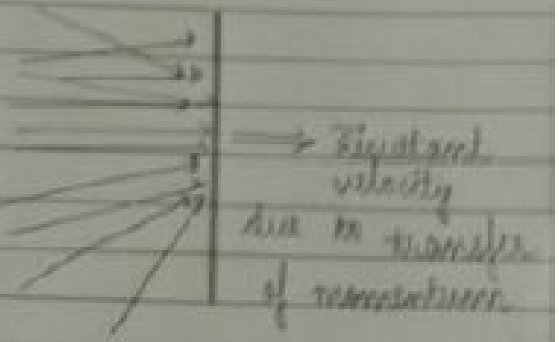
E= photon or flux energy
c = speed of light (3X108 m/s)
Solar sails have a reflective layer which reflects the light particle, which produces even more momentum as it pushes off the reflective layer.

Momentum of 1.0 reflectivity surface
p = 2E/c

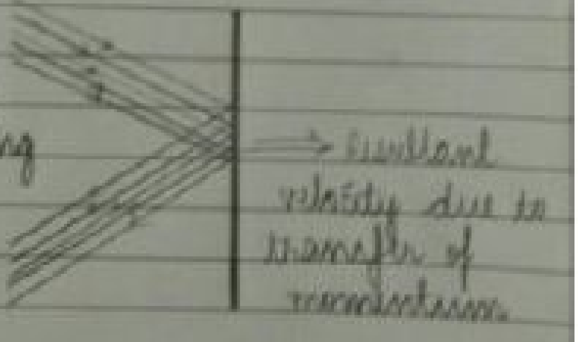
Calculations and Principle for Sail Propulsion

When sunlight / radiation (which has momentum) falls on a reflective surface, the photons are reflected back instead of being absorbed. The sail will accelerate at a higher rate than an absorbing sail, since reflected light transfers more of its momentum to the sail than when it is absorbed. The principle applies to all forms of sails.

1 Absorptive sail -
Radiation is absorbed, transferring momentum to the sail, pushing it forward.



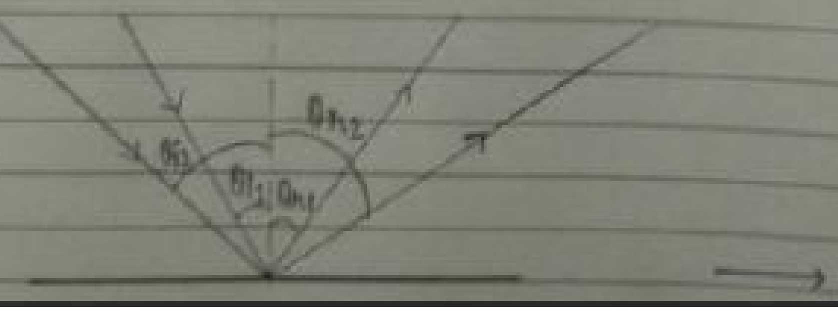
2 Reflective sail -
Here, radiation is reflected off the sail, which transfers momentum to the sail, pushing it forward.



By the law of Reflection.
cos θ is greatest when ∠ is closer to normal.

$\cos\theta_{i_1} > \cos\theta_{i_2}$

$V\cos\theta_{i_1} > V\cos\theta_{i_2}$



Calculation for Solar Radiation Pressure

solar radiation pressure at 1 AU $= \frac{F}{A}$

$= \frac{F_0 \cos^2\alpha}{R^2}$

$= \frac{2W \cos^2\alpha}{cR^2}$

$= \frac{2\times1361 \times \cos^2 0}{(3\times10^8 \times(1))^2}$

$= \frac{2\times\cancel{1361}\,453.33\times(1)^2}{\cancel{3}\times10^8}$

$= \frac{906.66\times(1)^2}{10^8}$

$= 906.66\times10^{-8}\ N m^{-2}$

$= 9.0666\times10^{-6}\ N m^{-2}$

$= \boxed{9.067\ \mu Pa}$

where α = angle with normal (0°)
W = solar constant
= 1361 W m⁻²
c = speed of light (ms⁻¹)
R = distance of sail from the sun (AU)

Since a sail does not function at 100% efficiency (the wrinkles in the sail, tilt from the sun, angle of incidence of radiation, absorbance, etc.) the actual number is closer to 90%.

$P_{actual} = (0.9)(9.067\times10^{-6})\ Pa$
$= 8.16\mu Pa$

The sail is responsible for providing at least 35%-40% of the total thrust required by the spacecraft, the rest being provided by the photoelectrolytic engine.

For a craft weighing 1500 kg (including weight of engine, scientific instruments and initial fuel for payload for a trans Mars journey) →

Calculations for Sail Area And Radius

Average velocity for the craft = 20 kms^{-1}

= 20000 ms^{-1}

Thrust required = P×A

Thrust required can be calculated by using thrust to weight ratio.

$T/W = 165.8$

$\Rightarrow \frac{\text{Thrust}}{\text{Weight}} = 165.9$

$= \frac{P \times A}{1300}$

$165.9 \times 1300 = 8.16 \times 10^{-6} \times A$

$A \times (8.16 \times 10^{-6}) = 165.9 \times 1300$

$A = \frac{165.9 \times 1300}{8.16 \times 10^{-6}}$

$= \frac{\cancel{165.9} \times 13 \times 10^{8}}{\cancel{8.16}}$

$= 264 \times 10^{8}$ m^2

Radius of sail (R) = $\sqrt{\frac{264 \times 10^{8}}{\pi}}$

$= \sqrt{\frac{264 \times 10^{8}}{3.14}}$

$= \sqrt{84.09 \times 10^{8}}$

$R = 9 \times 10^{4}$ m

Since the sail supplies 35%–40% of thrust

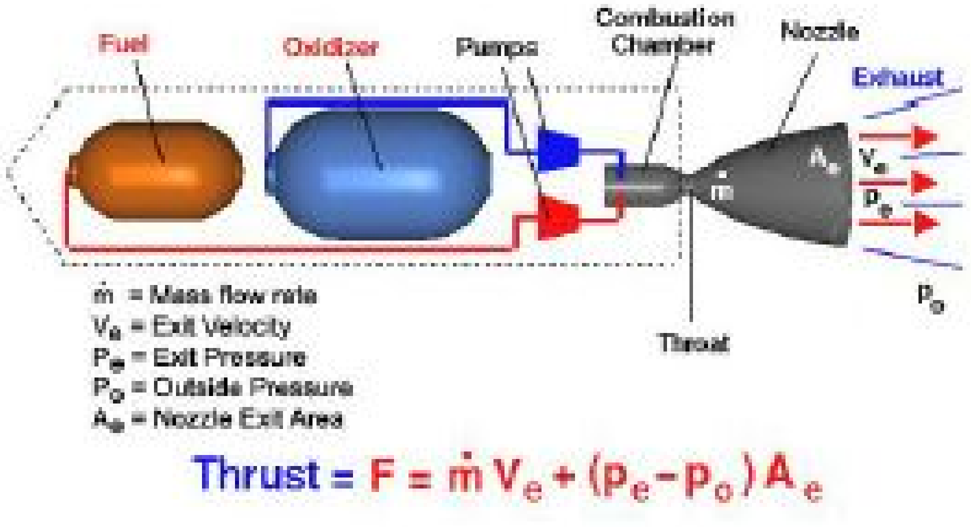
Radius = (0.4)(R)

= (0.4)(9×10^{4})

= 36 km

Thrust Equation of a Liquid Rocket Engine

In a rocket engine, stored fuel and stored oxidizer are ignited in a combustion chamber. The combustion produces great amounts of exhaust gas at high temperature and pressure. The hot exhaust is passed through a nozzle which accelerates the flow. Thrust is produced according to Newton's third law of motion.



