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Advanced Propulsion for Geostationary Orbit Insertion and North–South Station Keeping

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Solar electric propulsion technology is currently being used for geostationary satellite station keeping. Analyses show that electric propulsion technologies can be used to obtain additional increases in payload mass by using them to perform part of the orbit transfer. Three electric propulsion technologies are examined at two power levels for geostationary insertion of an Atlas IIAS class spacecraft. The onboard chemical propulsion apogee engine fuel is reduced in this analysis to allow the use of electric propulsion. A numerical optimizer is used to determine the chemical burns that will minimize the electric propulsion transfer times. For a 1550-kg Atlas IIAS class payload, increases in net mass (geostationary satellite mass less wet propulsion system mass) of 150–800 kg are enabled by using electric propulsion for station keeping, advanced chemical engines for part of the transfer, and electric propulsion for the remainder of the transfer. Trip times are between one and four months.

Introduction

SOLAR electric propulsion (SEP) is already being used for station keeping of geostationary satellites, most notably hydrazine arcjets on AT&T's Telstar 4 and SPT-100 Hall thrusters on the Russian GALS spacecraft.[1] A next step in the application of electric propulsion is its use in placing the spacecraft into geostationary orbit. For a given launch vehicle, the fuel-mass savings then could be directly used to increase the payload. For instance, this could allow more communication transponders. Even a small increase in mass might have large revenue impacts.

The current trend for geostationary spacecraft is toward longer lifetimes, increased masses, higher powers, and increased service bandwidth. The Intelsat series[1] of satellites present a good example of these trends. Intelsats 1 and 2, launched during the late sixties, had lifetimes under four years. Intelsats 4 and 5 had seven-year design lifetimes. Intelsat 7 had a full-capacity design lifetime of 10 years with propellant for 15 years. The planned Intelsat 8/8A series lifetime is 14–18 years using N_2H_4 arcjets for station keeping. Satellite masses, and the launch vehicles to deliver them, also have grown. Early Intelsats were well under 1000-kg dry mass. The planned Intelsat 8/8A series will have a 1530-kg dry mass. End-of-life power levels have increased from hundreds of watts for Intelsats 1–4, to over 5 kW for Intelsat 7A. Intelsat 8/8A will use the Lockheed Marietta Astro Space Series 7000 satellite which has a beginning-of-life power level over 7 kW. Finally, communication bandwidths on Intelsat spacecraft have increased from 50 MHz on Intelsat 1 to 2856 MHz on the planned Intelsat 8/8A series. These continuing trends toward larger, more capable, longer life and higher-power spacecraft were used to select the spacecraft characteristics in this study. Higher-power spacecraft permit expansion of the use of electric propulsion systems beyond the already-demonstrated station-keeping function to encompass a portion of the orbit transfer mission. Successful implementation of advanced propulsion systems will enable continued growth of geostationary satellite capability without requiring growth in spacecraft mass or launch vehicle and will permit continued expansion of communications capability.

Studies by various authors have shown the net mass benefits of using electric propulsion for transfer from various high Earth orbits[2–6] to avoid the long trip times and Van Allen belt radiation damage of low Earth orbit to geostationary Earth orbit (GEO) transfers[6,7] using electric propulsion. However, none of these starting orbits were found optimally as is done in this work. This paper describes the mission analyses, propulsion options, and the results for the three electric propulsion options.

The purpose of this paper is to show the performance advantages of advanced onboard propulsion technologies for near-term geostationary missions. This study evaluated the mass impact of replacing some portion of a geostationary spacecraft's chemical apogee propulsion system with either an N_2H_4 arcjet system, a Hall thruster system, or a xenon ion system, which also performs 15 years of station keeping. The analyses used conservative assumptions for these propulsion systems to make the results applicable to near-term missions. While an Atlas IIAS class spacecraft was assumed for this analysis, the application of this method was shown to be applicable to other launch vehicles by Oleson.[8] Two payload power levels, 10 and 15 kW, were assumed to be available for the electric propulsion orbit transfer. The numerical optimization program Solar Electric Propulsion Steering Program for Optimal Trajectory (SEPSPOT)[9] was used to identify the chemical burns of the Centaur upper stage and onboard propulsion system to minimize the electric propulsion transfer time.

