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Geosynchronous-Earth-Orbit Communication Satellite Deliveries with Integrated Electric Propulsion

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The use of electric propulsion was evaluated for transfer of communication satellites from geosynchronous transfer orbits to geosynchronous earth orbits. Recent communication satellite designs, normal launch vehicle delivery orbits, and integrated electric propulsion subsystems (with input powers less than nominal satellite power levels) were assumed to minimize required changes to present and near-term launchers and spacecraft. The capture fraction of recent communication satellites that could have been delivered was evaluated versus launcher delivery capability, launch site, and in-space propulsion characteristics. Electric propulsion significantly increases the capture fraction of launchers with geosynchronous transfer mass delivery capabilities less than (dependent on launch site) about 4500 to 5500 kg. Insertion times at given launch sites were found to be accurately specified by the satellite power-to-mass ratios and the assumed electric propulsion specific-impulse/efficiency characteristics. Insertion times less than 100 days were found for satellites with high power-to-mass ratios that used high thrust-to-power electric propulsion options. The influence of power for electric propulsion beyond that used for satellite payloads and housekeeping was also assessed, and insertion times less than a month will require powers significantly higher than presently installed.

Nomenclature

F_P	=	fraction of communication satellite beginning-of-life power added for the quasi stage
g	=	acceleration due to gravity, m/s^2
I_{SP}	=	apogee engine specific impulse, s
M_{BOL1}	=	initial estimate of communication satellite geosynchronous-Earth-orbit beginning-of-life mass, kg
M_{BOL2}	=	adjusted communication satellite beginning-of-life mass with integrated electric propulsion, kg
M_{BOLQ}	=	adjusted communication satellite beginning-of-life mass with quasi-stage electric propulsion, kg
M_{GTO}	=	calculated communication satellite geosynchronous-transfer-orbit mass, kg
M_{GTO1}	=	cited communication satellite mass in geosynchronous transfer orbit (from the Union of Concerned Scientists database), kg
M_O	=	beginning-of-life communication satellite mass, kg
P_D	=	reference Hall effect thruster discharge power, 4.5 kW
P_{diss}	=	power dissipated from power processor, kW
P_{HET}	=	power into the Hall effect thruster subsystem, kW
P_O	=	beginning-of-life communication satellite power, kW
α_{HET}	=	Hall effect thruster subsystem-specific mass, 7.2 kg/kW
α_{PV}	=	solar array specific mass, 20 kg/kW
ΔM_{EPI}	=	mass penalty for integrated electric propulsion subsystems, kg
ΔM_{EPQ}	=	additional communication satellite beginning-of-life mass penalty for the quasi-stage electric propulsion, kg
ΔV	=	launch-site-specific Delta V from geosynchronous transfer orbit to geosynchronous Earth orbit, m/s
η_C	=	reference Hall effect thruster magnet and cable efficiency, ~0.98
η_{PPU}	=	reference Hall effect thruster power processor efficiency, ~0.93

I. Introduction

A CONJUNCTION of developments has greatly increased the practicality and expectations for use of electric propulsion (EP) for energetic Earth-orbit transportation functions with mission energies equal to or greater than those for transfer of commercial communication satellites (COMSATs) from geosynchronous transfer orbits (GTOs) to geosynchronous Earth orbits (GEOs). Key among these advancements are trends of beginning-of-life (BOL), GEO COMSAT power-to-mass ratios and masses, an increasingly competitive situation for COMSAT launchers, major advancements in space power technology, and the broad and increasing use of EP.

Figure 1 shows average BOL COMSAT on-orbit power-to-mass ratios and masses as a function of the year of launch. COMSATs considered herein were those in the Union of Concerned Scientists (USC) database [1] that met the conditions described later in Sec. II.A and included nearly all recent COMSATs. Figure 1 shows that the average power-to-mass ratio of COMSATs has increased dramatically over the last two decades and the average mass has also risen strongly. As seen later, the transportation times with EP are directly reduced by higher power-to-mass ratios and the decreasing times implied by Fig. 1 are critical positive factors regarding the use of EP for high-energy transfers of COMSATs to GEO. The increased masses with time also increase the leverage of EP, which grows with increasing mission energy.

