Hybrid Propulsion System for Space Tugs

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Abstract

Demand for Moon transfer applications has increased lately due to its future role in Mars missions and the Artemis program. A significant number of satellites are going to be delivered to lunar orbit for studies on the Moon. Since lunar transfer requires a high budget, only big space agencies have the capabilities to achieve this transfer. Using hybrid-propellant propulsion system decreases the costs significantly and allows more companies to reach the Moon. Therefore, a hybrid propulsion system is designed to deliver satellites from Earth orbit to Lunar transfer orbit using Translunar Injection (TLI). The proposed design will utilize a hybrid rocket engine in which paraffin and nitrous-oxide (N_2O) will be used as propellants and act as a space tug. The system has the capability of delivering 500 kilograms of payload from Geosynchronous Transfer Orbit to Lunar Transfer Orbit.
Keywords: hybrid, propulsion, satellite, tug, moon, spacecraft

Nomenclature

μ	Standard gravitational parameter of earth
V	Velocity
r	Radius
sma	Semi-major axis of an orbit
Isp	Specific impulse
$\dot{m}$	Mass
g_0	Gravitational constant at sea level
O/F	Oxidizer to fuel ratio
G_{ox}	Mass flux of oxidizer
A	Area
P	Pressure
t_b	Burn duration
C_D	Nozzle discharge coefficient
$\dot{r}$	Regression rate
a	Regression rate coefficient
n	Regression rate exponent
L	Fuel grain length
c*	Characteristic velocity
C_F	Thrust coefficient
η_{ch}	Combustion efficiency
η_{3D}	Nozzle 3D efficiency

Subscripts

c	Circular
p	Perigee or port
t	Total
d	Dry
prop	Propellant
fu	Fuel
ox	Oxidizer
p	Propellant
i	Initial or inner
f	Final

Acronyms/Abbreviations

HRE	Hybrid Rocket Engine
LEO	Low-Earth Orbit
CEA	Chemical Equilibrium with Applications
TLI	Trans-lunar Injection
GTO	Geosynchronous Transfer Orbit
LOX	Liquid Oxygen
N_2O	Nitrous Oxide
NASA	National Aeronautics and Space Administration

1. Introduction

Hybrid rocket engines have a design that combines the elements of both solid and liquid rockets. HREs have fuel in a solid phase and oxidizer in liquid or gaseous phase. These engines have several benefits over other types of rocket engines such as their intrinsic safety, design flexibility, reignition capability, and cost-effectiveness [1]. Hybrid rockets are inherently safer than both solid and liquid rockets because unintentional combustion cannot occur easily. The fuel grain is unreactive at room temperature without an added activation energy. Fuel and oxidizer types used in hybrid rockets are non-toxic materials [2].

This paper aims to perform a high-level design for a hybrid propulsion system that can deliver microsatellites from GTO to TLI orbit. SpaceX's Falcon 9 was selected as the launcher for its suitable performance in the range of lunar inclinations and lower cost than other commercial launchers. Various initial orbit scenarios are investigated and the optimal orbit is chosen considering performance requirements, cost, and multi-payload mission probability.

To begin the hybrid propulsion system design, paraffin and nitrous oxide are chosen as fuel and oxidizer, respectively. Necessary formulas for hybrid propulsion are derived and solved. Then, the Simulink model in MATLAB was created to perform sizing and performance evaluation of the propulsion system. Results are presented with performance improvement suggestions. Finally, conclusion and future plans regarding the design of the space tug is presented.

1.1. Space Tugs

Space tugs are specialized spacecrafts used for various purposes in space exploration and satellite servicing. One of the primary potential uses of space tugs is to transfer payloads from one orbit to another. This is often necessary for placing multiple satellites into different orbits from the same spacecraft. Launch vehicles deliver the payload to a predetermined orbit, from there each satellite must be boosted to its destination, whether that is a geostationary orbit or beyond. Delta-V capability of micro satellites is limited. Therefore, deploying them close to their operational orbit is critical. Space tugs are equipped with engines and thrusters that allow them to change their own orbit and carry a payload along with them.

As the small satellite industry advances and develops, companies are seeking specific orbits for their operations. However, the cost of Falcon 9 to launch a small satellite to a specific orbit is too high to be profitable. On the other hand, space tugs can carry multiple small satellites to their desired orbits in a single launch for a reasonable cost.

