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Design Tradeoffs in Full Electric, Hybrid and Full Chemical Propulsion Communication Satellite

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Abstract

Full electric propulsion system becomes popular recently and getting more common in the communication satellite industry. Full electric or hybrid propulsion selection causes results in significant reduction in satellite propellant mass. Conventional full chemical satellite propellant mass is roughly 2/3 of launch mass so this huge reduction in propellant mass changes satellite design and preferences. Depending on satellite size and launch vehicle performance, full electric satellite requires 3-4 times less propellant compared to full chemical satellite and 1-2 times less propellant compared to the hybrid system. Satellite launch mass can be reduced by 40% for full electric propulsion and 15% for hybrid propulsion hence launch cost can be decreased by 40% and 15 % with the benefit of reducing propellant mass. It is possible to extend satellite communication capacity average 30 transponders for full electric and average 10 transponders for a hybrid system by using the benefit of propellant mass reduction. It takes 4-8 months to reach geostationary orbit for full electric satellite but in a chemical propulsion satellite, it takes a few days. Satellite subjects to additional radiation due to this long trip and hence additional aging. Expected revenue loss is another issue and conventional insurance policy needs an amendment to be in line with electric propulsion technology. The development of full electric satellite lowers the cost per transponder significantly.

Keywords: full electric satellite, full chemical, propulsion system, eps subsystem, satellite cost, satellite transponder

Haberleşme Uyduları İtki Sistemi Tasarımında Elektrikli, Hibrit ve Kimyasal Sistemlerin Ödünleşimi

Öz

Elektrikli itki sistemleri son yıllarda haberleşme uydularında kullanılmakta ve gittikçe yaygınlaşmaktadır. Elektrikli itki veya hibrit itki sistemi kullanıldığında uydu yakıt mikatrında ciddi azalma olmaktadır. Klasik kimyasal itki sistemlerinde uydu fırlatma ağırlığının ortalama 2/3 ünün yakıt olduğu düşünüldüğünde, yakıt kütlesindeki bu büyük azalma uydu tasarımını ve tercihleri değiştirmektedir. Elektrikli itki sistemleri, klasik kimyasal itki sistemlerine göre uydu büyüklüğüne ve fırlatıcı performansına bağlı olarak 3-4 kat daha az, hibrit uydulara göre 1-2 kat daha az yakıta ihtiyaç duymaktadır. Bu kazanım uydu fırlatma ağırlığının azaltılmasında kullanıldığında fırlatma ağılığı elektrik itkili uydularda %40 azalmakta, hibrit itkili uydularda %15 azalmakta dolayısıyla fırlatma maliyeti %40 ve %15 oranında azalmaktadır. Yakıt kütlesinden elde edilen bu kazanım, uydu haberleşme kapasitesinin artırılmasında kullanıldığında elektrik itkili uydularda ortalama 30 yansıtıcı (transponder), hibrit itkili uydularda ise ortalama 10 yansıtıcı daha ilave edilmesinin mümkün olduğu görülmüştür. Kimyasal itkili uydularda bir kaç gün olan transfer yörüngesinden yer sabit yörüngeye çıkma süresi, elektrik itkili uydularda 4-8 ay zaman almakta bu esnada maruz kalınan radyasyon uyduya ilave yapranma getirmektedir. Bu süre içinde beklenen gelirlerden mahrum kalınmakta ve klasik uydu fırlatma sigortası kapsamında yeni düzenlemelere ihtiyaç duyulmaktadır. Elektrik itkili uyduların geliştirilmesi ile transponder başına maliyet ciddi olarak düşmektedir.