The relatively sustained and significant COMSAT market has contributed to an extremely competitive situation for launch vehicles. Figure 2 shows the number and launch sites for COMSATs launched from 2002 to 2006 inclusive and for future COMSATs

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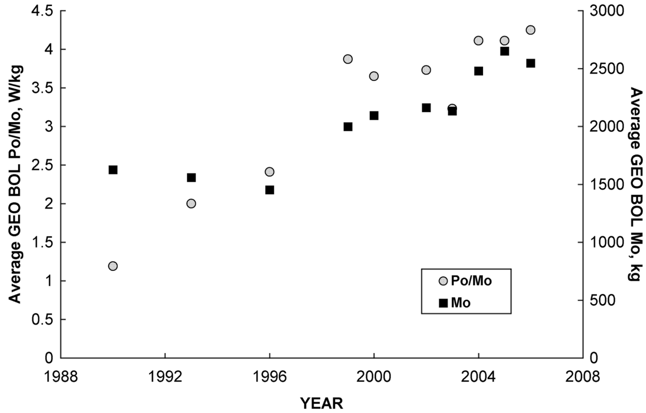

Fig. 1 Trends of COMSAT BOL power-to-mass ratios and masses.

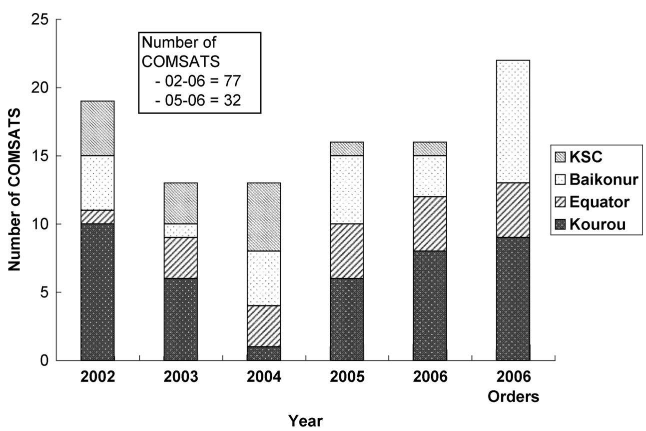

Fig. 2 COMSATs launched by year and launch site.

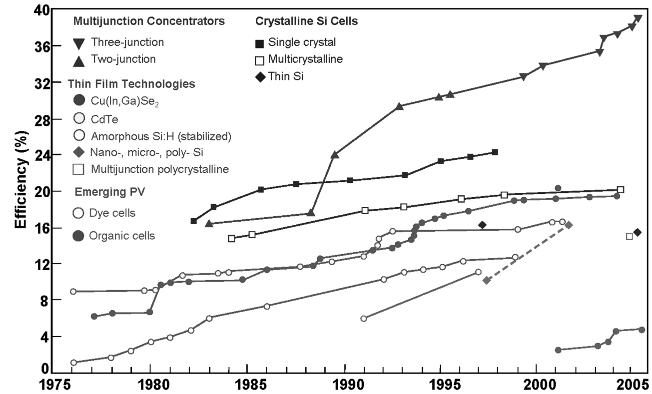

Fig. 3 Solar cell efficiencies versus time [4].

Propulsion Function \ Prime	Alcatel-Alenia	Astrium	Boeing	ISAS	Lockheed	Loral	Mitsubishi	Orbital Sciences	Russian	Spectrum Astro	Swedish Space Corp.
Stationkeeping & Orbit Insertion	•	•	•		•	•	•	•	•		
Planetary ΔV				•				•		•	•
High Energy Earth - Orbit Transfer					•						•

Fig. 4 EP applications on operational spacecraft (ISAS denotes the Institute of Space and Astronautical Science).

ordered in 2006. Also, multiple new launcher options are under development that may soon be available for COMSAT launches. In-space propulsion can dramatically affect the fraction of the COMSAT market captured by specific launcher/launch-site options and is therefore a key competitiveness consideration for specific COMSAT missions. The capture fraction of launchers is important and is defined as the fraction of the selected 32 COMSATs launched in 2005 and 2006 that could have been delivered from specified launch sites with specified launcher GTO mass delivery capability and transfer propulsion characteristics. The competitive condition is similar to the case for initial acceptance of EP for station keeping, favorable for acceptance of EP for high-energy GEO insertions.