The above figure depicts the **Thrust Equation** for a liquid rocket. In this equation –
m= Mass flow rate of gases through the exhaust nozzle which is equal to the mass rate of consumption of propellant of the engine.

Ae = Nozzle exit area. The amount of thrust produced by the rocket depends on the mass flow rate through the engine, the exit velocity of the exhaust, and the pressure at the nozzle exit.

Ve = Average Velocity of the axial component of gas velocity crossing the section
Pe = Exit Pressure

All of these variables depend on the design of the nozzle. The smallest cross-sectional area of the nozzle is called the throat of the nozzle. The area ratio from the throat to the exit Ae sets the exit velocity Ve and the exit pressure Pe

Po= Outside pressure which acts to reduce the thrust. At higher altitudes the outside pressure lesser than the exit pressure and it continously increases as flight crosses altitudes; maximum thrust is developed when Po= 0 i.e when rocket operates in vaccum.

Specific Impulse and Total Impulse

The performance of a rocket engines in generally stated in terms of it's specific impulse. It is defined as
Total impulse generated per unit mass of propellant.
Thus if a rocket motor operates for □t seconds, delivering average thrust F Newton consuming m_p mass
or propellant, then by definition

$$I_{sp} = \frac{F.\ \square t}{m_p} = \frac{F}{m}$$

where m = (m_p/□t) is the average rate of consumption of the propellants. Based on the equation an alternate definition for specific impulse can be made. It can be defined as the thrust developed by the engine per unit mass flow of propellant.

To obtain specific impulse the above equation must be divided by the standard acceleration of gravity at
sea level go= 9.8066 m/s^2 , Thus

$$I_{sp}(s) = \frac{I_{sp}\ (N/kg/s)}{g_o\ (m/s^2)}$$

The specific impulse of rocket engine is basically a property of the propellant combination burned in the
combustion chamber.

Total Impulse –
The total impulse, It is the thrust force obtained over the period of burning time of the propellants t

I_t= □ F.d_t

0
For Constant thrust the equation reduces to
It= F. □t
Using this definition ; Total impulse can be defined as total flow rate times the specific impulse of the
propellant combination

It= F. □t = mp.Isp

It is therefore inferred that if all other factors remain the same, total impulse has the same value for a
large thrust lasting for a short period and a small thrust lasting for a longer time.

Impulse Weight Ratio
Impulse weight ratio of rocket system is defined as the ratio of the total impulse to the initial vehicle
load (loaded vehicle). A high value of this ratios indicates the excellence of the overall design of the
complete rocket system
It = Isp.mp = Isp.□
W0 (mf + mp)go go

Electrolysis & Thrust Calculations –

ANODE – $H_2O \longrightarrow \frac{1}{2} O_2(g) + 2H + 2e^-$

CATHODE – $2H_2O + 2e^- \longrightarrow H_2(g) + 2OH^-$

NET REACTION – $H_2O(l) \longrightarrow H_2(g) + \frac{1}{2} O_2(g)$

$E_a = -1.23$ V
$E_c = -0.83$ V
$E_{cell} = -2.06$ V

After electrolysis, the mixture of gases is led by a pipe (whose outflow is controlled by a solenoid valve) to an ignition chamber, where hydrogen "burns" in oxygen to produce water vapour and heat energy as products.

The balanced reaction is as follows –

$$2H_2(g) + O_2(g) \longrightarrow 2H_2O(g) + \Delta H_c$$

ΔH = Released energy

Released energy = -484 kJ mol^{-1}

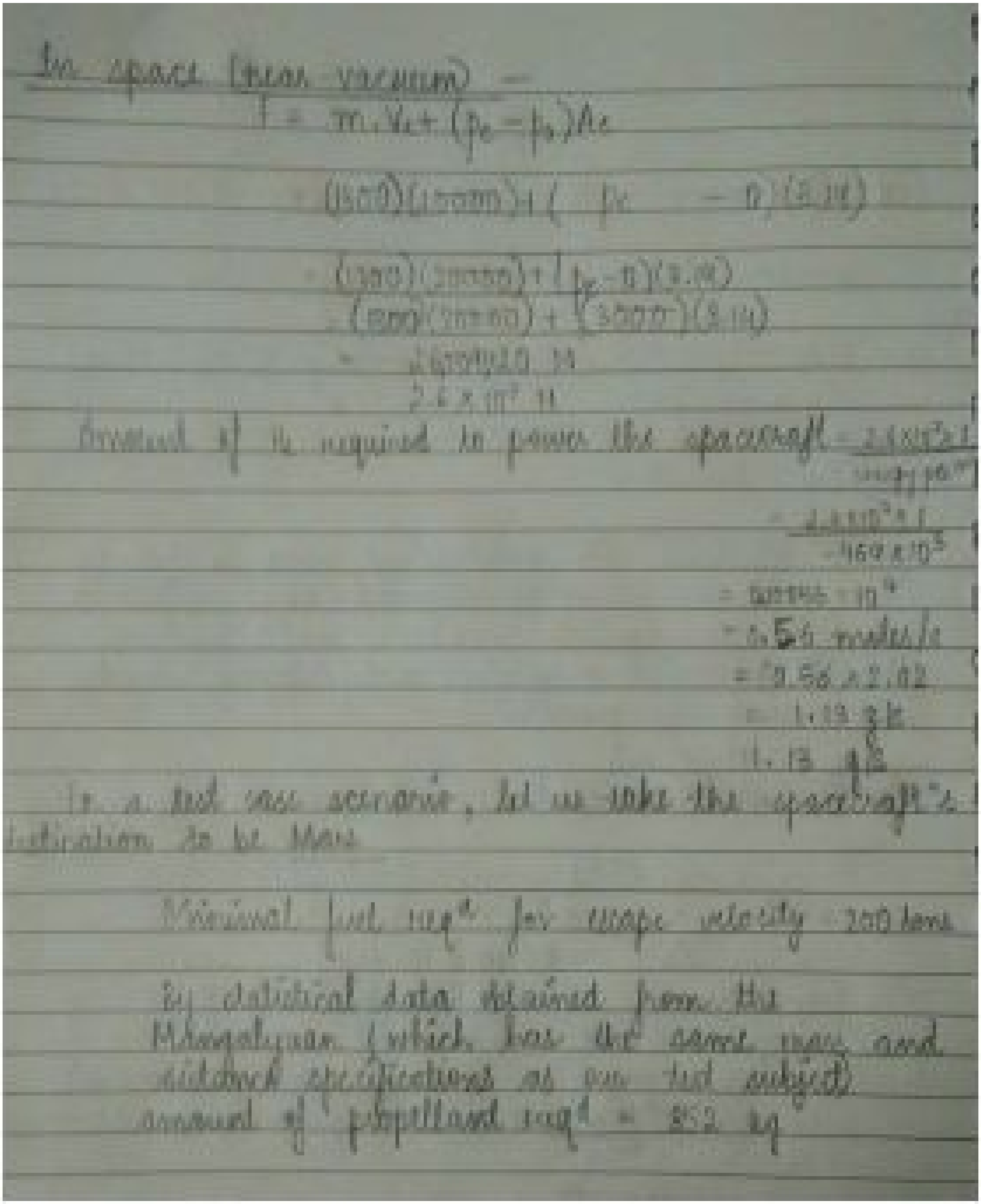
1 mol H_2 = 2.02 g

Thrust required to propel a spacecraft weighing 1300 kg at a speed of 20 kms^{-1} is given by the thrust e