Mission Analysis: Options and Assumptions

The approach is to utilize the numerical optimizer SEPSPOT with its option to perform optimal impulsive-stage analysis to minimize the SEP transfer time. All that is required for the high-thrust portion of the program is a final mass for this phase of the mission and an initial impulsive ΔV. The final mass of the impulsive phase is the starting mass for the SEP mission. The ΔV is the velocity or energy change required for an orbit transfer. Impulsive ΔV assumes an instantaneous burn and is assumed for all chemical propulsion burns in these analyses. The SEP transfer mission ΔV differs from the impulsive ΔV due to gravity losses associated with continuous thrusting and nontangential steering.[10]

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The launch vehicle assumed for this analysis is the Atlas IIAS with the large payload fairing.[11] After liftoff, the Centaur upper stage uses a portion of its fuel to place the payload satellite, including the necessary onboard propulsion systems to achieve geostationary orbit, into an assumed low 185-km-altitude circular parking orbit. Although Atlas launch vehicles sometimes use elliptical parking orbits to optimize the perigee burn, the high-thrust option of the SEPSPOT program is currently limited to circular starting orbits.[11]

After reaching parking orbit, the Centaur stage still carries approximately 4400 kg of fuel, which normally is used to place the payload spacecraft into geostationary transfer orbit (GTO). The GTO assumed in this analysis has a perigee altitude of 185 km and an apogee altitude of 35,785.5 km. The Centaur specific impulse (I_{sp}) is assumed to be 451.5 s. The starting mass in the parking orbit is 10,240 kg, which includes the spacecraft, onboard propulsion systems, and the partially fueled Centaur stage.[12]

The mission cases in which the electric propulsion system performs only station keeping use the Centaur stage to place them into GTO and the onboard chemical system to insert them into geostationary orbit. The mission cases where a portion of the geostationary orbit insertion is performed by the onboard electric propulsion system use the remaining Centaur stage fuel and the available onboard chemical fuel in an optimal one- or two-burn transfer to an optimal SEP starting orbit, as shown in Fig. 1. The Centaur portion of this transfer is not necessarily to GTO.

SEPSPOT determines the required one or two impulsive burns with the allotted ΔV to reach an SEP starting orbit that minimizes the SEP trip time. This SEP starting orbit can have any perigee, apogee, and inclination combination that is achievable with the given impulsive ΔV. This ΔV is the sum of the remaining ΔV capability of the Centaur stage and some portion of the onboard chemical apogee ΔV normally carried. This onboard portion is varied from 1800 to 0 m/s to show the trade between increased net mass and increased trip time. To illustrate these trades, Fig. 2 shows a variation between the onboard chemical ΔV and the transfer SEP ΔV for a case using 30-cm ion thrusters. Note that the Centaur ΔV is held constant, whereas the onboard chemical ΔV is reduced in increments of 100 m/s. The required SEP ΔV from SEPSPOT to replace the onboard chemical ΔV is greater because of gravity losses. This required SEP ΔV is further discussed in the "Results" section. The baseline mission, where the geostationary insertion is performed solely by the onboard chemical propulsion system (case 1), requires a chemical system ΔV of 1805 m/s. The mass of the satellite after all allotted chemical fuel is used and the dry 2180-kg Centaur[12] is separated is the starting SEP phase mass.

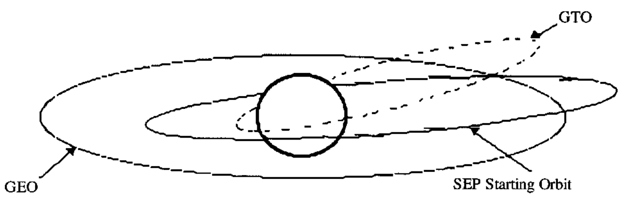

Fig. 1 Mission orbits.

The SEP phase optimization includes the impacts of shading, J2 (Earth oblateness), and the solar array degradation due to Van Allen belt radiation. The SEP system parameters of initial power level, I_{sp}, and efficiency are fixed in SEPSPOT. SEPSPOT assumes continuous thrusting except while the spacecraft is in shade. SEPSPOT finds the optimal steering for the minimum time trajectory.

The impact of power degradation on the trip time causes SEPSPOT to minimize time spent in the Van Allen belts. As power is degraded, SEPSPOT throttles the thrusters accordingly while maintaining the same I_{sp} and efficiency. Although thruster performance normally varies as a function of power level, this effect is neglected in this work. This SEPSPOT/SEP system modeling limitation is negligible for the desired short transfer time trajectories because the power degradation is negligible. The impacts of nonoptimal steering and guidance, navigation, and attitude-control limitations are not considered here. The impacts of these issues typically are minor.

In addition to the transfer, 15 years of north–south station keeping (NSSK) are assumed for all cases.[1] Although the yearly ΔV varies with satellite station longitude, 45.37 m/s is chosen as representative.[13] The daily station keeping burn time using electric propulsion is on the order of tens of minutes. The cosine losses encountered by not completing the whole burn instantaneously at the orbit node are small and neglected. East–west station keeping requirements are an order of magnitude smaller than NSSK requirements and are neglected in these analyses.

System Assumptions and Modeling

Onboard Chemical Propulsion System

For mission scenarios requiring an onboard chemical propulsion system for all or part of the orbit insertion, an advanced 328-s I_{sp} bipropellant system is assumed.[14] A 314.5-s I_{sp} system is assumed only for the state-of-art (SOA) case. Both systems have a fixed dry mass of 23 kg and a tankage fraction of 0.08. The advanced chemical system is deleted from the spacecraft for those missions where the SEP system takes over directly from the Centaur stage.