Hybrid rocket engines are well-suited for space tug applications due to their operational flexibility and intrinsic safety. Also, the ability of reignition allows them to change their orbit multiple times. Additionally, design simplicity make them a cost-effective option in the long run. These benefits make hybrid rocket engines a viable choice for propelling space tugs.

2. Lunar Transfer Tug Requirements

In designing a propulsion system for a lunar transfer spacecraft, selecting an appropriate launch vehicle to carry the spacecraft into its initial Earth orbit is essential. After considering various factors such as cost, performance range, and reliability, Falcon 9 was chosen as the launch vehicle for this study. The orbit of the Moon around Earth is elliptical and inclined with respect to Earth's equatorial plane. The inclination of the Moon varies between 18.4 and 28.6 degrees over the course of a year. Plane change maneuvers should be avoided to minimize the required fuel mass. Therefore, launch sites at latitudes between 18.4 and 28.6 degrees offer the best fuel consumption for orbital transfer to the Moon. The launch site of the Falcon 9, Cape Canaveral, with a latitude of 28.5 degrees, is also well-suited for this purpose as it can carry more mass into TLI orbit at the inclinations of interest required for a mission to the Moon [3].

Once the launch vehicle has been selected, it is necessary to design the propulsion system in a manner that is consistent with capabilities of the launch vehicle. Factors such as weight of the payload, altitude, delta-V requirement and overall mission objectives must be considered for the design of the propulsion system. Initial orbit and transfer strategy directly affects the delta-V requirement. Therefore, each scenario should be investigated, and the design should be based on the delta-V requirement. Various elliptical orbits are investigated, and appropriate GTO is chosen as the initial orbit.

2.1 Falcon 9 Capabilities

The performance capabilities of Falcon 9 (as shown in Table 1) are taken from the official manual [4]. In the manual, circular LEO capability of Falcon 9 is also listed. However, lunar transfer from circular LEO is too costly and having a perigee at high altitude is not beneficial in this case. Another disadvantage is that an inclination of 28.5 is generally not preferred for LEO launches and space tug should also be viable for rideshare missions. Thus, only maneuvers from GTO are considered. Elliptical orbits with different apogee altitudes are investigated and compared. These orbits are normally used for transferring payloads into geosynchronous orbits where communication satellites operate. Thus, they are commonly used in space applications. Since the transfer orbit is already elliptical, velocity requirements are lower compared to circular earth orbits. Payload capability of Falcon 9 to GTO with different inclinations is shown in Table 1.

Table 1. Falcon 9 Capability to GTO

Apogee Altitude (km) with 185 km perigee	**Payload Mass (kg)**				
	Inclination (deg)				
	19	**21**	**23**	**25**	**28.5**
10000	5447	5947	6391	6740	7002
20000	4374	4744	5060	5300	5471
30000	3853	4172	4442	4643	4784
35786	3660	3963	4216	4405	4536
40000	3550	3842	4087	4269	4394
50000	3351	3627	3857	4028	4144
60000	3212	3477	3697	3860	3969
70000	3109	3365	3578	3736	3841
80000	3029	3279	3487	3641	3742
90000	2966	3211	3415	3565	3664
100000	2914	3156	3356	3504	3601

2.2 Departure Orbit Selection

Required velocity change calculations are carried out using impulsive Hohmann transfer method for simplicity and ease of design [5]. Orbital transfer requirement for the space tug is that it should be able to raise its orbit from GTO to TLI orbit with a single powerful burn at the perigee. The difference between orbits is illustrated in Fig. 1 and Fig. 2. TLI orbit is in a free return trajectory such that the satellite should perform a maneuver inside the Moon's sphere of influence to enter its orbit.

