The Benefits of Continued Advances in the Propulsive Capability of the Electric GEO Communications Satellite

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Abstract: The implementation of electric propulsion systems on Earth orbiting satellites has dramatically improved the fuel efficiency, and in turn the capability and resilience, of such satellites when compared to their chemical propulsion counterparts. Electric propulsion systems flown to date have achieved power levels up to 5 kW per string for GEO communications satellites enabling electric orbit raising, orbit positioning, and station keeping functions. In this paper, the GEO mission capabilities and user benefits of further advances in the operating power and performance of electric systems are investigated. It is shown through mission analysis that a modest improvement in performance over the current flight systems results in a substantial improvement in mission and launch vehicle flexibility. The benefits of such an improvement are quantified with a comparison between reference missions that utilize existing commercial electric systems, and those that utilize an electric system with an incremental advancement in performance. The substantial mission improvements documented in this paper justify the need for continued advances in electric propulsive capability.

Nomenclature

AR	=	Aerojet Rocketdyne
GEO	=	Geosynchronous Equatorial Orbit
GSO	=	Geostationary Orbit
EELV	=	Evolved Expendable Launch Vehicle
EP	=	Electric Propulsion
FPR	=	Flight Performance Reserve
Isp	=	Specific Impulse
LEO	=	Low Earth Orbit
LV	=	Launch Vehicle
P	=	Power

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PPU = Power Processing Unit
T = Thrust
ULA = United Launch Alliance
W = Weight

I. Introduction

FOR several decades, the promises offered by electric propulsion (EP) systems have motivated their continued development. Prior works have often compared the advantages of an EP system with the capability of their chemical propulsion counterpart. For missions that can tolerate an increase in maneuver duration and monitoring, the benefits of using an EP system over a chemical system for increased payload capability, increased mission life, or reduction in spacecraft wet mass are now presumed.

Based on thruster technical readiness and spacecraft capability there has been a multi-path approach in the adoption of different EP technologies for GEO communications satellites[1]. For example, the first EP systems flown for commercial purposes were based on resistojet technologies and operated at powers of 300-660 W[2]. As additional technologies matured and bus powers increased, 1.6 kW and then 2 kW arcjet systems gained acceptance on commercial satellites for North-South-Station-Keeping[3]. Gridded-ion and Hall thruster technologies followed with increased operating powers between 1.5 kW and 4.5 kW. Today, all four thruster technologies are flying on GEO spacecraft, which have themselves achieved available bus powers of 25 kW and greater than 6,000 kg wet mass[1,4,5].

The application of EP technologies for GEO communications satellites has also continued to expand. Once used primarily for station keeping functions, the past decade has shown that EP systems are now routinely considered for orbit raising and maneuvering as well[1]. Space vehicle integrators maintain all-electric propulsion system variants which rely on EP systems for orbit transfer and insertion, station keeping, and de-orbit maneuvers.

This paper considers the current state-of-the-art EP systems and space vehicle capabilities for medium to large GEO communications satellites, and seeks to illustrate through mission analysis the advantages of further incremental improvement both in propulsion system performance, and in spacecraft power capability. The rationale for emphasizing specific technologies is discussed, and the analysis approach is described.

II. Analysis Approach

The scope of the analysis performed considered a parametric study of GEO satellite wet mass, insertion orbit, available bus power, launch vehicle type and site, and propulsion system performance. Existing capabilities were identified based on a survey of medium to large communications satellites commissioned or launched in the past five years, and near term future capabilities were inferred from recent trends in the industry. An initial baseline mission was set upon from which to parameterize satellite wet mass, and other parameters were adjusted incrementally to demonstrate the effect of propulsion system performance on each mission parameter. The baseline mission assumed electric orbit raising, station keeping, and de-orbit maneuvers would be performed by the on-board electric propulsion system similar to current all-electric vehicles. Two key outputs were sought: the effect of adjusting a given parameter on the time of flight experienced during the electric orbit raising phase of the mission, and the resulting effect of the same adjustment on satellite payload capability (identified as dry mass). Additional descriptions of the specific parameter values chosen are provided in the following sections.

A. Space Mission Architecture

Table 1. Evolved Expendable Launch Vehicle (EELV) Reference Orbits[6].