Thrust $F = \dot{m} . V_e + (p_e - p_a) A_e$

where p_e = exit pressure
p_a = outside velocity
$\dot{m}$ = mass
V_e = exit velocity
A_e = nozzle exit area

→



In space (near vacuum) –

$F = \dot{m} V_e + (p_e - p_a) A_e$

$= (1350)(15000) + (p_e - 0)(3.14)$

$= (1350)(20000) + (p_e - 0)(3.14)$

$= (1350)(20000) + (3000)(3.14)$

$= 26009420 \text{ N}$

$2.6 \times 10^7 \text{ N}$

Amount of H_2 required to power the spacecraft $= \frac{2.6 \times 10^7 \times 1}{[illegible]}$

$= \frac{2.6 \times 10^7 \times 1}{469 \times 10^5}$

$= 0.0554 \times 10^1$

$= 0.56$ moles/s

$= 0.56 \times 2.02$

$= 1.13$ g/s

1.13 g/s

In a test case scenario, let us take the spacecraft's destination to be Mars

Minimal fuel req^d for escape velocity 200 tons

By statistical data obtained from the Mangalyaan (which has the same mass and rideback specifications as our test subject) amount of propellant req^d = 852 kg

CONSIDERATION OF DESIGN VIABILITY

The solar sail subsystem occupies the lower 2/3 volume of the spacecraft. Sail closeout panels provide protection for the sail and booms during the launch phase of the mission. These panels need to have spring-loaded hinges that will be released on-orbit, under the command of the spacecraft bus.

Sail subsystem was divided into two primary components—the sail assembly and the boom mechanical assembly. Once assembled, the sail subassembly consisted of a standalone unit that bolted to the bus and connected to the release electronics.

Launch operations consist of a simple, timed two-actuation system. The initiating event consists of a burn-wire release of the door panels. The door panels protect the sail material and help to constrain it for the launch environment and ascent venting.

The sail membranes, fabricated from aluminum coated CP1™ material have been originally used in the ATK Solar Sail Ground Demonstration Project conducted by NASA are z-folded and rolled onto a sail spool. The Trac booms, developed by the U.S. Air Force Research Laboratory, are also rolled onto a boom spool. The stored strain energy of the rolled booms provides the driving force to simultaneously deploy both the booms and the sail quadrants.

Solar Sail Propulsion System Details

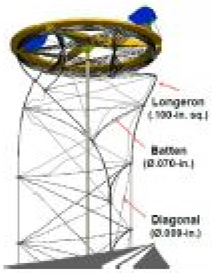
Technical Specifications –

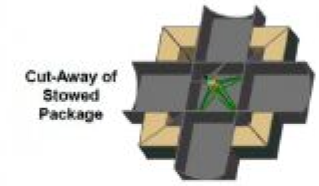
Operating Temperature – 16°C at .98 au
First Natural Frequency – 0.02 Hz

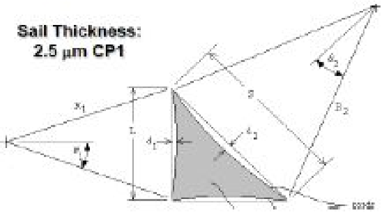
Stowed Package – 1.5 m dia. by 0.53 m
System Mass: □108 kg (w/ contingency)

Characteristic acceleration –
1. 0.76 mm/s2
2. 0.34 mm/s2 with 130 kg

Sail Thickness: 2.5 m CP1
Stowing Specfications – CoilABLE
Ø20-in. (50.5 cm) 18.7-in.-tall (<0.55% of length)
Coilable Mast Mass – 70 g/m (for cubesat)







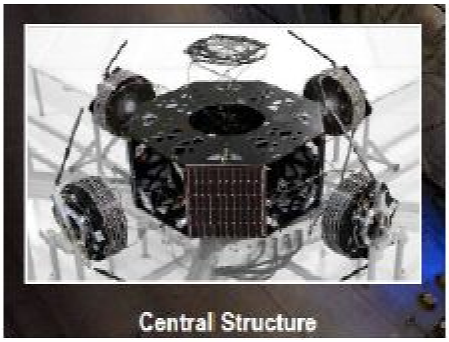
Solar Membrane Specifications

Membrane Design Specs:
- 16 quadrant disk sail
- AL coated 2.5 ▯m CP1, compliant border, 3 point attach
- Compliant Border interface between edge cable and membrane
- Shear insensitive, Cord/Material CTE mismatch insensitive
- Thermal Gradient insensitive Sail Material insensitive

Sail Material: CP1 Polyimide:
- High Operating Temperature (>200o C)
- UV Stable
- Essentially Inert
- Soluble (Wet Process), modifiable with variety additives –improve conductivity and thermal properties 2.5 micron polyimide
- Flight Proven –––flying on Numerous GEOCOM satellites

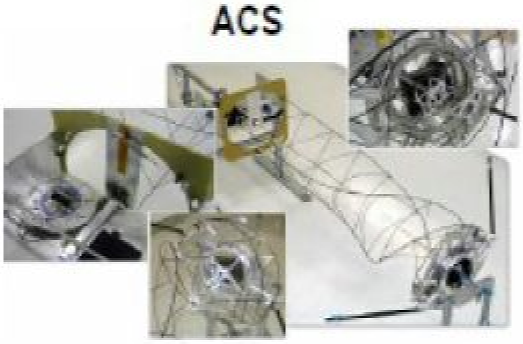
Sail Construction Methods:
- A gossamer film construction similar to gusseted, reflective blankets flying on numerous GEOCOM satellites
- Scalable Construction Methods –––current system >20m
- Adhesive less Bonding Methods –––eliminates sticking and contamination risks.