Onboard Electric Propulsion System

For mission scenarios using onboard SEP for NSSK and, in some cases, orbit insertion functions, the following technologies are considered: SOA 1.8-kW N_2H_4 arcjets[15] for NSSK function only, two advanced 2.17-kW N_2H_4 arcjets,[16] 1.5-kW xenon Hall thrusters,[17] and 2.5-kW 30-cm xenon ion thrusters.[16] The power given is the power into the power processing unit (PPU). All thruster parameters are shown in Table 1. Throughout this analysis, the same propulsion technology is used for both transfer and NSSK functions—no mixing of propulsion technologies is considered.

For the orbit insertion function, the assumed thruster-specific impulses are 600 s for the advanced arcjet, 650 s for the advanced

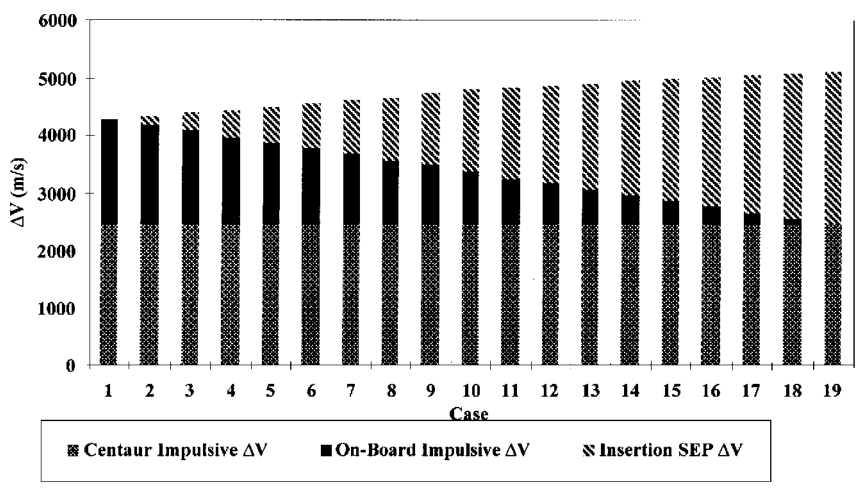

Fig. 2 Mission ΔV breakdown vs case: ion 10-kW class GEO satellite.

Table 1 SEP propulsion system parameters

	Propulsion system parameter			
	SOA N_2H_4 arcjet	Advanced, advanced+ N_2H_4 arcjet	Xenon Hall thruster	Xenon ion thruster
Desired PPU input power level, kW	1.8	2.17	1.5	2.5
I_{sp}, s	500	600, 650	1600	3160
Overall efficiency (PPU and thruster)	0.33	0.33, 0.31	0.45	0.60
Tankage, %	7	7	15	15
Life at power level, *h*	1000	1500	4000	8000
Cant angle for NSSK, deg	17	17	45	30
Equivalent cant I_{sp}, s	478	574, 622	1131	2736
Mass				
Thruster, kg	1	1	5	7
Gimbals, % of thruster	None	34	34	34
Support, % of gimbals of thrusters	31	31	31	31
Controller, kg/thruster	0.55	0.55	0.55	1.55
Total thruster + gimbal + support + controller, kg/thruster	1.9	2.3	9.3	13.8
Feed system, kg/kWe	0.8	0.8	1.5	1.5
PPU, kg/kWe	2.4	2.4	4.6	4.7
Cabling, kg/kWe	0.4	0.4	0.4	0.4
Thermal system (92% PPU), kg/kWt-disp.	31	31	31	31
Total PPU + feed + cabling + thermal, kg/kWe	6.1	6.1	9.0	9.1

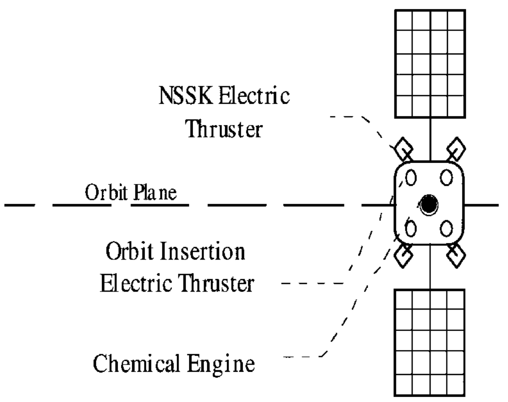

Fig. 3 Potential thruster placement.

+ arcjet, 1600 s for the Hall thruster, and 3160 s for the ion thruster. The overall PPU/thruster efficiencies, regardless of mission function, are 0.33 for the SOA arcjet, 0.33 for the advanced arcjet, 0.31 for the advanced + arcjet, 0.45 for the xenon Hall thruster, and 0.60 for the xenon ion thruster.