Orbit Description	Apogee Altitude (nmi)	Perigee Altitude (nmi)	Inclination (deg)	Mass to Orbit (lbm)
LEO	500	500	63.4	15,000
Polar 1	450	450	98.2	15,500
Polar 2	450	450	98.2	37,500
MEO Direct 1	9,815	9,815	50.0	11,750
MEO Transfer 1	10,998	540	55.0	9,000
GTO	19,323	100	27.0	18,000
Molniya	21,150	650	63.4	11,500
GEO 1	19,323	19,323	0.0	5,000
GEO 2	19,323	19,323	0.0	14,500

To fully illustrate the time of flight and payload impact of using an improved higher-powered EP system for geostationary orbit (GSO) satellite insertion, the baseline analysis inserted an all-electric satellite into the lowest Evolved Expendable Launch Vehicle (EELV) reference orbit possible, as seen in Table 1, (500 x 500 nautical miles or ~900 x ~900 km) in low earth orbit (LEO). For the baseline analysis the SpaceX Falcon 9 was launched from Cape Canaveral with a 28.5° inclination. The advantages of this approach are twofold. First, by launching into a low energy orbit, one is able to fully take advantage of EP's high specific impulse (compared with the launch vehicle's 2nd stage chemical engine performance) and eliminate from the analysis mass limitations of having to choose between specific launch vehicle providers. For the baseline insertion orbit analysis, initial satellite wet masses between 3000-7000 kg fell well within the ~13,600 kg insertion capability of the Falcon 9 using the autonomous spaceport drone ship (ASDS). However, for completeness, the sensitivity to commercial launch vehicle selection was investigated by fixing the satellite mass and having the launch vehicle insert the satellite into the highest energy orbit possible. All SpaceX Falcon launch vehicle performance was modelled based upon information from NASA's Launch Services Program (LSP) office. Seen in Figure 1, United Launch Alliance's (ULA's) Vulcan Centaur 2-solid rocket booster configuration performance was extrapolated using a logarithmic curve typical of launch vehicle performance between LEO to GTO. Ariane 6 payload to orbit capability came from the official user's guide[7].

Vehicle / Orbit *	Vulcan Centaur (2-solid)	Vulcan Centaur (6-solid)	Vulcan Centaur Heavy (2023)
LEO ER 28.7° (Ref.) 200 km circular	17800 kg [39200 lb]	27400 kg [60300 lb]	34900 kg [76900 lb]
LEO ER 51.6° (ISS) 407 km circular	15300 kg [33800 lb]	24200 kg [53400 lb]	31400 kg [69300 lb]
LEO WR 90° (Polar) 200 km circular	14300 kg [31500 lb]	22300 kg [49200 lb]	27900 kg [61500 lb]
GTO 1800 m/s 35,786 km x 185 km @ 27.0 deg	7400 kg [16400 lb]	13300 kg [29300 lb]	16300 kg [35900 lb]
GEO 35,786 km circular @ 0 deg	2050 kg [4500 lb]	6000 kg [13200 lb]	7200 kg [15900 lb]

Figure 1. United Launch Alliance's Vulcan Centaur Performance[8].

The second advantage of selecting a 900 km circular orbit was that it is high enough to eliminate atmospheric drag and re-entry concerns typical of electric satellites with low thrust-to-weight (T/W) and corresponding low acceleration. Previous Aerojet Rocketdyne (AR) mission analysis for a 3.5 kW EP system under no horizontal winds (rigid atmosphere) and the 1976 Standard Atmosphere model (ideal, steady-state model at a latitude of 45°N during moderate solar activity) showed re-entry concerns below 350 km depending upon assumptions for drag coefficient, photovoltaic array, and the solar cycle.

B. Space Vehicle Configurations

Two vehicle design configurations were studied, which used either a baseline 25 kW or improved 31 kW all-electric class bus. For simplicity of the analysis, a high-power throttle point of a given propulsion system was used for the LEO-to-GSO transfer trajectory. This information also aided the determination of the number of thrusters used during the orbit transfer when Power Processing Unit (PPU) efficiency and power reserved for the bus during transfer were factored against the power constraints of the bus (i.e. 25 kW or 31 kW). For all analyses, 1.0 kW was reserved for the bus during transfer. A low-power, relatively high specific impulse throttle point was used for station keeping analysis. The study assumed a 17 year mission with North-South and East-West stationing keeping requiring 61 m/s per year.