Anahtar Kelimeler: elektrik itkili uydu, kimyasal itki, itki sistemi, uydu güç alt sistemi, uydu maliyeti, uydu yansıtıcısı

1. Introduction

Propulsion system provides a change in the velocity of a satellite. There are three main types of propulsion systems in the communication satellite industry; those are full chemical, hybrid and full electric propulsion systems. The most common type is the full chemical propulsion system that uses the chemical reaction to produce a flow of fast-moving hot gas. Apogee boost motor provides strong thrust using chemical reactions. Consequently, a satellite can reach from its initial GTO to GEO within few days. Chemical propulsion provides also the necessary thrust to keep spacecraft in the desired mission orbit or moving it to another desired orbit and provides torque to keep a spacecraft pointed in the desired direction. The hybrid propulsion system consists of chemical and electrical propulsion parts. The chemical part mainly provides thrust to reach from GTO to GEO, thrust for emergency, and de-orbit operations while electric propulsion part mainly provides thrust for station-keeping maneuver. A hybrid satellite can reach the GEO belt within a few days like a full chemical satellite. Electric satellite propulsion systems produce thrust by using electric to accelerate a spacecraft. This technology provides higher exhaust velocities and higher thrust efficiency than conventional chemical propulsion systems. Higher specific impulse reduces necessary propellant mass for the same ΔV of a chemical propulsion system. However, satellite power requirement increases due to the power consumption of electric thrusters. This significant decrease in propellant mass and an additional increase in electric power system) changes the design approach in communication satellites. A full electric satellite can reach from GTO to GEO between 4-8 months depending on their electric propulsion system performance [1, 2, 3]. Satellite subjects additional radiation while crossing Van Allen belt during long orbit rising duration.

Chemical propulsion systems store their energy in the propellants but the energy required by electric propulsion systems is generated by solar panels. Electric propulsion systems are not energy limited while chemical propulsion systems are energy limited because propellants have a fixed amount of energy per unit mass. In an electric propulsion system disregarding component lifetime considerations, an arbitrarily large amount of energy can be delivered via satellite electrical power system (EPS) to a given mass of propellant so that the exhaust velocity (or specific impulse) can be much larger than that available from a chemical propulsion system. Electric propulsion systems are said to be "power limited" because the energy supplied from EPS to the propellant is limited.

Power reqired to obtain a desired thrust in electric propulsion system is given in Equation 1;

$$P_t = \frac{TV_e}{2} \tag{1}$$

Where: P_t : power required for desired thrust V_e: exhaust gas velocity, T: thrust

The efficiency of thuruster is given as;

$$\eta = \frac{P_t}{P} \tag{2}$$

Where: P: power required for propulsion

Electric propulsion systems can be categorized according to the method used to accelerate the propellant as electro thermal, electrostatic, and electromagnetic. Electric propulsion systems generally make simultaneous use of two or even all three of these methods practically.

Table 1 Electrical vs chemical propulsion system key parameters and propellants

	Electrical	**Chemical**
Specific Impulse (s)	1200-3000	290-320
Thrust (N)	1-300 mN	10-400
Propellent	Xenon, NH_3 etc.	Hydrazine, N_2H_4 etc

Electric propulsion system specific impulse (I_{sp}) values typically 1200 – 2000 s but some thrusters can provide much more Isp value while chemical propulsion systems provide 290-320 s specific impulse as shown in Table 1. Those values are selected from typical thrusters' parameters that are currently used for the communication satellite industry commonly.

Communication satellites first injected to geosyncronous transfer orbit (GTO). GTO is highly elliptical earth orbit with apooge altitude of 35786 km. Satellite uses onboard propulsion system such as electric, hybrid or chemical to reach desired geosynronus orbit (GEO). GEO is a circular orbit 35786 km above earth (42165 km radius circle) and following direction of erath's rotation. Typical orbit rising from GTO to GEO is shown in figure 1.

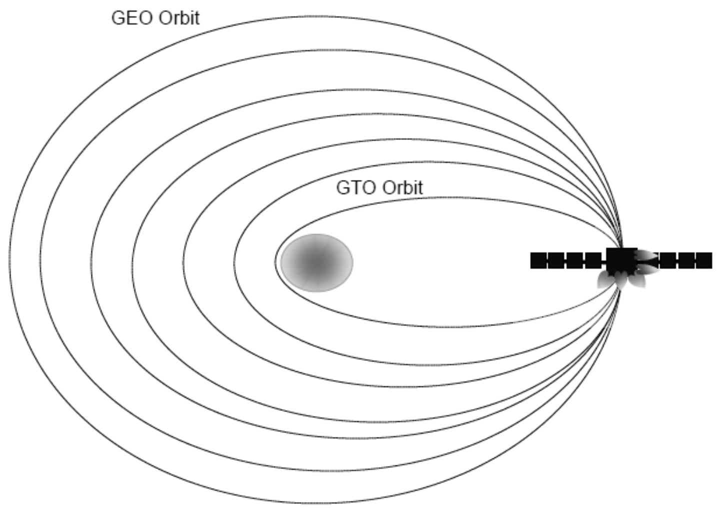

Figure 1 Orbit rising from GTO to GEO generic representation.