Each thruster unit includes structure, gimbal (except SOA arcjet), and controller. The resulting masses are 1.9 kg for the SOA arcjet, 2.3 kg for the advanced and advanced + arcjets, 9.3 kg for the Hall thrusters, and 13.8 kg for the ion thrusters. Each PPU unit includes cabling and thermal system, resulting in power densities of 6.1 kg/kW for the SOA arcjet, 6.1 kg/kW for the advanced and advanced + arcjets, 9.0 kg/kW for the Hall thrusters, and 9.1 kg/kW for the ion thrusters. A tankage fraction of 0.07 was used for arcjets and 0.15 for the Hall and ion thrusters. Thruster lifetime also is considered and extra thrusters are added when necessary. Assumed thruster lifetimes are 1000 h for the arcjet SOA, 1500 h for the advanced and advanced + arcjets, 4000 h for the Hall thrusters, and 8000 h for the ion thrusters. PPU lifetime was assumed adequate for both the transfer and station keeping missions.

Fifteen years of north–south spacecraft station keeping is performed by four thrusters, one pair placed on the north face and the other on the south face as shown in Fig. 3. These thruster pairs are canted 17, 45, and 30 deg for the arcjets,[16] Hall thrusters,[5] and ion thrusters,[16] respectively, from the vertical to minimize plume interaction with the array. The equivalent NSSK thruster I_{sp} is adjusted for the thruster cant cosine loss as follows: 478 s for the SOA NSSK arcjet, 574 s for the advanced arcjet, 622 s for the advanced + arcjet, 1131 s for the Hall thruster, and 2736 s for the ion thruster. To perform the north-south station keeping, either the south or north pair is fired about the appropriate orbit node on the order of tens of minutes. If one thruster fails, the opposite set is tasked with all NSSK burns. Four PPUs support the four NSSK thrusters.

Additional thrusters are added for performing the SEP transfer mission; 10- and 15-kW available power levels are assumed (see "Power System"). Thrusters are added for the transfer mission to take advantage of the available power. Consequently, the 10-kW spacecraft uses either 4 arcjets, 6 Hall thrusters, or 4 ion thrusters. The 15-kW spacecraft uses either 6 arcjets, 10 Hall thrusters, or 6 ion thrusters. Note that the arcjets do not use all available power because of their power input requirement. As mentioned previously, the thrusters are assumed identical to the NSSK thrusters except that they are placed about the chemical thruster on the aft portion of the spacecraft, as shown in Fig. 3. The transfer thrusters use the available four NSSK PPUs and have additional PPUs added for extra thrusters, for example, the six Hall thrusters have two PPUs added to the spacecraft. During SEP transfer, all of the transfer thrusters are firing except in shade. Additional thrusters for redundancy were not added.

Power System

The GaAs solar arrays that provide payload power in geostationary orbit are assumed to provide the 10 or 15 kW for the thruster operation during the SEP orbit transfer because the payload is inactive during this phase. These power levels were chosen as representative of next-generation power levels for geostationary communication satellites.[1] The battery system is assumed to power NSSK thruster operation whereas the payload uses direct solar array power as suggested by Free.[18] Extra batteries may be required to support the increase in charge/discharge cycling, but this mass is not determined here. The arrays are assumed to have an equivalent layer of 6 mils of fused silica shielding on both sides of the solar array for radiation protection.[13] Because the array is resident on the spacecraft for payload use, its mass is not charged to the propulsion system. However, transfer through the Van Allen belts will damage the array. This damaged array mass is charged to the propulsion system at a rate of 16.6 kg/kW.[19] Thus, the propulsion system is penalized for long transfers through the Van Allen belts. The radiation damage that may occur to the payload is not assessed here, but it should be less than that encountered by the array.

Results

SEP Starting Orbits

Optimal SEP starting orbits determined by SEPSPOT for the 10-kW spacecraft with ion technology are shown in Fig. 4. These SEP starting orbits vary little for the different SEP power levels, so Fig. 4 is representative of all results. The orbit parameters, including apogee altitude, perigee altitude, and inclination, are shown relative to the onboard chemical propulsion ΔV. This directly relates to chemical propulsion fuel loading. Only one or two burns are allowed by the code. The three cases with 200 m/s or less of onboard chemical ΔV use only one perigee burn to lift apogee as high as possible. A slight plane change is also performed. In practice, several perigee burns might be used. Increasing the onboard chemical ΔV capability above 200 m/s, up to 1800 m/s, allows an optimal two burn case. The first burn raises apogee above geostationary orbit altitude, and the second burn raises the perigee some amount along with performing some portion of the required plane change.

By setting the apogee above and the perigee below the target orbit, SEPSPOT increases the time the spacecraft spends out of the most damaging portions of the radiation belts. The higher apogee results in a lower velocity location for plane changing. An example of the optimal steering determined by SEPSPOT is shown in Figs. 5 and 6. Note in the figures that the in-plane component of the thrust vector is pointed in roughly an inertial direction whereas the out-of-plane vector is maximum at the orbit nodes (intersection of the transfer orbit and the GEO orbit) and zero at the antinodes. Results for the rest of the 30-day 10-kW ion mission, as well as the other power levels and technologies, are similar.