Dry mass presented in the results is a careful accounting of the wet mass used over the entire 17 year mission. The results assume a fixed initial wet mass in the stated insertion orbit which were decremented by transfer propellant consumption from trajectory analysis utilizing NASA's SEPSPOT software, and Tsiolkovsky's rocket equation for the station keeping. The authors further assumed Xenon residuals were 2.5% of the total loaded Xenon propellant with a flight performance reserve (FPR) based upon 5% of the total mission delta-v.

C. Description of Propulsion Capabilities

A number of EP systems were considered for analysis. For the purposes of this paper, it must be assumed that, for medium to large GEO commercial satellites, electric orbit raising, station keeping, and de-orbit functions are becoming more commonly performed by two to four thrusters per vehicle, and as such, only those systems with the impulse capability to perform all-electric missions were considered. Additionally, multiple competing suppliers exist for similar delivered product capabilities in the Hall thruster and gridded-ion thruster markets. It is the intent of the authors to assess the mission effects across various classes of systems. So, for multiple competing products a common performance table is estimated which represents the class of thrusters as they might be operated for GEO type applications, which have trip time considerations. For example, the QinetiQ T6[9] and L3 Harris XIPS 25 ion thrusters[10] are represented by a single 4.5 kW-class Ion Thruster, as noted in Table 2. Similarly, a single 4.5 kW-class Hall thruster system is included[11,12,13,14,15].

An assessment of vehicle design optimization was not performed here, but instead it was assumed that maintaining a four-thruster-per-vehicle approach is preferred. This assumption is not without merit, as additional thruster strings require more mass, more vehicle integration, and likely more total vehicle cost. Given the considerations mentioned above, this results in a representative maximum power per thruster string of 6 kW for a 25 kW vehicle, and 7.5 kW for a 31 kW vehicle. Accounting for efficiency losses in the power processing unit, 5.7 kW and 7 kW would be available to the input of each thruster, respectively. Because station keeping maneuvers can be adjusted in terms of duration and frequency, and payloads may be active during station keeping, operating two thruster strings between 2 kW to 3.2 kW each for station keeping remains consistent with existing thruster capabilities and was presumed for a potential improved propulsion system as well.

A survey of near term gridded-ion engines in the 5 kW to 7 kW range results in finding the NASA NEXT-C thruster[16] and the ArianeGroup RIT 2X[17]. For its potential to operate with relatively high thrust-to-power (51mN/kW), the NEXT-C thruster is used to represent the evolutionary capability provided by the technology for comparison purposes with an understanding that higher Isp ion thrusters are possible at the added expense of orbit transfer durations for GEO missions. The projected performance of the NEXT-C thruster is shown in Table 3.

Currently there are no known flight Hall thruster systems which operate up to 7.5 kW. Such a system could be designed to emphasize high thrust to power, or high specific impulse, or a balance between the two. Based on previous Aerojet Rocketdyne development activities and an understanding of the notional performance range of Hall thrusters[18,19] each bounding case is considered. This is accomplished by approximating Hall thruster thrust-to-power values measured or projected and scaling for thruster input discharge powers of 7 kW and 3 kW to obtain thrust and specific impulse, shown in Table 4.

Table 2. Representative state of the art 4.5 kW thruster performance table[10,11].

Thruster	Thruster Input Power (W)	Thruster Input or Beam Voltage (V)	Thrust (mN)	Isp (s)
4.5 kW-class Hall	3000	400 (input)	168	1850
4.5 kW-class Hall	4500	300 (input)	282	1800
4.5 kW-class gridded ion	2067	1215 (beam)	79	3400
4.5 kW-class gridded ion	4215	1215 (beam)	165	3500

Table 3. 7 kW-class gridded ion (NEXT-C) thruster projected performance table[16].

Thruster	Thruster Input Power (W)	Beam Voltage (V)	Thrust (mN)	Isp (s)
NEXT-C	2880	1567	99.9	3910
NEXT-C	6735	650	338	2640

Table 4. Notional bounds of a 7 kW-class Hall thruster designed for either high thrust or high Isp.