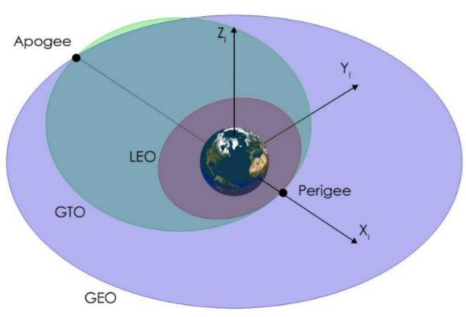

Fig. 1. LEO, GTO and GEO trajectories [6]

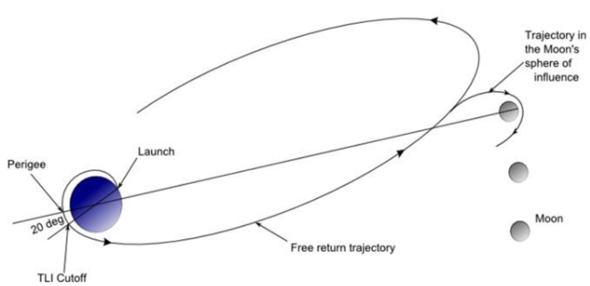

Fig. 2. TLI Orbit [7]

In GTO departure scenario, both initial and transfer orbits have elliptical shapes. Thus, the velocity at perigee is calculated using Eq. (1). Required velocity change is calculated by subtracting the initial perigee velocity from the final perigee velocity (see Eq. (2)). The apogee altitude of TLI orbit is decided as 380,000 km since the Moon's distance from Earth varies between 363,229 and 405,400 km. Thus, an apogee value that is equal to the mean is chosen.

$$V_p = \sqrt{\mu\left(\frac{2}{r_p} - \frac{1}{sma}\right)} \quad (1)$$

$$\Delta V_{GTO} = |V_{p1} - V_{p2}| \quad (2)$$

Lower delta-V requirement of the GTO scenario makes it a better candidate for the tug system. A shorter burn at the perigee in an elliptical orbit also decreases the thermal stresses on the spacecraft. Considering the goals of this study, the GTO departure case appears to be more appropriate compared to other initial orbits, not only in terms of mass but also in terms of other parameters such as cost, multiple payloads, in-space demonstration, and ease of operation.

By considering the potential for multiple payloads and remaining within a reasonable range for delta-V requirements, an elliptical orbit with a perigee altitude of 185 km and apogee altitude of 35786 km was selected as the departure orbit. As depicted in Table 1, this orbit also offers more than sufficient mass capability, with a capacity of nearly 4500 kg for Falcon 9 to carry. Fig. 3 shows the required delta-V to TLI orbit from orbits that have different apogee altitudes (calculated using Eq. (1) and (2)).

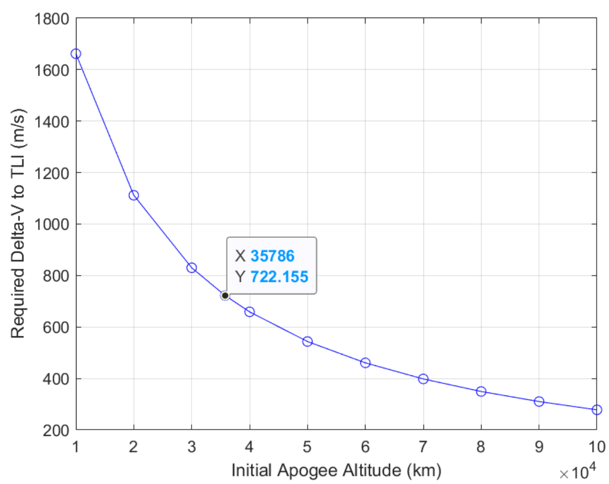

Fig. 3. Required delta-V for TLI maneuver from different initial orbits.

Transfer to TLI orbit from the chosen orbit requires 722 m/s delta-V. However, in practice, there are uncertainties such as mid-course correction, perturbations, and inefficiencies due to burn duration. The cost of mid-course correction burn during transfer phase is generally around 50 m/s [6]. With the addition of 100 m/s margin for other uncertainties, the total delta-V requirement for the propulsion system design can be estimated at around 870 m/s.

2.3 Physical Constraints

In addition to the main requirements derived from the launch vehicle, physical constraints must be considered. According to the Falcon 9 user manual, the dynamic envelope of the nosecone is fixed for every launch, and all payloads must be able to fit within the specified volume, as illustrated in Fig. 4.