Spacecraft power has always had a predominant influence on the performance and acceptance of EP. This is especially so for primary propulsion functions, due to the increased leverage of power-system characteristics on the overall mission cost and performance. There have been rapid and significant improvements in all space power characteristics relevant to primary propulsion with EP [2–4]. In particular, and likely benefiting from very large terrestrial photovoltaic programs and competitions between different space solar array concepts, major advancements have occurred for space solar array specific costs and masses, environmental (including both radiation and plasma) robustness, packing density in launch fairings, and cell/array efficiencies. A review of space solar array characteristics is beyond the scope of this paper. However, the dramatic advancements in space solar arrays, such as that shown in Fig. 3 for BOL solar cell efficiency increases with time, have materially improved the potential performance and enhanced the expectations for use of high-power electric propulsion.

As shown in Fig. 4, EP has been broadly accepted for operational applications for multiple propulsion functions. EP is now routinely used for stationkeeping on spacecraft produced by an international set of satellite suppliers. In some cases, EP is also used for final GEO insertion (the so-called apogee-topping maneuver). Planetary missions have also been successfully performed by several countries, and additional planetary missions using EP are firmly planned. The Earth-orbit-transfer function in Fig. 4 is defined as missions with EP transfer energies about equal to or greater than those typical of COMSAT GTO-to-GEO transfers. The SMART-1 mission used EP for transfer of a scientific satellite from an Ariane 5 GTO to the moon [5,6]. This was the first planned use of EP for high-energy Earth-orbit transportation and it very successfully demonstrated the functions required for low-thrust transfers and, importantly, the robustness of standard spacecraft designs for Earth-orbit environments, including the effects of an extraordinary solar storm [7]. The advanced extremely-high-frequency satellites will soon use EP for high-energy transfers to GEO. These applications are notable in that they will use EP transfers of over 100 days for Earth-orbit transportation of very-high-value assets [8].

The developments cited previously provide strong bases for selection of EP for Earth-orbit transportation of COMSATs to GEO. This use of EP, with and without a separate chemical propulsion subsystem for part of the transfer, has been extensively studied. Many evaluations, particularly the earlier examples, assumed low-Earth-orbit (LEO) starting orbits, often with EP powers much higher than typical of present COMSATs [9–12]. GTO and other initial orbits have also been assumed in many recent analyses [13–18]. In general, the previous studies evaluated the effects of EP on the GEO payload with specified masses in the orbits in which EP operation was initiated. The analyses herein adopt alternate approaches to maximize relevance to near-term acceptance of EP for high-energy Earth-orbit transfers. The GEO payloads are the COMSATs from the Union of Concerned Scientists (UCS) database [1] with BOL powers and masses obtained as discussed in detail subsequently. Also, integrated EP subsystems are assumed with input power levels of 0.95 of the BOL COMSAT powers. Use of recent COMSAT designs and integrated EP should minimize impacts on present spacecraft designs. Launcher GTO delivery capabilities necessary to deliver the specific COMSATs were then calculated for different launch sites as a function of the characteristics of the chemical and electric in-space propulsion subsystems. It was then possible to evaluate the COMSAT market capture of launchers as a function of their GTO delivery capability and launch site. Knowledge of the capture capability and the associated delivery times may be useful to providers of COMSAT services, launchers, and spacecraft.

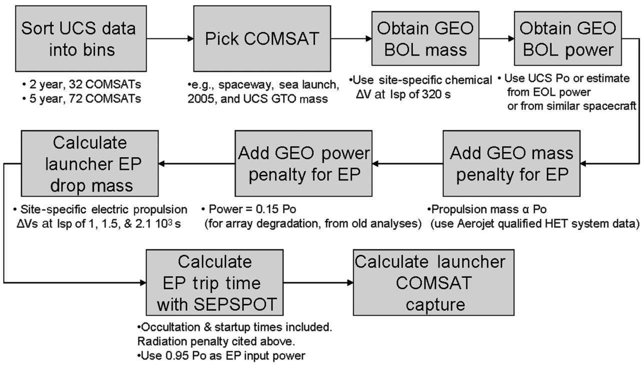

Fig. 5 COMSAT mission capture and insertion-time calculation flow.