GTO has different perigee and inclination due to the launch vehicle performance and wet mass of a satellite. The following equations calculate the necessary ΔV to reach the final GEO. The required ΔV for inclination change is calculated as shown in Equation 3.

$$\Delta V = 2V sin(\frac{\Delta i}{2}) \tag{3}$$

The inclination change maneuver is combined with the orbital circularization (apogee) maneuver to reduce the total ΔV. The combined ΔV is the vectorial sum of the circularization maneuver and inclination change maneuver. The total combined ΔV to reach GEO can be calculated as shown in Equation 4.

$$\Delta V = \sqrt{V_{t,\alpha}^2 + V_{GEO}^2 - 2cos\Delta i V_{t,\alpha} V_{GEO}} \tag{4}$$

where $V_{t,\alpha}$: magnitude of velocity at the apogee, V_{GEO}: velocity in GEO 3074.7 m/s

It takes 4-8 months to pass through the Van Allen radiation belts where it has severe radiation dose. A satellite with a nominal shielding of 4mm of aliminium (Al) subject to additional radiation dose due to trapped electrons may be equivalent to up to 6.7 years of operations at GEO as shown in Figure 2 [4]. Orbit rising duration may be higher depending on the selected launch vehicle and launch strategy [5, 6, 7, 8]. Figure 2 highlights the importance of additional radiation protection. When electric orbit raising takes place, the operational considerations and the other risks from the Van Allen radiation belts must be considered. We should also discuss the new business opportunities for space insurance, and the need for space situation awareness in medium earth orbit.

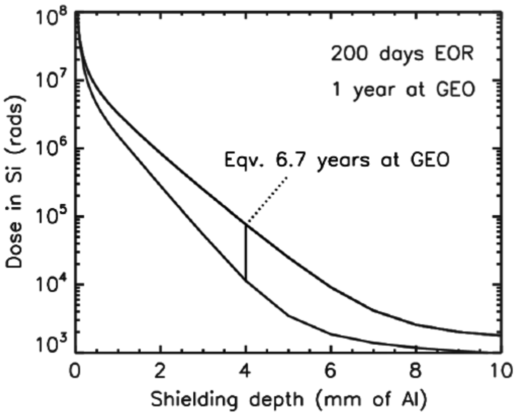

Figure 2 Comparison between accumulated radiation dose of electric orbit rising for 200 days and accumulated radiation dose of 1-year operation at GEO.

Propellant consumption in all operations of a satellite is a significant factor in trade-offs. To examine propellant consumption, relevant ΔV should be calculated. Table 2 shows the required ΔV to reach from GTO to GEO and ΔV budget for 20 years of operation in the orbit. We use three main commercially available launch vehicle performance, which are Falcon 9, Ariane 5 and Proton in this study to examine orbit rising ΔV. Satellites are controlled in the GEO with north-south station-keeping (NSSK) and east-west (E/W) maneuver. Those maneuvers ΔV requirements are calculated to find the total necessary ΔV. E/W maneuver ΔV depends on satellite longitude and the average annual requirement is1.9 m/s. Others column includes deorbit and wheel unloading ΔVrequirement for 20 years.