Fuel grain design will be made using the equations given in Propulsion System Design section and by considering the constraints of the fairing. Since the fuel grain is in the center of the space tug, six N_2O tanks can be placed around the motor (which contains the fuel grain). The dimensions of the fuel grain and N_2O tanks are chosen in a way such that six microsatellites can be attached around the tug without exceeding Falcon 9 fairing limits.

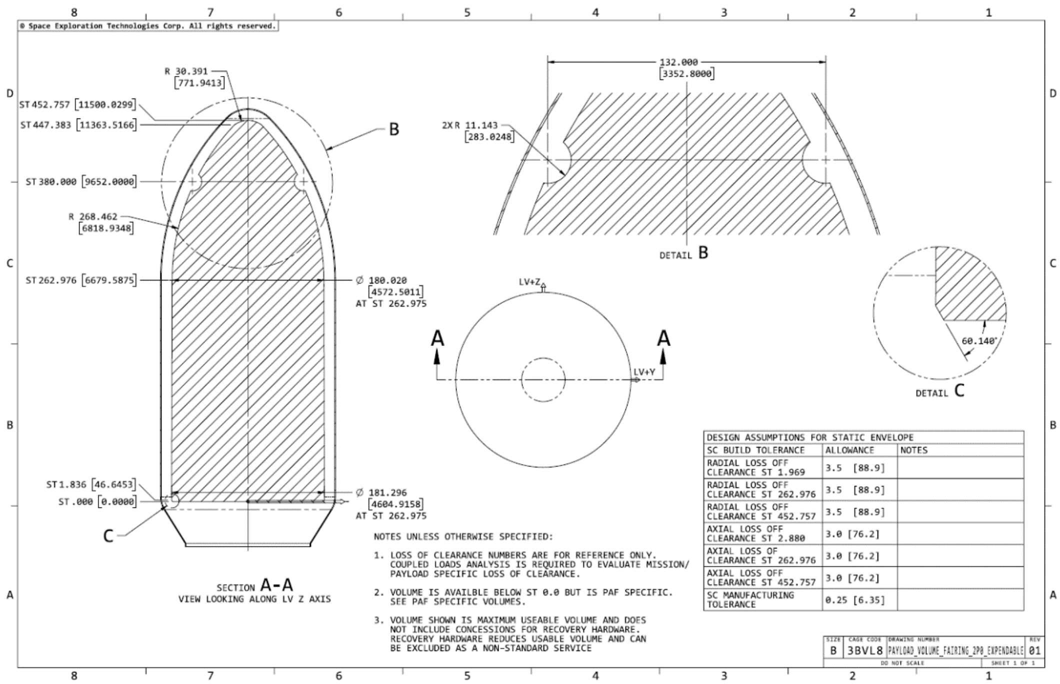

Fig. 4. Falcon 9 fairing dimensions, inches (millimeters) [4]

The largest diameter for a payload is 4.6 meters and the height for that diameter is 6.6 meters. Total height requirement for the payload is 11.4 meters. And the narrowest part of the envelope has a diameter of 1.3 meters. Eventually, the whole system must fit in this volume.

Placing a single space tug inside this envelope is inefficient since it would not cover the whole volume. Thus, stacking multiple space tugs vertically inside the fairing maximizes the number of satellites it can carry and decreases the cost significantly. Number of tugs that can fit the fairing can be decided after the design process.

3. Tug Design

In this chapter, a hybrid propulsion system will be presented as a solution to take the spacecraft to TLI orbit for the Moon after the separation from Falcon 9. The design process is iterative, starting with initial parameters and improvements are applied until mission requirements are met.

3.1 Propulsion System Design

Certain parameters should be estimated before the ballistic design of the propulsion system. The first constraint is delta-V, which is derived from orbital calculations. Utilizing the delta-V requirement (870 m/s) and giving an initial estimate for Isp make it possible to calculate total propellant mass. For the initial parameters, the total mass of the space tug is assumed to be 300 kg without propellant and payloads. The specific impulse is estimated to be around 280 seconds considering previous static fire test of hybrid rocket motors [9].

$$\Delta V = Isp g_0 \ln\left(\frac{m_t}{m_d}\right) \qquad (3)$$

Propellant mass is one of the major deciding factors in satellite design since total propellant comprises most of the mass. O/F ratio must be selected before calculating the mass of the oxidizer and fuel separately. Each fuel-oxidizer combination has different O/F ratios where they perform most efficiently. Thus, the selection of fuel and oxidizer should be made carefully. For fuel selection, paraffin has more heritage and is a well-known fuel for hybrid rockets. Therefore, it is chosen as fuel for the hybrid propulsion system. For oxidizer, liquid oxygen (LOX) and nitrous oxide (N_2O) are considered since they are commonly used in hybrid rockets.