Sail Membrane

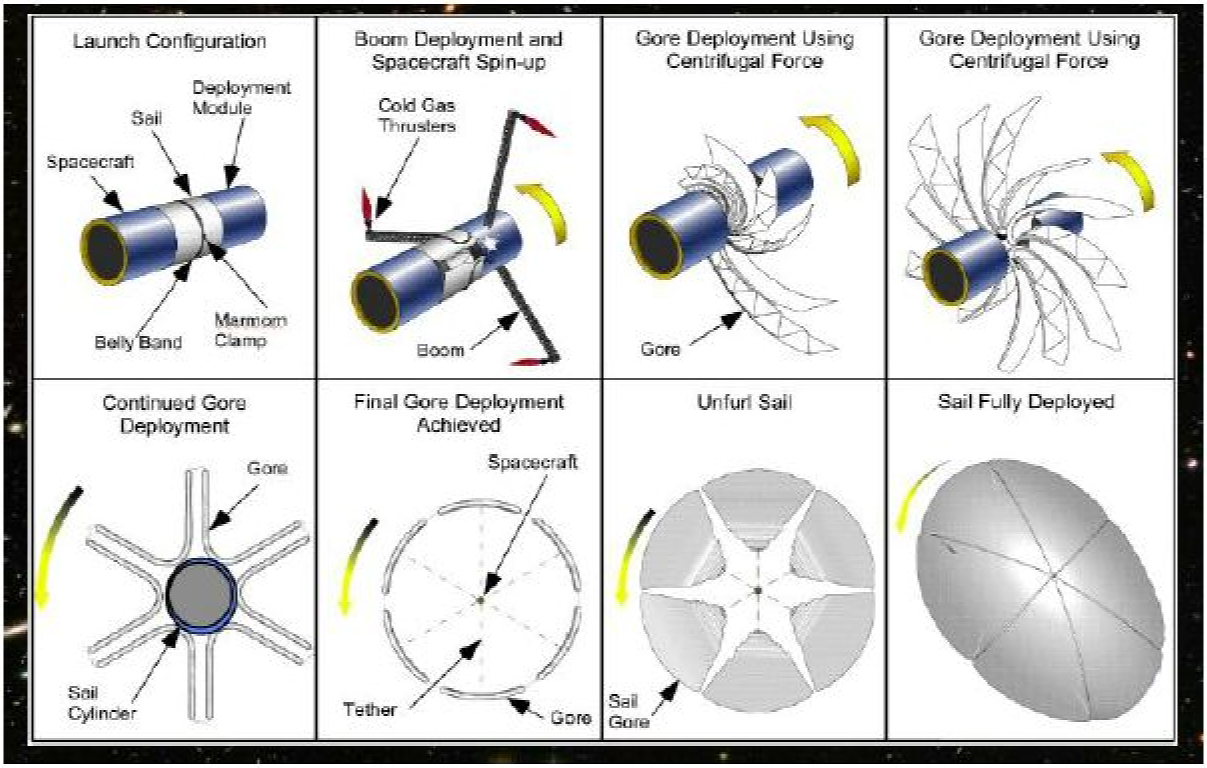


Central Structure

ACS



Sail Deployment Sequence –



The above given illustration of the deployment sequence is applicable to most disk-shaped solar sails as well as to other heliogyro designs as well. Here, the cold-gas micro thrusters attached to the deployment module are fired, pulling the main structure/gore until it is fully deployed. The different sail pieces (in this case, the pie-shaped pieces making up the sail) are then unfurled. For retraction, the thrusters are fired counterclockwise and then the sail is pulled in. This is preliminary idea for the deployment sequence and has been taken from existing designs.

Test Case Scenario Illustration

Calculations to determine minimum speed for penetration of CP-1 polyimide —

For a particle of size / approximate dimensions $1\,m \times 1\,m \times 1\,m$

mass of space rock with given dimensions $= V \times \rho$

$= (1 \times 1 \times 1) \times (2650)$

$= 2650\ kg$

where V = volume

ρ = density of rock (average density)

Assuming the space rock has a velocity v_{av}

where v_{av} = average velocity (space rock)