II. Analyses

The overall goals of the analyses were to quantify the COMSAT market capture potential of launchers and the associated trip times from GTO to GEO as functions of GTO delivery capability, launch site, and chemical and electric transfer propulsion characteristics. Figure 5 shows the overall approach, which will be briefly described subsequently.

A. Reference COMSAT Data Set and Chemical Apogee Subsystem

The UCS online spacecraft data sets [1] were the primary sources of COMSAT information. In all cases, data were checked against other public online sources, which resulted in a small number of modifications of the UCS databases. The UCS data set includes only active spacecraft so that the number of spacecraft cited as launched in a given year decreases with time. The masses delivered to GTO were always provided, and in many cases, the BOL or end-of-life (EOL) power was also given. When power data were not available, estimates were made as discussed subsequently. On the basis of available data, COMSATs were limited to those launched from Baikonur, the equator, Kourou, or the Kennedy Space Center (KSC); provided by Western primes; planned solely for civil communications; and operated in GEO. These constraints eliminated a very small number of COMSATs launched in 2005 and 2006, and the number launched per year is shown in Fig. 2. As seen in Fig. 1, critical COMSAT characteristics vary with time. UCS data from 1990 to 2006 were used to generate Fig. 1. However, as a compromise between sample size and relevance to present COMSAT designs, the set of 32 COMSATs launched in 2005 and 2006 was used for all other analyses presented herein. Nearly all calculations shown used the characteristics of the individual COMSATs as a starting point. This approach was taken to maximize the relevance of the results to near-term COMSAT designs. Although EP transfers can provide nonincremental increases in GEO masses, exploitation of those benefits may require significant changes to present COMSAT concepts with potential schedule and resource implications.

The initial COMSAT mass in GEO was the payload of interest but was almost never available, and an initial estimate for a selected COMSAT BOL mass was obtained using the following equation:

$$M_{\mathrm{BOL1}} = [\exp -(\Delta V / g I_{\mathrm{sp}})] M_{\mathrm{GTO1}} \tag{1}$$

The Delta V in Eq. (1) is that associated with both the actual launch vehicle site and a chemical apogee propulsion system. Table 1 shows the four launch sites, their associated inclinations, and the chemical and electric propulsive Delta V used herein. The Delta V assumed standard GTOs obtained from launcher entities with perigees and apogees at LEO and GEO, respectively. The Baikonur Delta V assumed use of the low-inclination (31 deg) option. Options for a combination of both chemical and electric propulsion transfers to GEO have been studied, and a standard GTO starting orbit may be less than optimal for electric propulsion transfers [16,17]. It is recognized that slightly different GTOs are sometimes used, and in a few cases, EP has been used for the final phase of transfer: apogee-topping. However, details of GTOs and such EP applications are rarely available for specific launches. Therefore, typical single values of Delta V in Table 1 were calculated assuming an apogee burn for simultaneous circularization and plane change. Electric propulsion Delta V in Table 1 are calculated for time optimal transfers [18]. Herein, only pure chemical or electric propulsion options from a standard GTO starting orbit were considered, both for reasons of convenience and to allow direct comparison.

Table 1 Delta V for GTO–GEO transfers

Launch site	Inclination, deg	GTO–GEO ΔV, m/s Chemical	Electric
Baikonur	~46	1837	2647
Equator	0	1478	2187
Kourou	7	1502	2215
KSC	28.5	1831	2641

The reference chemical apogee propulsion subsystem was assumed to have an I_{sp} of 320 s. This value was representative of recent-vintage chemical apogee engines from Europe [19] and the United States [20]. Of course, different chemical apogee engines with slightly different I_{sp} were used on different COMSATs. However, details of chemical apogee propulsion are seldom provided for specific launches, and the global mission impact due to use of different state-of-art chemical apogee systems is very small. In these preliminary analyses, only the propellant, as determined from the Delta V of Table 1 with no margins added, was specifically eliminated from the chemical apogee propulsion subsystem. It was assumed that the chemical and electric propellant tanks were of equal mass and so neither was specifically accounted for in these analyses. That assumption is very conservative for the higher-EP specific-impulse cases considered. The masses of the remainder of the chemical apogee propulsion subsystem, including the pressurant storage and control elements, lines and associated thermal control and structure, thruster(s) and any associated thermal control, and apogee-subsystem-specific structure were not considered. It was assumed that those masses would conservatively account for redundancies, nonintegral numbers of single strings of thrusters, and uncertainties associated with the EP subsystem. Future analyses that include models of a chemical apogee propulsion subsystem are recommended to obtain refined, and less conservative, estimates of masses.