Figure 2 shows the corresponding required transfer SEP ΔV for the varied onboard chemical ΔV for the 10-kW ion class. These required transfer SEP ΔV vary little for the different SEP technologies. Cases 1–18 show the trade in chemical and SEP ΔV. As onboard chemical ΔV capability is replaced by SEP ΔV, the total ΔV increases because of the gravity losses incurred by the continuously thrusting SEP system. Case 19 shows the limit when the GTO-to-GEO transfer is performed completely by the SEP system and the Centaur stage. Comparing cases 19 and 1 clearly shows the increased ΔV required. However, the higher I_{sp} of the SEP system more than offsets this increased ΔV by significantly reducing the total fuel mass. This is shown by the net mass advantage in the next sections.

Figures of Merit

The figures of merit of the advanced propulsion systems in this study are the net mass delivered and SEP transfer time. Net mass refers to the usable satellite mass once the wet propulsion system and any damaged array are removed. The added net mass can be used for additional payload to increase revenue.

Chemical System Performance

The SOA system is assumed to be a 314.5-s I_{sp} onboard chemical system that delivers the spacecraft into geostationary orbit, and a 500-s I_{sp}, 1.8-kW arcjet system that only performs the NSSK. These systems are termed the SOA technologies and both are described in the "System Assumptions" section. Figure 7 exhibits the net masses achievable with the SOA or advanced chemical system performing the entire orbit insertion. The SOA chemical and SOA arcjet system delivered net mass is 1551 kg. This is referred to as the baseline-SOA case. The impact of replacing the SOA chemical system with the advanced chemical system while retaining the SOA arcjet increases the net mass to 1598 kg, for a gain of 47 kg. This 1598-kg net mass case is considered the baseline (and termed baseline-advanced chemical case) for all further evaluations of the added performance of advanced electric propulsion technologies.

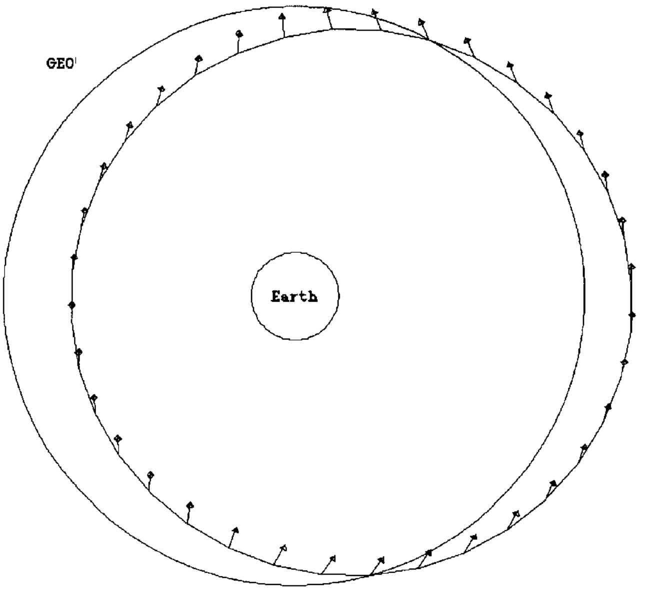

Fig. 5 Top-down view of the first day of steering for the 30-day, 10-kW class ion case.

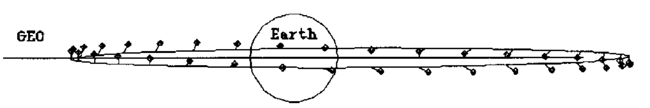

Fig. 6 Side view of the first day of steering for the 30-day, 10-kW class ion case.

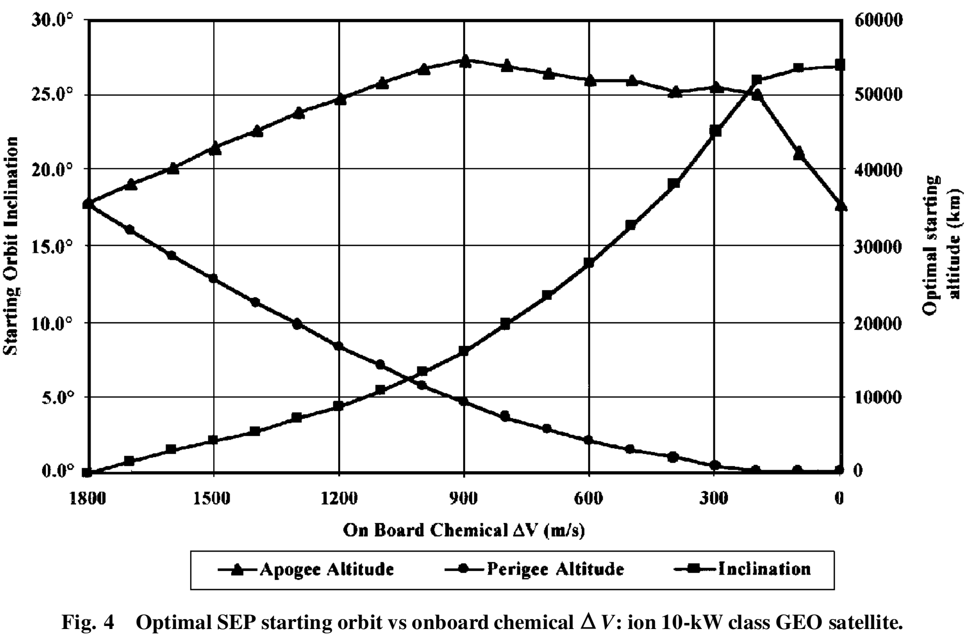

Fig. 4 Optimal SEP starting orbit vs onboard chemical ΔV: ion 10-kW class GEO satellite.