$v_{av} = 25000\ m/s$

Let time of contact be 1 s

$F = \frac{mv}{t}$

$= \frac{(2650)(25000)}{1}$

$= 66250000$

$= 6.625 \times 10^7\ N$

Over an area of 1 m², pressure $= \frac{F}{A}$

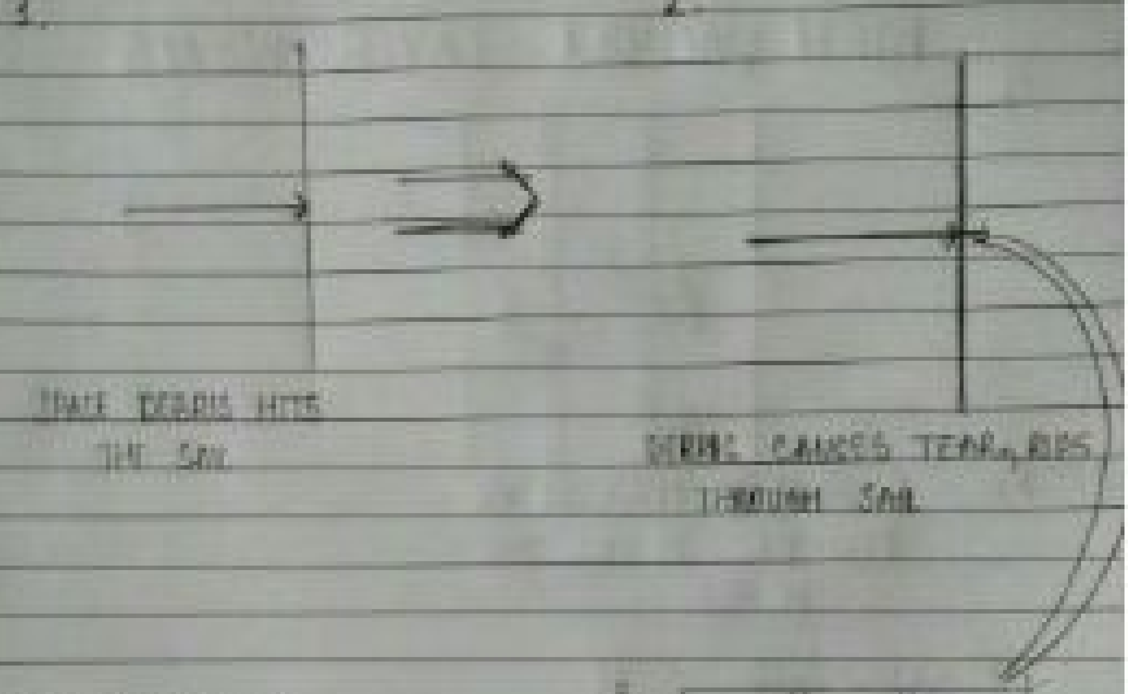
$= \frac{6.625 \times 10^7}{1}$

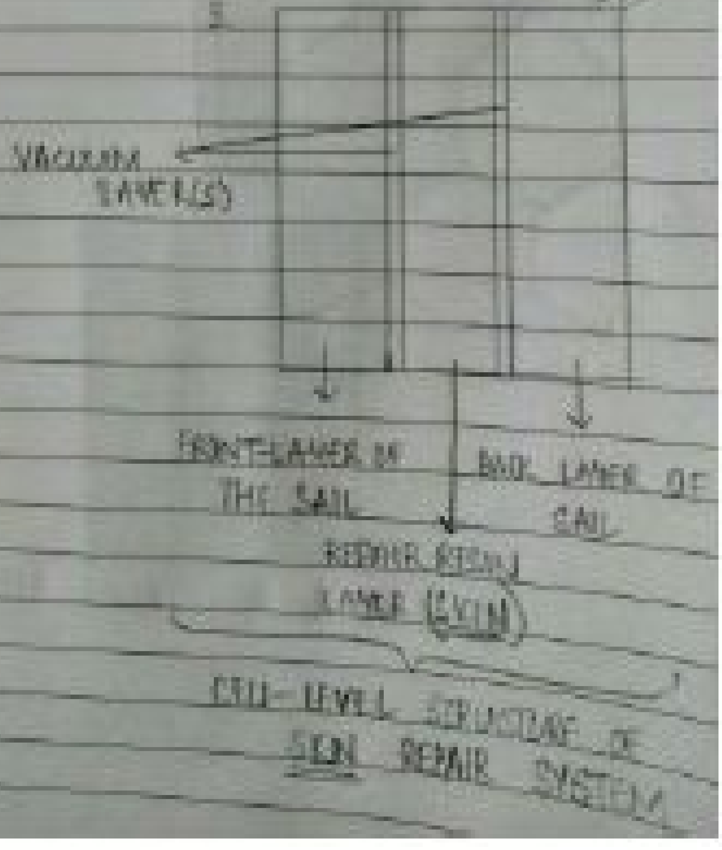
$= 6.625 \times 10^7\ Pa$

Compressive resistance of CP-1 Polyimide = 0.2 kN/m²

The rock will tear a hole through the sail

TEST-CASE SCENARIO

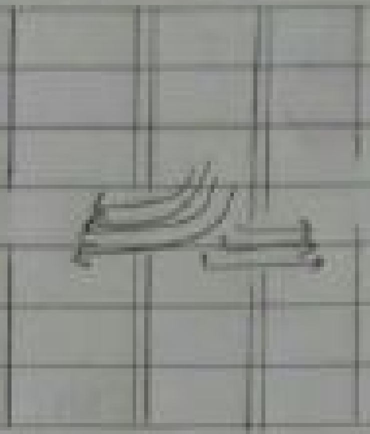




WHEN VACUUM LAYER BREAKS -

WHEN VACUUM LAYER BREAKS

4.

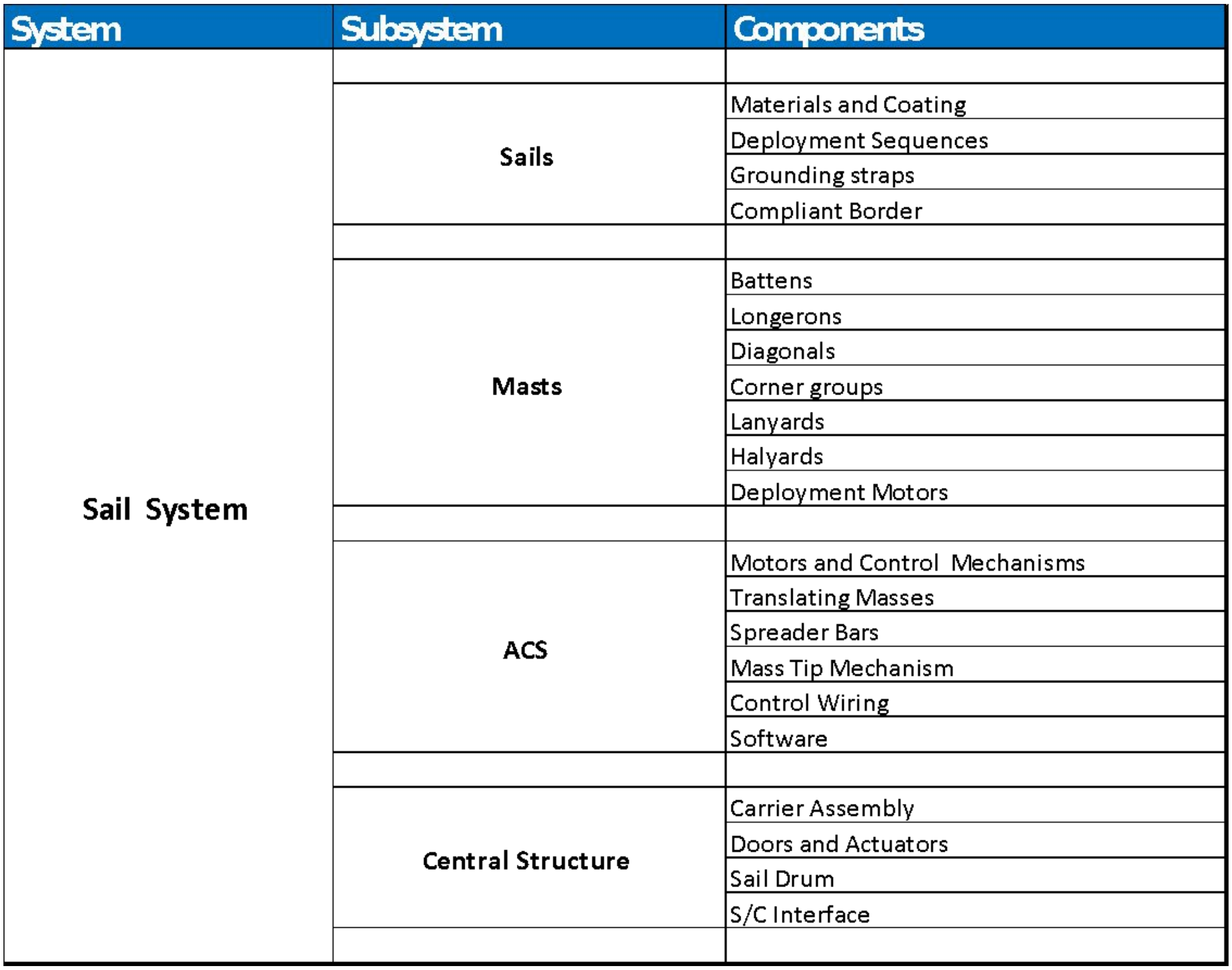


RESIN FLOWS THROUGH THE STRUCTURE TO FILL THE HOLES).

5.

CP-1 SOLIDIFIES, SEALING THE HOLE.

Solar Sail System and Sub systems at a glance



System	Subsystem	Components
Sail System		
	Sails	Materials and Coating
		Deployment Sequences
		Grounding straps
		Compliant Border
	Masts	Battens
		Longerons
		Diagonals
		Corner groups
		Lanyards
		Halyards
		Deployment Motors
	ACS	Motors and Control Mechanisms
		Translating Masses
		Spreader Bars
		Mass Tip Mechanism
		Control Wiring
		Software
	Central Structure	Carrier Assembly
		Doors and Actuators
		Sail Drum
		S/C Interface