Table 2 Launcher performances and ΔV requirement to reach GEO and to operate a satellite for 20 years in orbit

Launcher	**Inc. degree**	**Perigee (km)**	**Apogee (km)**	**ΔV to GEO m/s**	**NSSK m/s**	**E/W m/s**	**Others m/s**	**Total ΔV m/s**
Falcon-9	28,5	185	35786	1837	976	38	19	2870
Ariane-5	6	250	35786	1490	976	38	19	2523
Proton	12	9800	35786	962	976	38	19	1995

The most propellant consuming maneuver is NSSK in GEO orbit. NSSK's annual ΔV varies with year because of the main perturbation sources orbital behavior. Figure 2 shows the necessary annual ΔV for NSSK for 20 years.

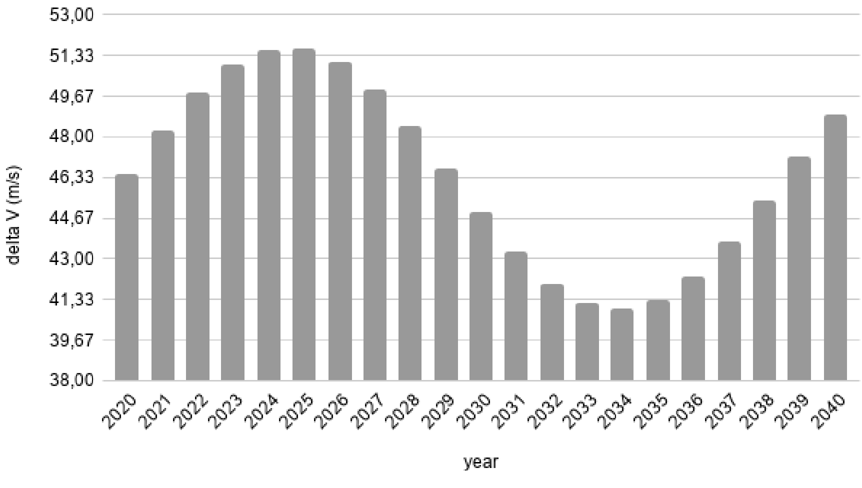

Figure 2 NSSK's annual ΔV between year 2020 and 2040.

2. Methods

This study examines the trade-off between power, mass, payload, reliability, and insurance implication in full electric, hybrid and full chemical geostationary communication satellites [9, 10, 11]. In the first case, we fixed satellite dry mass and launch mass then investigate to reduce the launch costs. In the second case, we fixed satellite launch mass by taking the full chemical system satellite as a reference then we examined satellite dry mass and relevant payload mass and their associated benefits.

Electric thruster PPS-5000 snecma with I_{sp} 1700 s and thrust 300 mN represents average electric thrusters in this study [12, 13]. This thruster is taken into account in calculations for full electric propulsion while a thruster with Isp 290 s and 320 s are taken for full chemical propulsion.

Satellite necessary propellent mass for the whole mission is key driving point in design trade offs.

We calculated propellent usage by using the rocket equation given in Equation 5.

$$m_1 = m_0 e^{\frac{-\Delta V}{g_0 I_{sp}}} \tag{5}$$

Where m_1: final satellite mass, m_0: initial satellite mass

A full chemical propulsion satellite mass breakdown shows that approximately 30% of satellite, total mass is payload mass, %30 is satellite electrical power system mass and 40% is structure mass as shown in Table 3. Propellent mass is around 2000 -4000 kg and total launch mass (wet mass) is around 3000-7000 kg [3]. Propellent requirement is approximately 1.5 times satellite dry mass for a full chemical satellite. Communication satellite laınch cost is between Eur 20000-30000 per kg. So to save launch cost, satellite wet mass (launch mass) must be treated. Full electric or hybrid system selection provides a reduction in the wet mass. Satellite power requirement increases in this case so that satellite EPS mass increases due to larger solar array, larger battery, larger capacity power conditioning unit (PCU) and power processing unit (PPU). So satellite dry mass increase for the same payload mission in full electric and gybrid satellite. There is a tradeoff between propellent mass reduction and satellite EPS mass increase.