Performance of the hybrid motor combustion of paraffin with LOX and N_2O are shown in Fig. 5. Characteristic velocity is the measure of the combustion performance of a rocket engine. Different from Isp, it is independent of the nozzle geometry and ambient conditions. Although LOX has higher c*, storage and thermal control of liquid oxygen in a satellite require a complex and an expensive system. Furthermore, LOX requires an additional pressurizing system which would increase total mass significantly. Thus, N_2O is chosen for its easy storability and self-pressurizing characteristics. O/F ratio of seven is initially decided for paraffin-N_2O combination since it offers high Isp and c* [9].

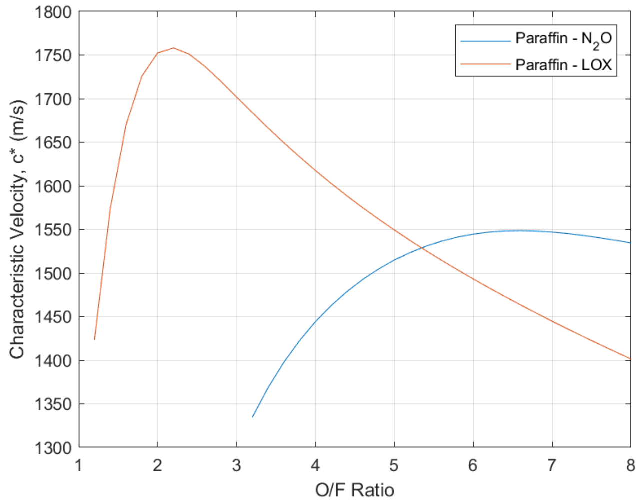

Fig. 5. Characteristic velocities of different propellant combinations

$$M_{fu} = \frac{M_{prop}}{\frac{O}{F} + 1} \qquad M_{ox} = \frac{O}{F} M_{fu} \qquad (4)$$

Before the sizing calculations of the propulsion system, burn time must be considered. Factors such as overheating, material endurance, and efficiency must be considered in determining burn time. Burn time should not be too long as this can damage the materials and also decrease orbital maneuver efficiency. However, short burn at high thrust can exceed the acceleration limit for the spacecraft. Therefore, a burn time of 300 seconds is chosen initially. Burn time is also important for calculating oxidizer mass flow rate.

$$\dot{m}_{ox} = \frac{M_{ox}}{t_b} \qquad (5)$$

After deciding burn time, the design of fuel grain and the fundamental understanding of regression rate should be discussed. Regression rate directly affects hybrid rocket's performance because it defines fuel's mass flow rate, which is highly correlated with thrust. As fuel burns, a thin film of liquid paraffin is created on the inner surface due to combustion temperature. During combustion, some of the liquid on surface vaporizes and some of the liquid droplets detach. The lost liquid is compensated continuously by melting from the surface [10].

The regression rate formula is shown in Eq. (6) where a and n are coefficients unique to each oxidizer and fuel combination. These coefficients are derived empirically and formularized using the data from firing tests with different durations [9]. The instantaneous flux (see Eq. (7)), is the oxidizer mass flow rate per unit area and is calculated by dividing the oxidizer mass flow rate by the instantaneous fuel grain port area.

$$\dot{r} = aG_{ox}^{n} \qquad (6)$$

$$G_{ox} = \frac{\dot{m}_{ox}}{A_p} \qquad (7)$$

According to Karabeyoğlu et. al. coefficients for paraffin-N_2O combination are given below [10].