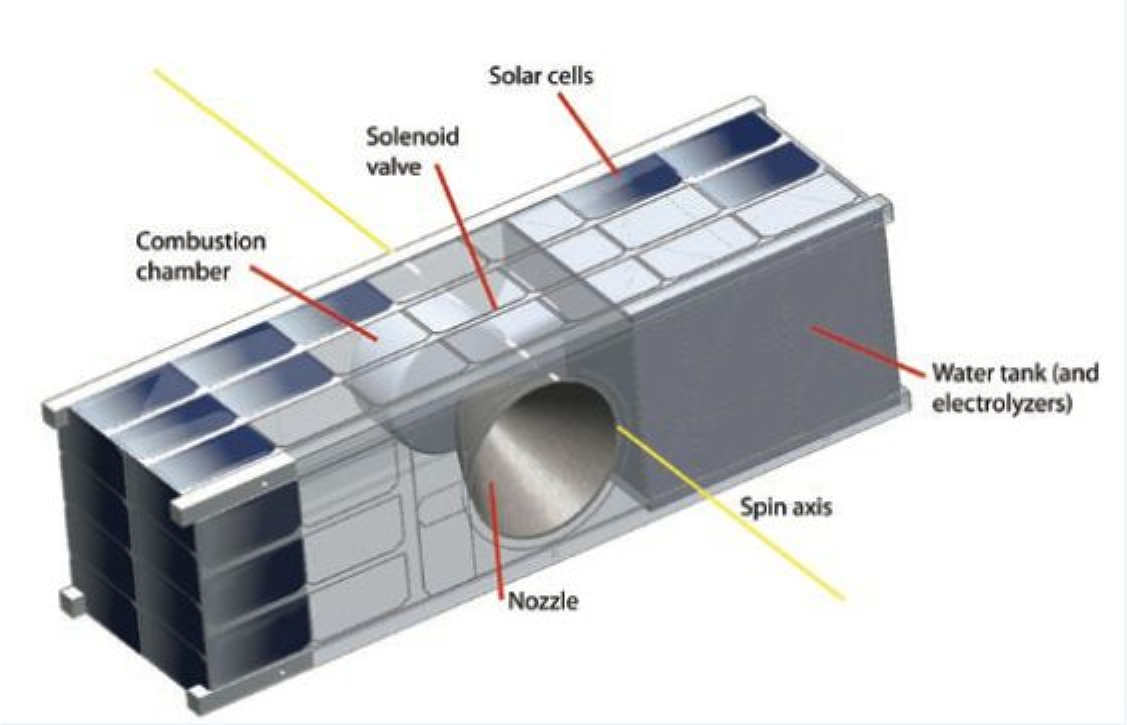
Photo Electrolytic Engine System Details

System Description –

During the summer College program at the Sibley School of Mechanical and Aerospace Engineering at Cornell University for Engineering attended by Hibah Mirza, there were specific discussions around the research into electrolysis propulsion systems. The research at Cornell University has led to a prototype that consists of three major components.

The water tank is where the liquid water propellant is stored, and also contains the electrolyzers that break the liquid water into its component gases of hydrogen and oxygen. Each burst of gaseous propellant is ignited inside the combustion chamber. Lastly, the gases are expanded through a nozzle, generating thrust.

A candidate arrangement of these components inside a 3U satellite bus is shown in Figure 1.



The satellite spins about an axis through the center of gravity (CG) and normal to the face that includes the thruster, as shown in Figure 1. The system is meant to occupy most of 2U, leaving at least one U in a3U satellite free for payload and other components. The propellant tank is placed in the outboard most section of the spacecraft, and is connected to the combustion chamber via an actuated solenoid valve.

The combustion chamber's axis of symmetry is aligned with the thrust axis. The nozzle is centered in the bus structure, directly in front of the combustion chamber. As the propellant is consumed, the center of gravity of the satellite shifts away from the side with the propellant tank, displacing the spacecraft's spin axis. The nozzle's axis of symmetry is placed midway between the spin axis at the start of the mission and the spin axis at the end of the mission to minimize the overturning torque due to thrust pulses, as discussed below.

Figure 2 shows a schematic of the energy and propellant flow between the different components of the propulsion system. Energy for the operation of the electrolysis propulsion system comes from solar panels.

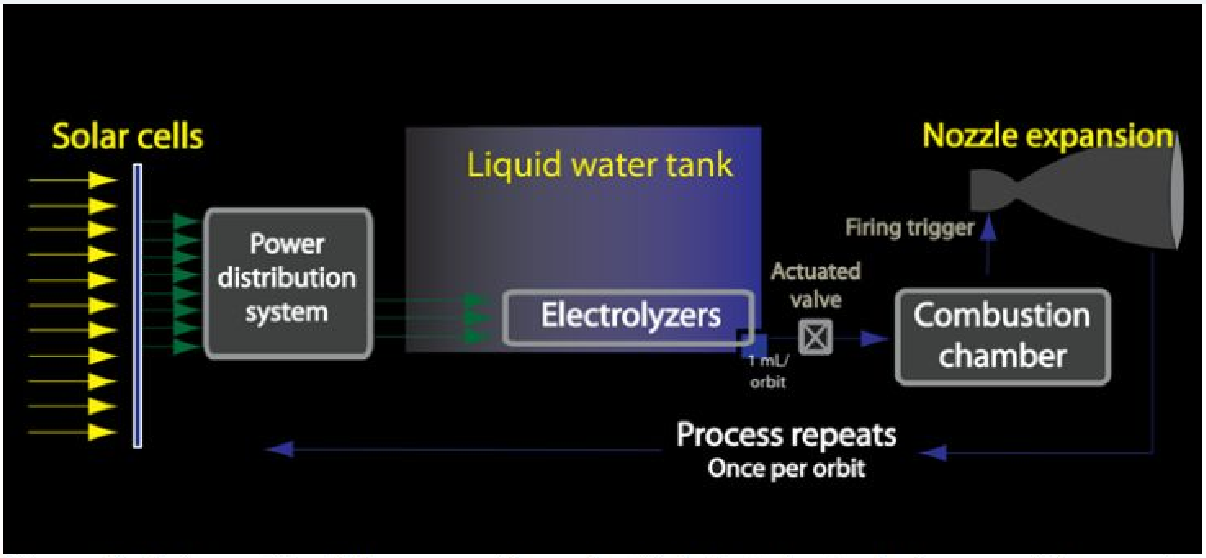


Figure 2. Schematic of the operation of a CubeSat electrolysis propulsion system.

Several concepts for deployable solar arrays for CubeSats exist, which provide power on the order of 30 W to 50 W.11 However, because electrolysis propulsion systems are flexible in their power consumption, this paper considers only the use of body-mounted solar panels. Electrical power from the solar panels is used to power several electrolyzers.

The specific heat of formation of water is -15.87 kJ/g, meaning that it takes 15.87 kJ to electrolyze a gram of water into its component gases. This required energy can be supplied quickly, drawing as much available power from the solar panels as possible, or slowly, leaving plenty of power for communications equipment, payload and other subsystems. The energy is effectively stored in the chemical potential energy of the electrolyzed gases.

This flexibility in the rate of electrolysis allows for a tradeoff between the power used for electrolysis and the time taken to electrolyze the water. This tradeoff means that electrolysis propulsion systems, unlike many electric thrusters, only require high power if the mission does not allow for extended periods of time between thruster firings.

Thrust Generation –

The electrolyzers inside the water tank of this prototype are commercially-available proton exchange membrane (PEM) electrolyzers. The platinum electrodes in these electrolyzers are separated by a proton exchange membrane that allows protons to cross and complete the circuit. PEM electrolyzers do not require dissolved electrolytes in the water; so, distilled water is used as propellant in the propulsion system.