Thruster	Thruster Input Power (W)	Thruster Input Voltage (V)	Thrust (mN)	Isp (s)
7 kW-class Hall (High Thrust)	3000	125	204	1100
7 kW-class Hall (High Thrust)	7000	250	525	1580
7 kW-class Hall (High Isp)	3000	500	144	2350
7 kW-class Hall (High Isp)	7000	500	364	2550

D. Summary of Mission Analysis

As aforementioned, NASA's SEPSPOT software was used for the low-thrust transfer analysis. The program outputs the propellant consumed, time of flight, array degradation (due to Van Allen belt transit), and the ideal delta-v when performing a low thrust transfer between two closed conics[20]. In this case, it was the transfer from the launch vehicle insertion orbit (e.g. 900 km x 900 km at 28.5°) to a standard GSO (35,786 km x 35,786 km at 0.0°). The program provides medium fidelity results as it solves the optimal control problem by an orbital averaging integration method and calculus of variations to find minimum-time transfers[21,22].

System string efficiency is a required input into SEPSPOT and was calculated. Efficiency of the thruster is calculated from knowledge of the thruster operating power, specific impulse, and thrust. PPU efficiency of 92 – 95% is obtained from market analysis of American[23] and European[24] commercial PPU solutions for existing 5 kW systems. Together both efficiencies are used to calculate the system string efficiency for trajectory analysis.

III. Results and Discussion

Analysis results are shown below in two parts which first capture the improved capability offered by increases in the propulsion system performance, and second by improvement in the bus power available. The baseline 25 kW mission is shown across varying satellite wet mass for each propulsion system identified. The sensitivity of insertion orbit is illustrated for the bounding high-thrust and high-Isp 7 kW-class Hall thruster. Also, the effect of varying launch vehicle and site is shown for the same 7 kW-class Hall thruster bounding conditions. Second, the effect of a higher power capable 31 kW bus is shown. A final section discusses the practical outcome and user benefits of the capabilities presented in the first two parts.

E. Current (25 kW) Satellite Bus Capability

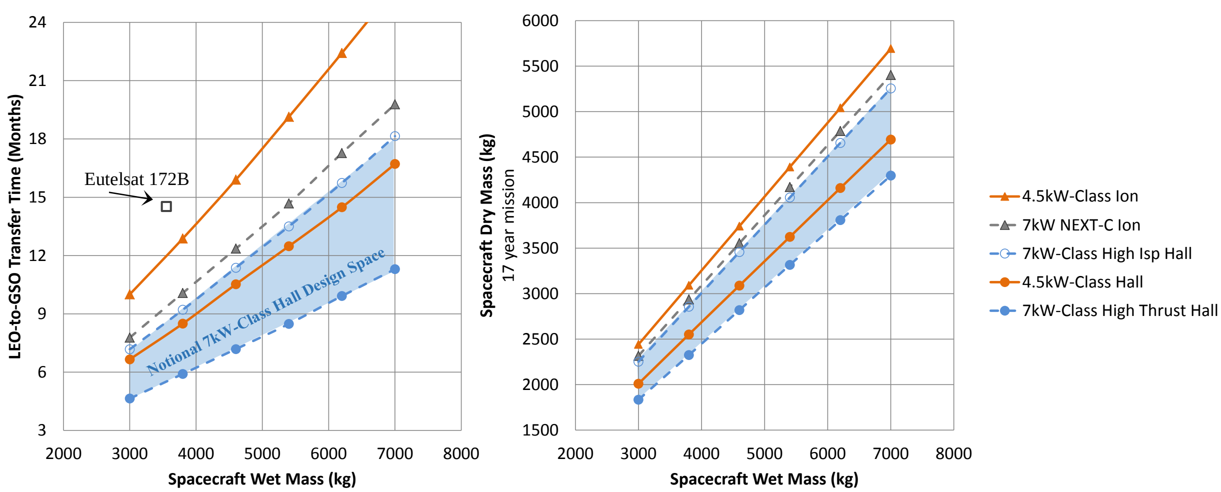

Figure 2. Transfer Time and Dry Mass of 25 kW Satellite Bus with notional 7 kW Hall and NEXT-C thrusters versus state-of-the-art commercial options (900 x 900 km at 28.5° insertion orbit).