Table 2 Summary of 10-kW class spacecraft cases

Thruster	NSSK only	Orbit insertion case				
N_2H_4-AJ						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	8	27	51	85	123
Power degradation fraction	1.000	0.996	0.987	0.919	0.803	0.736
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.55	2.10	2.62
Final net mass, kg	1,635	1,647	1,685	1,719	1,759	1,857
Δ Net mass over chemical only transfer, kg	0	12	50	84	124	222
SEP start-orbit periapsis altitude, km	35,786	28,937	16,790	7,431	1,968	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,633	52,697	49,830	35,435
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.8	19.1	26.7
N_2H_4-AJ+						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	9	31	59	100	150
Power degradation fraction	1.000	0.995	0.986	0.913	0.794	0.713
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.55	2.10	2.62
Final net mass, kg	1,653	1,647	1,730	1,780	1,837	1,934
Δ Net mass over chemical only transfer, kg	0	21	76	127	184	281
SEP start-orbit periapsis altitude, km	35,786	28,937	16,790	7,438	1,871	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,633	52,538	51,419	35,305
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.9	19.4	26.6
Xe-Hall						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	16	53	109	194	310
Power degradation fraction	1.000	0.993	0.979	0.888	0.733	0.621
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.56	2.12	2.64
Final net mass, kg	1,718	1,735	1,921	2,111	2,320	2,531
Δ Net mass over chemical only transfer, kg	0	17	203	393	602	813
SEP start-orbit periapsis altitude, km	35,786	28,937	16,791	7,415	1,875	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,633	54,477	51,270	35,502
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.8	19.3	26.8
Ion						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	21	71	151	286	461
Power degradation fraction	1.000	0.991	0.974	0.869	0.684	0.571
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.56	2.12	2.63
Final net mass, kg	1,746	1,809	2,059	2,321	2,607	2,969
Δ Net mass over chemical only transfer, kg	0	63	313	575	861	1223
SEP start-orbit periapsis altitude, km	35,786	28,937	16,791	7,427	1,897	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,636	54,025	50,450	35,671
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.8	19.1	27

[a]SEP transfer times include shade, J2, optimal steering, and degraded power.

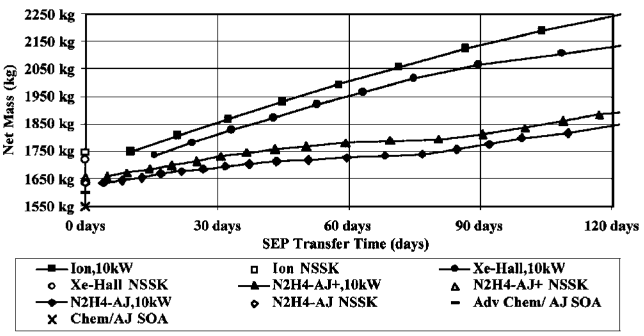

Fig. 7 Final net mass vs SEP transfer time: 10-kW class GEO satellite.

Ten-Kilowatt Class Spacecraft

Figure 7 contains the results of this analysis for a 10-kW class spacecraft in terms of net mass vs SEP transfer time. A partial summary of 10-kW class data is given in Table 2. Figure 7 also shows the NSSK missions where the orbit transfer is completed by the Centaur stage and onboard chemical system and the station keeping is performed by the electric propulsion system. These NSSK-only SEP missions show the net mass benefit of implementing the advanced electric propulsion technologies just for NSSK. As expected, the higher I_{sp} systems provide a greater net mass. These higher I_{sp} NSSK systems also may require more power and/or longer burn times, but these impacts are not considered here. The advanced chemical transfer and advanced SEP NSSK systems provide an 80- to 200-kg increase in net mass over the baseline-SOA system.

Figure 7 also shows that by expanding the electric propulsion system to provide part of the transfer, even greater net mass gains may be realized. Although ion and Hall transfers are longer for the lower onboard chemical ΔV cases, only SEP transfers up to four months are shown. For the same onboard chemical ΔV, each SEP system requires a different transfer time because of the differences in I_{sp} and efficiency and thus thrust level. The initial steepness of each technology's curve is reduced somewhat for longer transfer times because of the increased rate of solar array damage (see Fig. 8), which is subtracted from the net mass.