Table 3 Mass breakdown of some chemical propulsion communication satellite

	NAME	Payload	Structure	Power	dry mass (kg)
1	Anik E	27,60%	43,70%	28,70%	1270,00
2	Astra 1B	30,00%	39,40%	30,60%	1178,60
3	Fordsat	28,90%	37,90%	33,20%	1094,00
4	HS 601	49,80%	30,90%	19,30%	1459,00
5	Intelsat VII	30,80%	42,70%	26,50%	1450,00
6	Intelsat VIIA	29,80%	39,80%	30,40%	1823,00
7	Olympus	28,50%	44,20%	27,30%	1158,00
8	Satcom K3	19,00%	45,40%	35,60%	1018,00
9	Telstar 4	30,00%	34,90%	35,01%	1621,00
	Average	30,49%	39,88%	29,62%	1341,29
	StdDev	8,07%	4,76%	5,02%	266,17

Full electric propulsion satellite additional power requirements start from 1500 W and reach up to 6000 W for communication satellite [12]. Electric propulsion needs 19.5 % mass increase in solar array mass and battery mass. Figure 3 shows battery mass and power relations and figure 4 shows solar array mass and required power relations. We calculated solar array size according to end of life performance of a satellite and considering two strings failure and 7.5% power margin and similarly battery size according to end of life performance of a satellite with 2 pack failure and maximum 75% depth of discharge. Those are insurance requirements of generic satellite. Triple junction solar cells and advanced lithium-ion battery whch are most commonly selected in the industry represent all solar arrays and batteries in this study for trade-offs.

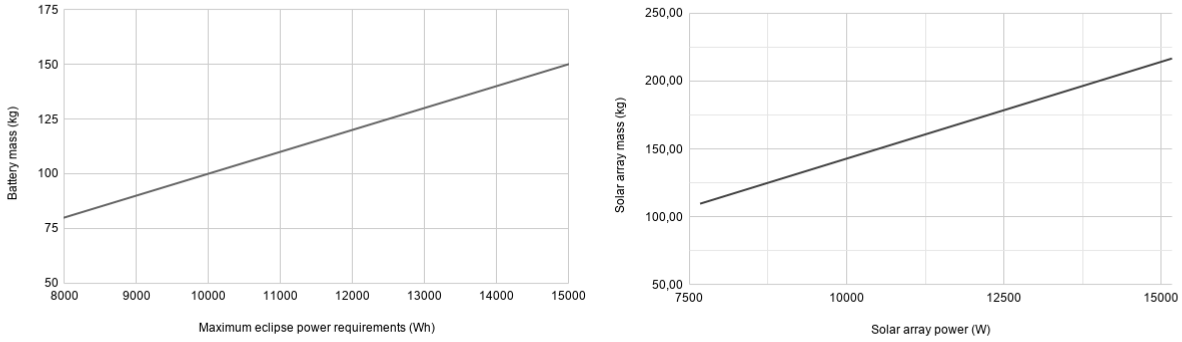

Figure 3 Battery mass as a function of maximum eclipse required power and Solar array mass as a function of required power.

We kept satellite maneuver life 20 years and obtained the satellite launch mass (wet mass) for three main launch vehicles for three sizes, compact, medium and large size satellite. We assume compact, medium and large size satellite dry masses are around 1500 kg, 2000 kg and 2500kg respectively.

3. Results and Discussion

In the first approach, we kept satellite dry mass and maneuver life fixed and we mentioned details above. The second approach is to keep wet mass, maneuver life fixed, and investigate dry mass. We took full chemical satellite the best launch mass as reference launch mass for three satellite sizes. Full electric propulsion satellites dry masses were evaluated accordingly for three-satellite size and three different launch vehicles. Table 4 shows full electric, hybrid and full chemical satellite wet mass for different launch vehicles with fixed dry mass and maneuver life. Falcon 9 maximum lift capacity for GTO mission is 5500 kg so Falcon 9 cannot launch large size satellite practically.