Figure 2 illustrates the time of flight and dry mass advantages of using a higher powered EP system to perform a LEO to GSO transfer at a fixed 25 kW power budget. As stated previously, by using a LEO insertion orbit the satellite operator is launch vehicle agnostic and can choose a provider based upon cost and payload capability. Xenon consumption, including residuals and flight performance reserve, can be computed by subtracting the dry mass from the initial wet mass presented on the x-axis. The 7 kW-class Hall thruster in the high thrust configuration can shorten transit times over current state-of-the-art 5 kW class thrusters – such as AR's XR-5, Fakel's SPT-140, and Safran Aircraft Engines' PPS-5000 – by two to five months over the initial satellite wet mass range of 3,000 to 7,000 kilograms, respectively. In comparing with gridded ion capabilities, the high thrust-to-power of Hall thrusters results in an advantage in trip times although at the expense of dry mass to orbit capability. The 7 kW-Class NEXT-C ion thruster approaches the Hall thruster performance range with the added benefit of high design maturity, illustrating the effect of a relative increase in thrust-to-power vs the existing 4.5 kW-class gridded ion thrusters. An additional benefit of using a 7 kW thruster over a 4.5 kW thruster is reducing the system complexity (e.g. one less thruster and associated plumbing, electrical connections, and valves) of the satellite.

The open square marker on the transfer time plot represents the Airbus-built fully-electric Eurostar-3000 Electric Orbit Raising (EOR) bus at 13 kW utilized for the Eutelsat 172B mission, but launched into the reference orbit considered here, for the purpose of showing the challenge presented by a 900 km LEO orbit raising transfer. In actuality, for the orbit in which Eutelsat 172B was launched, electric orbit raising to the final GSO station was performed in four months[25].

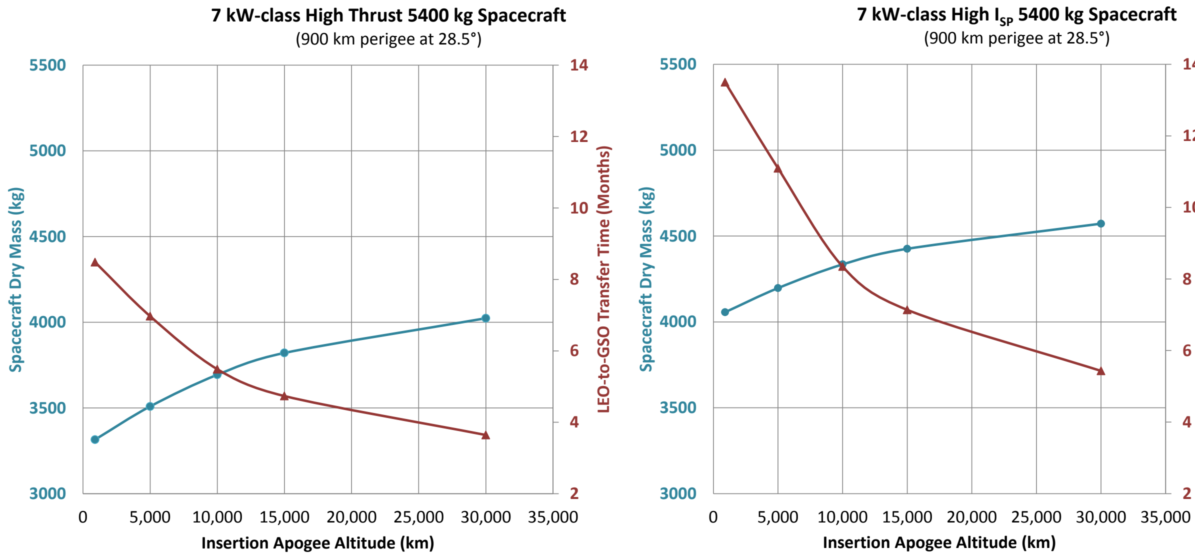

Figure 3. 5400 kg satellite insertion apogee sensitivity to 7 kW Hall thruster configuration (900 km perigee at 28.5°).