B. Integrated Electric Propulsion

COMSATs were launched from four sites, and to be relevant to individual entities, it was necessary to normalize the information to allow estimates of the market capture and trip times for specific launcher GTO delivery capabilities and launch sites. The following describes the approach used to obtain those data for the integrated EP options. The approach for evaluation of a quasi-stage EP approach will then be discussed.

Integrated EP subsystems were assumed herein unless otherwise stated. This approach assumed that the EP power was less than that installed for other bus and payload requirements and that most other functions required for EP transfer (such as guidance, navigation, and control and spacecraft management) were also provided by available COMSAT subsystems. One deviation from this philosophy is the potential use of power dissipated from the power processors of the EP subsystem for thermal control during the EP transfer to GEO. It was the intent that the EP transfer subsystem would be separate from other COMSAT propulsion functions. Details of the configuration and thermal integration of the integrated EP subsystems will be highly spacecraft-specific and are beyond the scope of this paper. It was assumed, however, that the EP gimbal(s), thruster(s), and associated mounting structure could be placed where the chemical apogee engine is usually located and the PPUs and associated structure would be placed inside the spacecraft box, as dictated by thermal considerations. With this approach, the transfer EP subsystem is independent and would not interfere with stationkeeping, repositioning, or attitude control system functions.

The PPU dissipated power, which would be about 0.07 of the BOL COMSAT power, was available, along with some fraction of the 0.05 P_O not used for EP, as appropriate for thermal management. It is assumed that the available power is adequate for thermal control during the EP transfer phase. It is recognized that this assumption will depend on the specific COMSAT thermal management approach, including considerations of deployable versus fixed radiators, variable conductance versus other heat transfer concepts, appropriate thermal survival temperature limits, etc. The analyses used a representative specific mass for the Hall effect thruster (HET) subsystem and did not attempt to identify specific numbers of thruster strings. This approach results in a nonintegral number of thruster strings and the mass margin to account for that was handled as described previously. Finally, HET subsystems are typically qualified for operating times in excess of 200 days, with a margin [21], and so are judged to be appropriate without modification for the COMSAT transfer missions described herein.

The BOL COMSAT power was a key parameter for the analyses. The USC database or online data often provided spacecraft BOL or EOL power levels. In all cases in which only the EOL power was given, the BOL power was estimated by increasing the EOL power by 15% to account for degradation during the GEO mission. The degradation is, of course, a function of the solar cell technology and GEO lifetime, and 15% degradation was used as representative of the different recent GEO spacecraft situations. When no power data were available, the BOL power was estimated using data from similar spacecraft from the same manufacturer. The input power to the integrated EP system was then assumed to be 0.95 of the BOL COMSAT power. This approach constrained the EP power level to less than that installed for GEO COMSAT operations and also accounted for housekeeping power to the COMSAT during the transfer to GEO.

It was then necessary to adjust the initial COMSAT BOL mass to account for propulsion and power penalties associated with the use of integrated EP. The propulsion-subsystem penalty was taken to be the BOL power times a representative specific mass of a HET subsystem. A HET approach was chosen to maximize the thrust-to-power-ratio options available to minimize EP transfer times. The qualified BPT-4000 subsystem [21], with additions for gimbals and structure, was used to calculate a reference HET-subsystem-specific mass, and Table 2 provides the values used. For the integrated EP subsystems, it was assumed that all the electric propulsion elements except the power processor radiate directly to space and so do not impose a thermal-control penalty. The power processor was assumed to use the COMSAT thermal control system and involve no additional penalties, beyond a fraction of the 0.2 structure allocation, as the rejected heat that requires a radiator is of the order of 0.07 of that of the operational COMSAT. Specific-impulse values of 1000, 1500, and 2100 s were selected for analyses, as they were consistent with the range of data available. The thruster thrust-to-power ratios, or efficiencies, were obtained from the highest values in [22] at the three cited values of specific impulse [22]. To account for electrical line losses, an additional 2% efficiency loss was then applied to the data.