Table 4 Compact, medium and large size satellites wet mass in kg for fixed 20 years maneuver life and fixed dry mass

	Compact satellite, 8 kW payload power			**Medium satellite, 9.5kW payload power**			**Large satellite, 11 kW payload power**		
Launch Vehicle	**Full electric**	**Hybrid**	**Full chemical**	**Full electric**	**Hybrid**	**Full chemical**	**Full electric**	**Hybrid**	**Full chemical**
Falcon 9	1833	2981	3915	2470	4016	5218	3106	5051	6524
Ariane 5	1795	2659	3491	2419	3583	4655	3042	4506	5820
Proton	1739	2235	2935	2343	3011	3913	2947	3787	4891

There are enormous mass saving in electric propulsion systems. We obtained dry mass increase due to the equipment of electric propulsion systems, and solar array and battery mass increase as 2.5%, 4% and 5% for compact, medium and large size satellite respectively.

Table 5 shows that mass saving possible from 500 kg to 3400 kg dependeng on satellite dry mass size and selected launch vehicle. Satellite size have minimal effect in launch mass reduction. However, launcher performans have more importance on launch mass saving.

Table 5 Satellite launch mass (kg) reduction taking full electric satellite as reference for three sizes

	Compact		Medium		Large	
Launch Vehicle	**Hybrid**	**Full chemical**	**Hybrid**	**Full chemical**	**Hybrid**	**Full chemical**
Falcon 9	32,35%	47,93%	32,35%	47,93%	32,36%	47,63%
Ariane 5	25,74%	42,84%	25,74%	42,84%	25,74%	42,51%
Proton	14,40%	34,13%	14,40%	34,13%	14,40%	33,72%
Average	24,16%	41,63%	24,16%	41,63%	24,17%	41,29%

Electric propulsion systems provide a 40% mass reduction, which may be used for launch cost-saving or additional payload transponders.

This reduction in launch mass is equivalent to more than 30 transponders compared to full chemical propulsion. Hybrid propulsion systems provide around 15% mass reduction that is equivalent to more than 10 transponders as shown in Table 6. A satellite operator can decide taking its business and strategic plan into account to select launch cost saving or having additional transponders.

Table 6 Satellite fixed wet mass and fixed 20 years maneuver life, gained mass for additional payload mission in kg

	Compact satellite (kg)		Medium satellite (kg)		Large satellite (kg)	
	Full electric	**Hybrid**	**Full electric**	**Hybrid**	**Full electric**	**Hybrid**
Initial dry mass	1532	1532	2065	2064	2596	2596
Dry mass for Falcon9	3273	2012	4362	2682	5454	3354
Dry mass for Ariane5	2980	2012	3973	2682	4967	3354
Dry mass for Proton	2586	2012	3447	2683	4309	3353
Room for payload mass Falcon 9	522	144	689	185	857	227
Room for payload mass Ariane 5	434	144	573	185	711	227
Room for payload mass Proton	316	144	415	186	514	227
Average payload room mass	424	144	559	186	694	227
Number of additional transponder	22+	8+	29+	10+	37+	12+

Satellite launch mass depends on types of propulsion systems and launch vehicle orbital injection performance. Table 6 shows that for evaluated satellite sizes Proton provides the best performance. Ariane 5 gets the second rank and Falcon 9 gets the third rank.

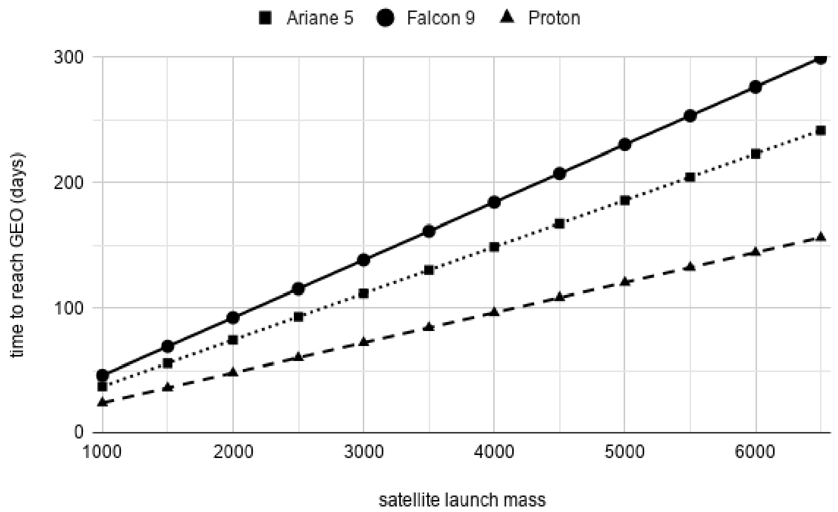

Figure 4 Time to reach GEO for different launch vehicle as a function of satellite wet mass

Satellite wet mass, GTO inclination, and perigee altitude and electric thruster performance are the main driving parameters to evaluate time to reach desired GEO. Figure 4 shows time to reach GEO as a function of satellite wet mass. Heavier satellites need more time to reach GEO as expected.