This increased damage rate is due to longer exposure times in the more damaging portions of the Van Allen belts. For the shortest transfer times, where the onboard chemical system is providing most of the transfer, the radiation damage is small, and the net mass gain increases quickly as allowable SEP transfer time is relaxed. This region of slight degradation occurs for onboard chemical ΔV above approximately 1000 m/s. The net mass gain for the arcjet technologies smoothes out after about 40 days because of the appearance of substantial radiation damage. Hall and ion technologies smooth out at longer transfer times because of their lower thrust but at the same point of notable radiation damage.

The net mass gains to be made with any of the advanced SEP technologies are considerable. The 600- and 650-s I_{sp} arcjets provide an additional 60 and 80 kg, respectively, of net mass over NSSK alone for a one-month transfer time. The Hall and ion systems provide even greater net mass gains just performing the NSSK mission. After about 10–15 days of transfer time, both systems add even more net mass. Below this transfer time, the additional equipment dry mass overwhelms the higher I_{sp} advantage. These data are not shown in Figs. 7 and 9 for the sake of clarity. For a one-month transfer time, which is roughly equivalent to a geostationary satellite's checkout time, the use of Hall thrusters or ion thrusters for part of the orbit transfer increases the satellite net mass by 110 and 120 kg, respectively over NSSK only. Compared to the baseline-advanced

Table 3 Summary of 15-kW class spacecraft cases

Thruster	NSSK only	Orbit insertion case				
N_2H_4-AJ						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	6	18	33	54	77
Power degradation fraction	1.000	0.997	0.990	0.936	0.838	0.779
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.55	2.10	2.60
Final net mass, kg	1,635	1,630	1,668	1,700	1,746	1,843
Δ Net mass over chemical only transfer, kg	0	−4	33	65	111	208
SEP start-orbit periapsis altitude, km	35,786	28,937	16,790	7,427	1,980	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,633	52,829	49,534	35,391
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.8	19.0	26.6
N_2H_4-AJ+						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	6	20	39	64	92
Power degradation fraction	1.000	0.997	0.989	0.931	0.825	0.763
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.55	2.10	2.61
Final net mass, kg	1,653	1,657	1,713	1,761	1,823	1,922
Δ Net mass over chemical only transfer, kg	0	4	59	107	170	269
SEP start-orbit periapsis altitude, km	35,786	28,937	16,790	7,437	2,001	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,633	52,584	49,203	35,416
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.9	18.9	26.6
Xe-Hall						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	9	31	62	109	167
Power degradation fraction	1.000	0.995	0.985	0.913	0.787	0.703
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.55	2.11	2.63
Final net mass, kg	1,718	1,661	1,847	2,036	2,239	2,504
Δ Net mass over chemical only transfer, kg	0	−57	129	318	521	786
SEP start-orbit periapsis altitude, km	35,786	28,939	16,790	7,433	1,953	185
SEP start-orbit apoapsis altitude, km	35,786	40,593	49,633	52,650	50,409	35,479
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.8	19.2	26.7
Ion						
Onboard impulsive ΔV, m/s	1,805	1,600	1,200	800	400	0
SEP transfer time, days[a]	0	14	47	97	176	279
Power degradation fraction	1.000	0.993	0.981	0.895	0.746	0.637
Transfer low-thrust ΔV, km/s	0.00	0.31	0.94	1.56	2.11	2.6
Final net mass, kg	1,746	1,763	2,012	2,271	2,552	2,903
Δ Net mass over chemical only transfer, kg	0	17	266	525	805	1,157
SEP start-orbit periapsis altitude, km	35,786	28,937	16,791	7,431	1,785	185
SEP start-orbit apoapsis altitude, km	35,786	40,594	49,633	53,857	52,216	35,432
SEP start-orbit inclination, deg	0.0	1.4	4.4	9.9	19.4	26.7

[a]SEP transfer times include shade, J2, optimal steering, and degraded power.

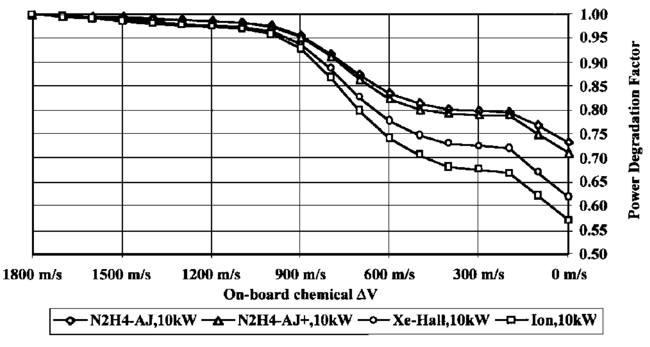

Fig. 8 Power degradation vs onboard chemical ΔV: 10-kW class GEO satellite.