$a = 15.5 \ x \ 10^{-5}$
$n = 0.5$

To find port dimensions of fuel grain, regression rate equation can be integrated over burn time using port area in terms of initial and final port radius.

$$\int_{r_i}^{r_f} dr\pi^n r^{2n} = \int_0^{t_b} am_{ox}^n dt$$
$$\pi^n \left(\frac{r_f^{2n+1}}{2n+1} - \frac{r_i^{2n+1}}{2n+1}\right) = am_{ox}^n t_b \qquad (8)$$

The optimum ratio between final and initial port diameter is taken as two because higher ratios cause fuel grain to crack during manufacturing process. Also decreasing the ratio alters the O/F since it will result in a longer fuel grain [11]. Hence, fuel grain inner and outer diameter is calculated using Eq. (9) using the ratio of two.

$$r_i = \left(\frac{(2n+1)am_{ox}^n t_b}{\pi^n(2^{2n+1}-1)}\right)^{\frac{1}{2n+1}} \qquad (9)$$
$$r_f = 2r_i$$

Length of fuel can be calculated by using Eq. (10) by using density, mass and dimensions of fuel grain.

$$L_{fu} = \frac{M_{fu}}{\rho_{fu}\pi\left(r_f^2 - r_i^2\right)} \qquad (10)$$

For design feasibility it is a reasonable to assume exit diameter of nozzle is equal to outer diameter of fuel grain (see Eq. (11)).

$$A_e = \pi r_f^2 \qquad (11)$$

Chemical Equilibrium Application (CEA) is a program developed by NASA to accurately calculate various chemical reactions [12]. It is also accurate for calculating performance parameters of a rocket engine combustion such as chamber pressure, characteristic velocity and thrust coefficient. A Simulink model has been developed in MATLAB to accurately simulate the hybrid propulsion system. CEA is integrated into this model to obtain the chamber pressure, thrust and Isp at each time step (see Eq. (13)). Simulation was run multiple times with different inputs to obtain optimum design.

In HREs throat area directly affects the chamber pressure inside the fuel grain. Considering previous hybrid motor tests, 30 bar is chosen as the initial pressure for the motor. By using Eq. (12), throat area can be calculated but the formula requires c* which is obtained from CEA. Thus, initially CEA function (see Eq. (13)) should be executed with the initially guessed parameters. Then, iteratively parameters must be estimated.

$$A^* = \frac{\dot{m}_{tot} c^* \eta_c}{P_{ch} C_D} \qquad (12)$$

$$f_{CEA}\left(\frac{O}{F}, P_{ch}, P_o, AR\right) = (c^*, C_F, Isp) \qquad (13)$$

Again, using Eq. (12), chamber pressure is calculated and iteratively updated until convergence.

$$P_{ch} = \frac{m_{tot} c^* \eta_{ch}}{A^* C_D} \qquad (14)$$

Since Isp is given by CEA and total mass flow rate is also calculated in the Simulink model, thrust can be calculated using Eq. (15).

$$T = m_{tot} Isp\ g_0 \eta_c \eta_{3D} \qquad (15)$$

Total impulse can be calculated by integrating thrust over time (see Eq. (16)).

$$I_{tot} = \int_0^{t_b} T\mathrm{dt} \tag{16}$$

$$Isp_{new} = \frac{I_{tot}}{M_p g_0} \tag{17}$$

Then, the more accurate specific impulse of the system replaces initially guessed Isp value and model is run iteratively until required delta-V is reached within a tolerance of 0.01 m/s.

$$\Delta V_{new} = Isp_{new} g_0 \ln\left(\frac{m_t}{m_d}\right) \tag{18}$$

$$tol = \Delta V_{req} - \Delta V_{new} \tag{19}$$

Analytical calculations above and CEA are integrated into the Simulink model. Also, Simscape toolbox has been used for accurately modeling N_2O pressure and flow rate throughout burn. Performance parameters derived from this model is shown in Table 2.

Fig. 6. Ground test of a hybrid propulsion system at DeltaV Test Facility [13]

Table 2. Parameters obtained from the model

Parameter	Value
Total Propellant Mass	300 kg
Spacecraft Dry Mass	300 kg
Payload Capacity	500 kg
Fuel Grain Mass	37.5 kg
Oxidizer Mass	262.5 kg
Initial Port Radius	0.104 m
Final Port Radius	0.208 m
Fuel Grain Length	0.400 m
Delivered Isp	280 sec
Delta-V Capacity	870 m/s
Total impulse	819 kNs
Average Thrust	2.73 kN

3.2. Conceptual Tug Design

Using Autodesk Inventor, a conceptual space tug design is created with the design parameters obtained from MATLAB model. Six nitrous tanks are placed around the motor which contains the fuel grain made by paraffin. Six microsatellites (each weighing around 80 kg) can be attached onto fairing of the tug as shown in Fig. 7. This layout allows the vertical stacking of tugs without disrupting the microsatellites. Five space tugs can fit inside the Falcon 9 fairing (see Fig. 4 and Fig. 8). However, due to mass constraints of the launcher putting four space tugs inside the fairing is feasible and considered for the study.