Based on the reduced trip time capability introduced by a 7-kW class Hall thruster, the following trades are focused on the potential design space between high thrust and high Isp configurations. Figure 3 explores the mission impact of designing a 7 kW-class Hall thruster for either high thrust or high Isp with respect to launch vehicle insertion orbit. The selection of the thruster is customer dependent as one is trading transfer time (solid red line with triangles) for dry mass (solid teal line with circles). It is seen that trip times for the example 5400 kg (wet mass) satellite are reduced and dry masses increased as the launch vehicle inserts the satellite into a higher orbit, as expected. The relative difference between high thrust and high Isp designs also decrease for higher orbits as the total impulse delivered by electric orbit raising in a high thrust mode, or a high Isp mode, decreases.

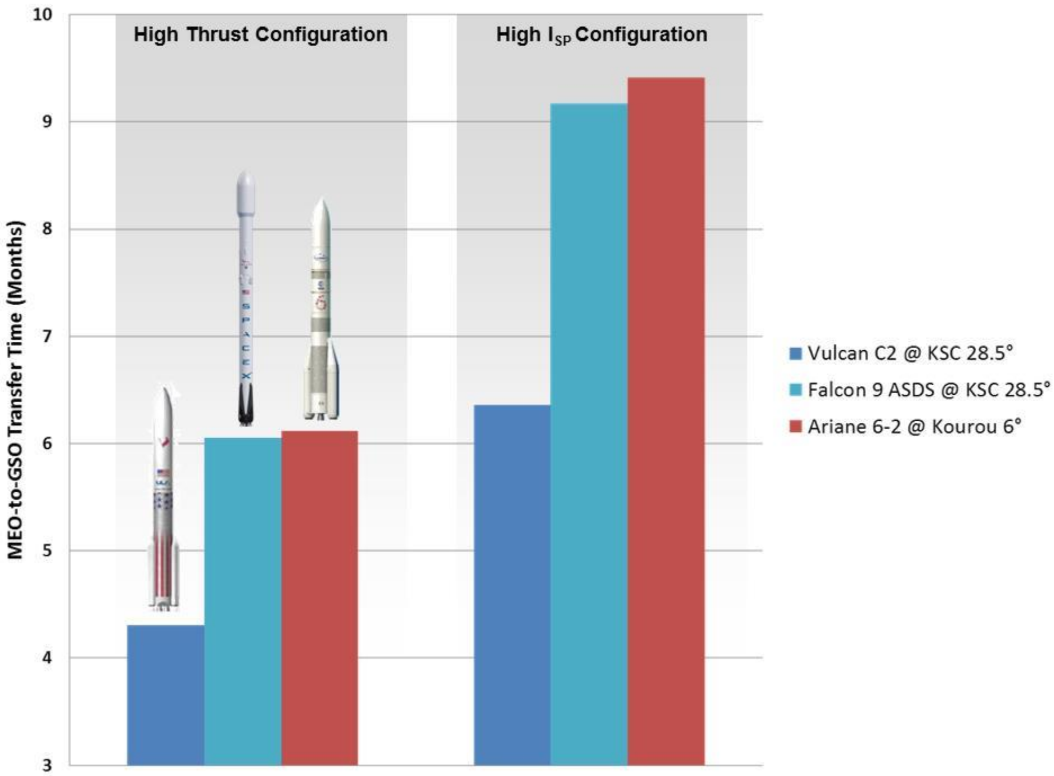

Figure 4. 7000 kg satellite launch vehicle sensitivity to 7 kW-class Hall thruster configuration (utilizing LV's maximum payload capacity).