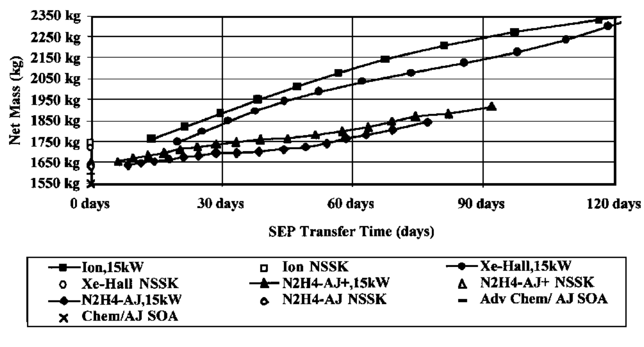

Fig. 9 Final net mass vs SEP transfer time: 15-kW class GEO satellite.

chemical case, the net mass increase is 230 kg with the Hall thruster and 270 kg with the ion thruster. These provide net mass gains of 14% for the Hall to 16% for the ion. A more appropriately powered (~2.5 kW) Hall thruster may have an increased net mass benefit.

Allowing two months of trip time adds more net mass for all of the thruster systems, with Hall and ion outperforming arcjets. The rate of net mass increase for the arcjets with transfer times greater than two months is minor, mainly because of the increasing damage to the solar array. Hall and ion thrusters add over 350 and 400 kg, respectively, when compared to the baseline-advanced chemical case. At three- and four-month transfer times, the ion thrusters add over 550 and 650 kg for a substantial 34 and 40% increase over the baseline-advanced chemical case. The cost of these transfer times is not considered here.

Not only can net mass be significantly increased, but spacecraft growth during design and production can be handled easily merely by removing some of the onboard chemical propellant and adding some SEP propellant. Thus by designing the SEP fuel tanks for extra fuel, substantial net mass flexibility can be attained at the cost of larger tanks. For the arcjet cases, the chemical and arcjet systems use a common N_2H_4 fuel tank so that larger SEP tanks would not be necessary.

Fifteen-Kilowatt Class Spacecraft

Results for the 15-kW spacecraft, shown in Fig. 9, are similar to those of the 10-kW spacecraft. A partial summary of 15-kW class data is given in Table 3. Faster transfer times due to higher SEP powers are offset by additional thrusters and PPUs. The NSSK-only scenario points are identical to those of the 10-kW spacecraft because the additional power is not used for the NSSK system.

An additional 60–90 kg is provided compared to the baseline-advanced chemical system by adding six transfer 600- and 650-s I_{sp}

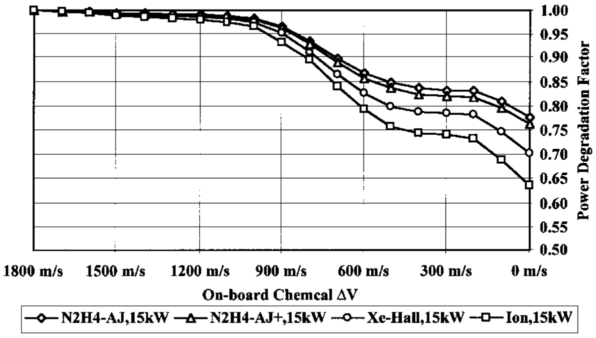

Fig. 10 Power degradation factor vs onboard chemical ΔV: 15-kW class GEO satellite.

arcjet thrusters and two PPUs, respectively, for a two-week transfer time. Between 100 and 140 kg can be added if the transfer time is set to one month. Using a Hall system yields an additional 250–440 kg, and the ion system yields 290–500 kg over the baseline-advanced chemical system for one- and two-month trip times, respectively. Thus the ~30% net mass gain is obtained in two months for the 15-kW class spacecraft as compared to three with the 10-kW spacecraft. Although greater net mass gains can be obtained for longer transfer times, increased radiation is encountered that also would have an adverse effect on the payload.

The radiation dose encountered by the 15-kW spacecraft is less than that of the 10-kW spacecraft (Fig. 10). Although the shapes of damage factor are similar for the 10- and 15-kW spacecraft, the extent of damage is greater for the relatively slower 10-kW spacecraft. As with the 10-kW spacecraft, the transition to significant radiation damage occurs for onboard ΔV below 1000 m/s for the 15-kW spacecraft.

Conclusions

The use of advanced onboard propulsion systems to perform both the NSSK and part of the orbit transfer was examined for GEO spacecraft. Substantial increases in net mass were obtained for moderate trip times, showing the possibility of significant payload enhancements. Upgrading to advanced onboard chemical transfer systems alone can increase the net mass by over 3%. If, in addition to upgrading to the advanced onboard chemical transfer system, an advanced solar electric NSSK system is utilized, an increase in net mass of as much as 13% is realized. Use of advanced SEP for a portion of the orbit transfer increases the net mass by as much as 20 to 45% for one- to four-month transfer times, respectively. The trip time depends on power levels, which were set to 10 or 15 kW in this study, based on current trends in geostationary satellite technology. The use of SEP for portions of the transfer also allows spacecraft design and production mass growth by extending the transfer time.

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