The electrolyzers are powered only during the sunlit portion of the orbit. Electrolysis increases the pressure of the water tank as the amount of gas increases. Once the hydrogen and oxygen mixture reaches a high enough pressure a solenoid valve opens between the propellant tank and the combustion chamber. This event supplies the combustion chamber with a mix of hydrogen and oxygen ready to ignite. A small spark plug driven by a capacitive ignition circuit causes the gas mixture to combust. The gas then expands through a convergent-divergent nozzle, which produces the thrust necessary to impart a ΔV on the satellite.

The thruster relies on the successful separation of electrolyzed gases from the liquid water. To accomplish this propellant management, the spacecraft is in a constant spin about its major axis of inertia. This spin provides several advantages to the spacecraft. The centrifugal force caused by the spin separates the gas and liquid, causing the gas to accumulate toward the inboard side of the satellite, while the water is pushed to the outside edge of the water tank.

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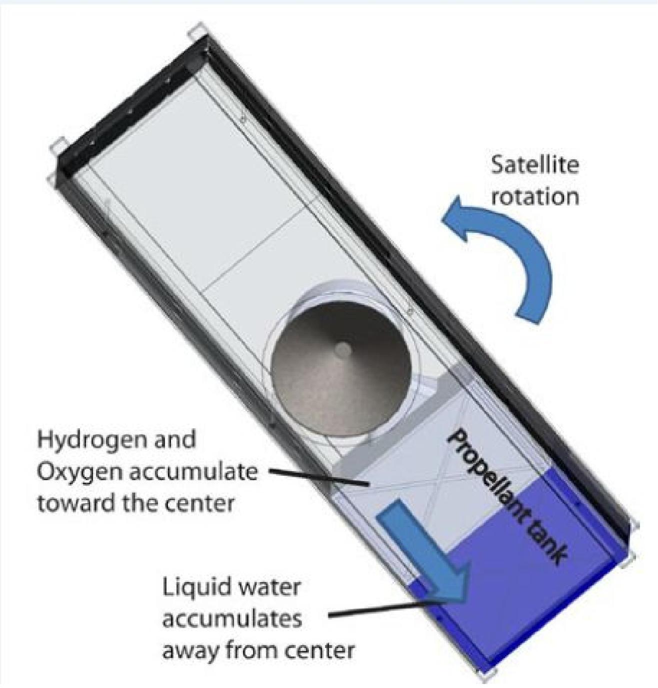
A diagram of this separation is shown in Figure 3. A spin set up by magnetic torquers at the beginning of the mission would also provide passive robustness to misalignments in the thruster and torques due to firing.


Satellite rotation
Hydrogen and Oxygen accumulate toward the center
Propellant tank
Liquid water accumulates away from center

Prototype Electrolysis Propulsion System at Cornell University –

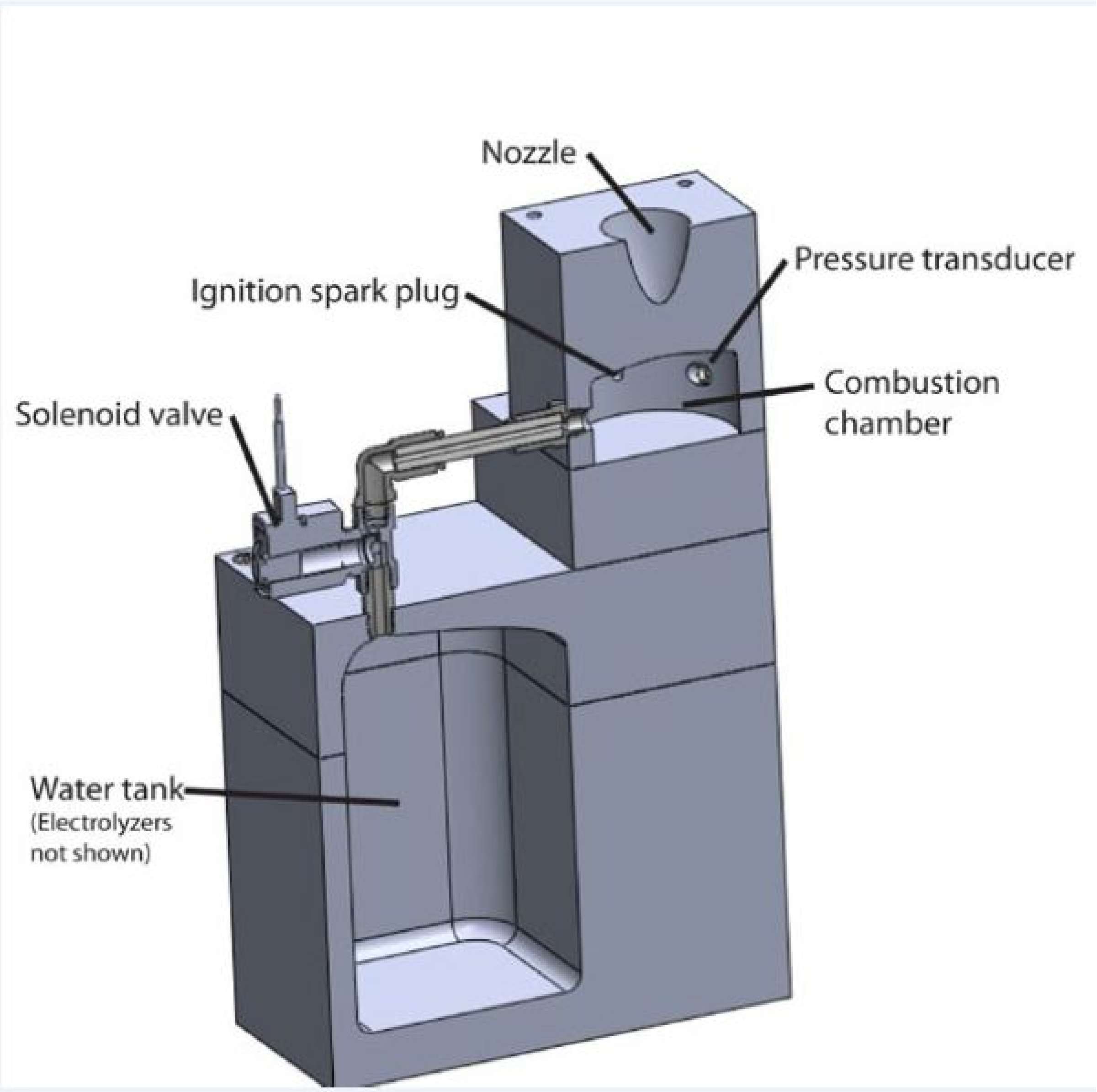
Successful characterization of the specific impulse and ▯V capabilities of an electrolysis propulsion system requires the testing of a prototype brass board system in a relevant laboratory environment. The prototype system developed at Cornell University's Space Systems Design Studio has the same inner dimensions that a flight version would have, and it is designed for testing in a vacuum environment. Testing in earth's gravity simplifies the operation of the thruster because there is no need to spin the propulsion system to provide the acceleration field that achieves gas separation.

The components of the brass board prototype are analogous to the components of the flight version described above. A single tank stores the liquid water propellant and electrolyzed gases. Three electrolyzers are installed inside the tank. One of these is shown in Figure 4 mounted in the propellant tank. Electrical power is supplied to the electrolyzers from an external power supply, through an electrical feed through installed in the tank. A pressure transducer mounted on the tank monitors the internal pressure of the tank to indicate when the pressure is high enough for a firing to occur.