The major disadvantage of full electric satellite is time to reach GEO. The traditional space insurance policy covers space asset, which is a newly launched satellite but does not cover full electric propulsion satellite some failure scenarios. For example, reducing the level of redundancy and increasing the duration of orbit rising may recover a failure. The increased duration of orbit raising results in a financial impact and revenue loss, which is not covered by traditional satellite insurance policy [14]

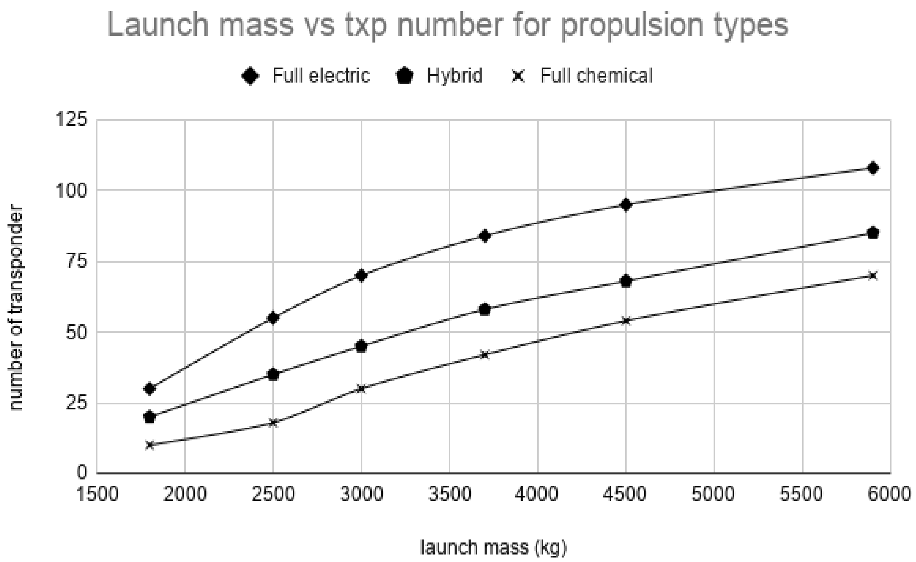

Figure 5 Satellite launch mass vs number of transponders for full electric, hybrid and full chemical satellites

Figure 5 shows the graphical representation of the number of transponders as a function satellite launch mass. In the evaluation, we take the average value of the launch vehicle's performance into account. Similarly, satellite dry mass is selected as the average weight of the most available satellite buses. Large size full electric satellites have more benefits than full electric compact size satellites. A satellite operator can decide to lower the launch cost or to have additional transponders according to its business and strategic plan.

4. Conclusion

Full electric propulsions are more efficient than chemical systems. They require significantly less propellant to generate the same thrust. The total mass of the full electric satellite is approximately half of the traditional chemical propulsion system. Due to lower mass, operators may employe a dual launch configuration. Satellite operators have the option of tailoring their spacecraft to be more lightweight allowing the satellite to be launched on smaller rockets. Launching two full electric communication satellite on a Falcon 9 rocket put launch costs for each satellite at about $30 million as of today's market price. This is 1/3 even ¼ of dedicated launch. Electric trusters regulate the force applied to the satellite very accurately, making it possible to control satellite position and orientation along its orbit with improved precisions. Therefore, during the execution of the maneuver in the GEO, signal variations will be reduced.

Full electric satellite offers potential disruptive cost savings that have implication across the value chain. The development of a full electric communication satellite lowers the cost per transponder of a satellite.

Acknowledgments

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