Figure 4 investigates the sensitivity of using several commercial launch vehicles for satellite deployment. This analysis is performed by flying a fixed 7,000 kilogram wet mass 25 kW satellite to the highest possible orbit achievable by the Falcon 9 ASDS, the Vulcan Centaur with two solid rocket motors, or the Ariane 6 with two solid rocket motors. All rockets were able to launch into a Medium Earth Orbit (MEO) which further reduces the transfer time as compared with LEO insertions. As seen on the red bar, the lower vehicle performance of the Ariane 6 is offset by its launch site inclination as compared with the Falcon 9 ASDS. For the Ariane 6 the insertion was at 6° to the equator corresponding with the latitude of the Guiana Space Center in Kourou French Guiana. The Falcon 9 ASDS was assumed to launch from Kennedy Space Center in Cape Canaveral Florida with latitude of 28.5°. Thus the inclination change, to GSO at 0°, required by the spacecraft was reduced by 22.5°. Furthermore the results of using the Vulcan Centaur with 2 solids illustrate that a zero solid configuration may provide additional cost reduction for the operator if transit time can be relaxed. Overall, the Vulcan C2 option reduces the transit time 1.5 – 2.5 months compared with Falcon or Ariane options. The authors were not able to analyze the zero-solid configuration as the launch vehicle performance information was not available in the public domain.

F. Evolutionary (31 kW) Satellite Bus Capability

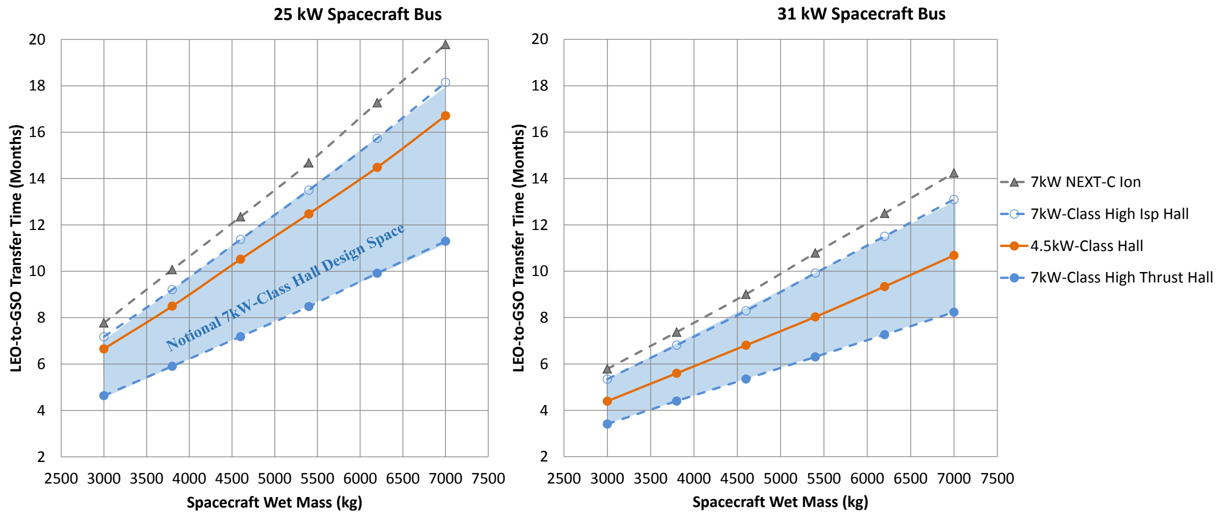

Figure 5. Transfer time of state-of-the-art 25 kW satellite bus (left) vs notional 31 kW Satellite Bus (right) with notional 7kW Hall thruster and NEXT-C versus state-of-the-art commercial options (900 x 900 km at 28.5° insertion orbit).

Figure 5 examines the impact of future GEO satellite bus capability evolving to 31 kW. In combination with high-end PPU efficiencies of 95%, the extra bus power allows the use of an additional 7 kW class thruster (from three thrusters to four) and further reduces the transit time compared to a 25 kW bus. The high thrust configuration allows larger satellites between 5500 kg to 6800 kg to be deployed at transfer times of six-to-eight months from LEO, which still meets the mass conditions for dual manifested launches on a Falcon 9 ASDS. For 3000 kg to 5000 kg satellites, the reduction in trip time vs the current 25 kW bus decreases the time cost of not providing communication services and generating revenue while meeting existing standards for orbit raising trip times at a much lower initial orbit. The electrostatic technological impact of using a 7 kW hall effect thruster versus a 7 kW ion engine can be observed by contrasting the NEXT-C ion engine line. A comparison to using 5 kW class engines is obtained from the power budget which allows for six thrusters operating on a 31 kW bus.