PEM Electrolyzer

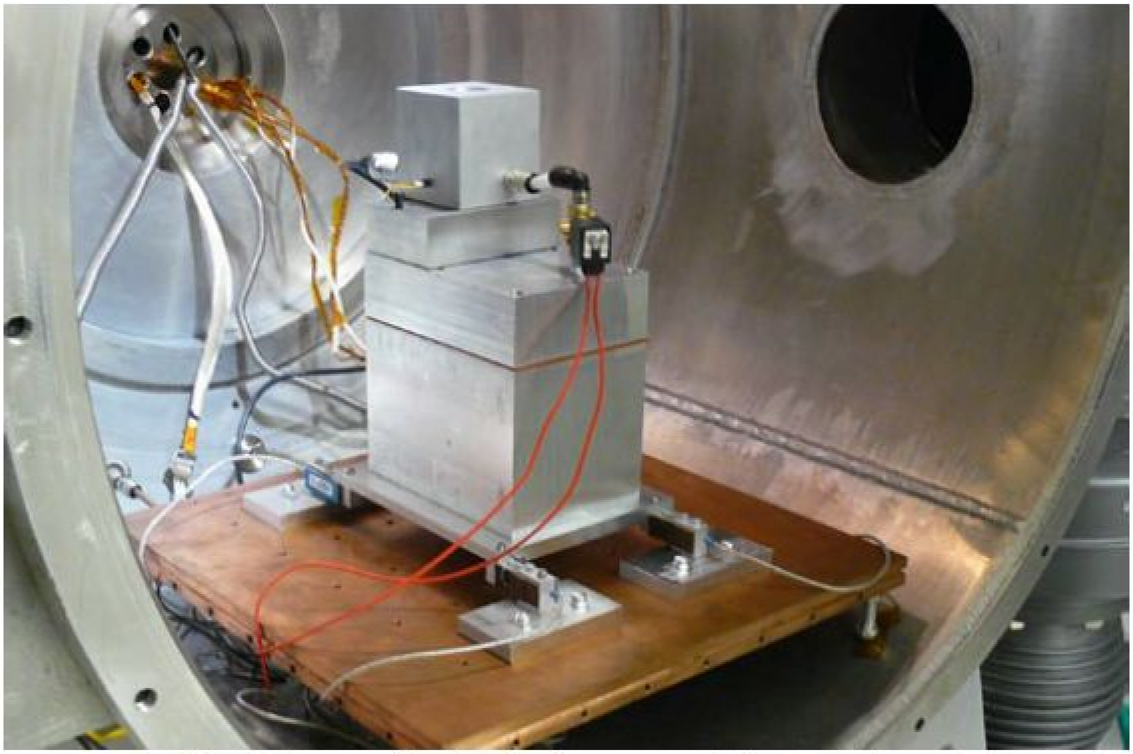
Gas is allowed to flow into the evacuated combustion chamber when the pressure is sufficiently high, above 10 bar for the test prototype. A solenoid valve controls the flow of gas into the combustion chamber. A miniature spark plug ignites the hydrogen and oxygen mixture inside the combustion chamber moments after the gas is initially allowed to flow into the combustion chamber.

A microcontroller sets the timing between the opening of the valve and the firing of the spark plug. This delay is optimized to produce the maximum ▯V per firing. The nozzle has the same internal dimensions as a flight nozzle but is built into a solid block of aluminum for simplicity and in order to provide a more rigid attachment with the combustion chamber and main tank assembly. Figure 5 shows a cutaway view of the prototype propulsion system, with the main components and sensors labeled.



The entire assembly is oriented such that the nozzle's axis of symmetry is perpendicular to the ground and so that firing causes a downward force. Force measurements are taken by four strain gauges mounted on a plate upon which the prototype assembly is set. Both force and the change in mass of the prototype are measured, to give a clear picture of both the thrust profile and total ΔV per burst. Force measurements are taken at millisecond intervals and the data is recorded through a data acquisition card outside of the thermal vacuum chamber. Pressure inside the combustion chamber is also monitored through a pressure transducer. Figure 6 shows the prototype and force measurement setup inside the thermal vacuum chamber. The system is analogous to a flight version, but designed for tests in a vacuum chamber.

Figure 6. Prototype electrolysis propulsion system on thrust measurement assembly in Cornell's thermal vacuum chamber

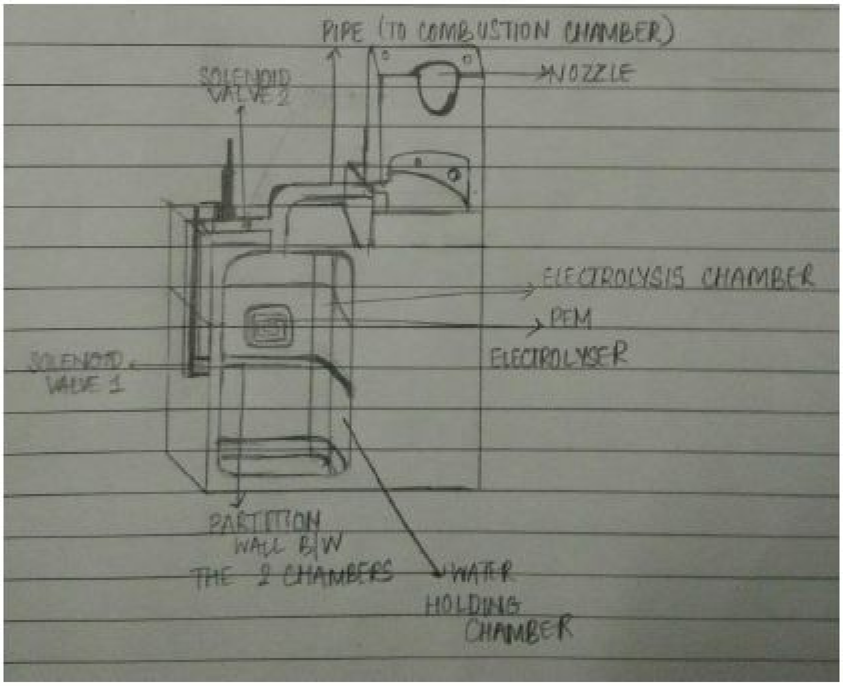


Prototype electrolysis propulsion system

Modifications to the prototype

A key problem with the original prototype is that it relies on spinning the CubeSAT on its primary axis in order to separate the electrolysed gases and the unelectrolysed water on the basis of density difference. The method is functional, but not desirable for a craft which has a solar sail attached to it. A key modification is that the holding-cum-electrolysis chamber has been divided into two, separating the electrolysis and water holding chamber. The two are connected by a solenoid valve which controls the amount of water in the electrolysis chamber. The electrolysis chamber has a pipe leading to the combustion chamber, its flow regulated by another solenoid valve. This removes the need for constant spinning in order to separate the electrolysed and non-electrolysed substances.

The modified engine design drawing is given below -



The drawing shows the modified design of the engine, outlining the boundary between the two different chambers, the two solenoid valves which control flow of (a) water to the electrolysis chamber and (b) the electrolysed gases to the combustion chamber. It also shows the location of the PEM electrolyser.