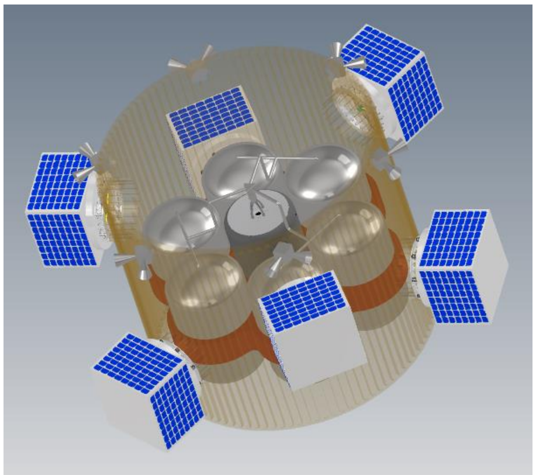

Fig. 7. Space Tug with six microsatellites attached

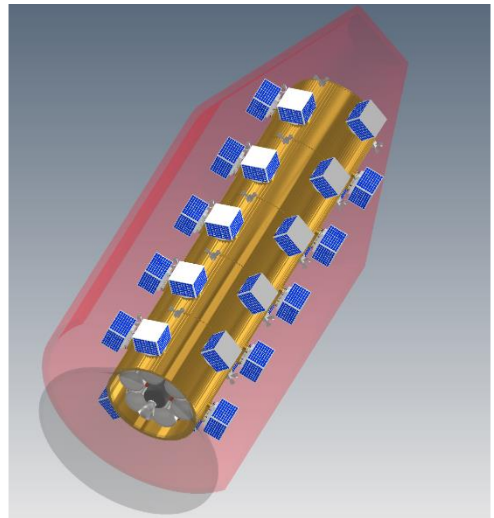

Fig. 8. Conceptual layout of stacked tugs in Falcon 9 fairing

4. Discussion

As the demand for the Moon missions increases, space tugs are becoming more viable due to their flexibility and cost-effectiveness. Hybrid propulsion offers a cheap, flexible and safe solution for space tugs. By carrying multiple microsatellites to specific orbits, space tugs will make it easier for small companies to reach the Moon. Since Falcon 9 can carry four tugs as payload, in total 24 microsatellites can be transferred to TLI orbit. Hybrid propulsion has not been tested in space yet. However, ground tests show promising results and with the space tug project it can prove itself [13]. The design is easily scalable since the regression rate is independent of the port diameter of the fuel grain. Space tug can be easily customized for delivering larger satellites.

4.1. Performance Improvements

Performance improvements can be done to increase specific impulse which increases payload capacity of the spacecraft. Increasing the specific impulse means that less propellant is required to achieve the same delta-V. Ground tests show that the addition of aluminum particles to fuel grain increases the overall Isp. According to Doran et. al., Isp increased around 10% by adding aluminum (20% of total mass) to the fuel grain.

5. Conclusions

In conclusion, a space tug with a hybrid propulsion system that is capable of delivering microsatellites from GTO to TLI orbit is proposed. Fuel grain parameters are obtained using analytical formulas and system simulation is created with MATLAB in Simulink. As a results, wet mass of the space tug is around 600 kg in which 300 kg is propellant. It has the ability to deliver 870 m/s delta-V with 500 kg payload which means that it can carry five microsatellites (each weighing around 80 kg). Decreasing payload mass increases the amount of delta-V it can deliver, thus with different payload configurations more orbital transfer options can be considered.

As for future work, detailed orbit analysis rather than Hohmann transfer can be implemented, and requirements should be defined in detail. On propulsion side, more improvement and different mixture of fuel oxidizer combinations will be investigated. Finally, mass budget with other subsystems will be included.

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