G. Benefits of Improvement in Current State-of-the-Art

A significant drive in the commercial satellite industry for increased performance is increased revenue, which is primarily a factor of increased data throughput capability, decreased transfer time (in order to have more time on orbit generating revenue or to meet standard lead times when considering vehicle integration and launch durations), or decreased satellite wet mass to allow dual manifested launches or launch vehicle step downs.

To increase data throughput, satellite companies look to increase transponder throughput capability along with increasing the payload mass capability to orbit (thus the number of transponders). Higher frequency Ka-band transponders are becoming more commonly used allowing higher data throughput per mass and bandwidth and smaller antennas, along with emerging strategies such as spot beaming. In this regard, the capability offered by high Isp EP systems is most appealing. Gridded ion engines currently in use for all-electric vehicles capitalize on the existing demand for this end of the mission design space, and it is inferred from the results above that as vehicle powers increase, adding an additional 4.5kW gridded ion thruster will continue to be an effective solution. Higher power ion thrusters may gain acceptance as the need to drive down the number of thrusters (i.e. cost and complexity) per vehicle reaches a tipping point, or as the thrust to power is increased for users in need of reduced trip times during transfers.

More dramatically, the opportunity to reduce electric orbit raising trip times through the development of high thrust Hall thruster systems suggests improvements over current state of the art are possible for existing 25 kW buses, and even more once vehicle powers reach 31 kW. As with gridded ion thrusters, the addition of a 4.5kW Hall thruster will be possible to capitalize on increases in available bus powers. However, it is also possible that in the time required to develop and flight qualify a modified high-thrust 4.5kW Hall thruster bus powers will have increased. Anticipating this increase in available power coupled with a desire to minimize integration cost and complexity, a 7 kW high thrust Hall thruster (7.5 kW system) would most effectively benefit from further development. Table 5 presents the improvement made possible by launching a 3800 kg satellite into LEO, where it is seen that an eight month electric orbit raising duration can be reduced to four months with the proper advancement in propulsion system and bus power capability.

Table 5. Summary of mission inputs/outputs for 17 year, 3800 kg wet mass satellite launched to 900 km x 900 km circular orbit.

Bus Power	EP System	No. Thrusters	PPU Efficiency	EOR transfer (months)	Satellite dry mass (kg)
25 kW	4.5 kW-class gridded ion	5	0.92	13	3092
	4.5 kW-class Hall	4	0.92	8	2551
	7 kW-class gridded ion (NEXT-C)	3	0.92	10	2936
	7 kW-class Hall (High Thrust)	3	0.92	6	2325
	7 kW-class Hall (High Isp)	3	0.92	9	2856
31 kW	4.5 kW-class Hall	6	0.92	6	2543
	7 kW-class gridded ion (NEXT-C)	4	0.92	7	2934
	7 kW-class Hall (High Thrust)	4	0.95	4	2322
	7 kW-class Hall (High Isp)	4	0.95	7	2851

IV. Conclusions

The past decade of expanded use and capability of EP systems for GEO communications satellite missions demonstrates the increasing preference for EP over heritage chemical systems. All-electric satellites are being used for electric orbit raising, station keeping, maneuvering, and de-orbit functions. The current state of the art in satellite bus and EP string power is near 25 kW and 5 kW, respectively. Multiple providers of Hall thruster and gridded-ion thruster technologies are emerging around these common design parameters.

Further increases in available bus power, EP string power, and EP string performance have been shown to result in meaningful benefits to satellite users. By increasing from 25kW to 31kW in bus power, additional thrust can be provided during electric orbit raising which reduces transfer times by one to three months depending on vehicle mass. By increasing the EP string power capability full utilization of the additional bus power results without the added burden or expense of installing additional thrusters. Lastly, by targeting a specific purpose such has high thrust to power for a 7 kW thruster system, electric orbit raising transfer times were shown to decrease by up to five months for large GEO satellites launched into orbits with appreciable electric orbit raising, bringing them to the six month transfer regime.

The resulting improvements in orbit operations motivate a continued trending for EP systems for flight GEO communications satellites in the coming years to take advantage of potential higher available bus powers in ways that can be designed to reduce transfer times as well as the more commonly known reasons of maximized payload to orbit, increased mission life, or savings in launch vehicle costs.

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