An arbitrary 15% solar array mass penalty, at an assumed specific mass of 20 kg/kW, was also added to account for degradation of the array during the transfer with EP. This penalty is conservative, especially due to the advances in solar array environmental robustness and assumption of GTO start orbits in which the total time spent in the Van Allen belts during EP transfers to GEO has been calculated to be less than one day [13]. For reference, the degradation of the SMART-1 solar arrays due to Van Allen belt effects [6] was about 0.08, and lower degradations are anticipated with next-generation solar arrays.

Table 2 Reference 5 kW HET system masses

System element	Mass, kg	Comments
Thruster	12.3	Includes cable and brackets [21]
Power processor	12.75	——
Propellant management	0.5	——
Gimbal	4.1	Estimated at 33% of thruster
Structure	5.93	Estimated at 20% of subsystem
Total HET subsystem	35.6	——

The integrated EP mass penalty added to the COMSAT BOL mass then was obtained via use of Eqs. (2–5). The total power of the reference HET subsystem was

$$P_{\mathrm{HET}} \approx (P_D)/(\eta_{\mathrm{PPU}}\eta_C) \approx 5\ \mathrm{kW} \tag{2}$$

The specific mass of the reference HET subsystems was then

$$\alpha_{\mathrm{HET}} \sim 7.2\ \mathrm{kg/kW} \tag{3}$$

The specific mass of array power was assumed to be

$$\alpha_{\mathrm{PV}} \sim 20\ \mathrm{kg/kW} \tag{4}$$

A penalty of 0.15 of the initial power was assumed from Eq. (4) that is equivalent to an added specific mass penalty of 3 kg/kW. The mass penalty added to the BOL COMSAT mass to account for integrated EP is then

$$\Delta M_{\mathrm{EPI}} \approx [3 + 7.2]P_O \tag{5}$$

The total power of the COMSAT was used to calculate the EP mass penalty, and the slight overestimate of EP mass due to the assignment of 0.05 P_O to housekeeping functions was ignored.

The final GEO BOL mass was then calculated as

$$M_{\mathrm{BOL2}} \approx \Delta M_{\mathrm{EPI}} + M_{\mathrm{BOL1}} \tag{6}$$

After the GEO BOL mass is estimated, the associated GTO mass may be calculated for a given launch site (with an associated GTO–GEO Delta V) and transfer propulsion specific impulse:

$$M_{\mathrm{GTO}} \approx \exp(\Delta V/gI_{\mathrm{sp}})]M_{\mathrm{BOL2}} \tag{7}$$

Herein, the Kourou and KSC launch sites are considered to illuminate the effects of launch-site inclination. The Delta V were taken from Table 1 for the appropriate launch sites and propulsion approaches. The GTO mass in Eq. (7) is that required of a launcher to deliver the selected COMSATs to GEO from the launch-specific GTO with a transfer subsystem that operates at the specified I_{sp}.

In the case of an assumed chemical apogee transfer approach, no changes were assumed to the spacecraft, and the GTO masses were varied only to be consistent with the different Delta V for chemical propulsion shown in Table 1.

The GTO masses required for each of the 32 COMSATs launched in 2005–2006 were calculated for Kourou and KSC launches. The mission capture of the 32 COMSATs launched in 2005–2006 may now be calculated as a function of launcher GTO delivery-mass capability and transfer propulsion characteristics.

The transfer times with EP were calculated using the SEPSPOT [23] code. SEPSPOT is a time-optimal-trajectory tool for geocentric-attitude-constrained analyses. In initial analyses, occultation and EP startup times were included. The startup penalty was imbedded in the SEPSPOT code and was representative of engine conditioning times when condensable mercury propellant was used with ion thrusters. The startup times are therefore longer than appropriate for inert-gas propellants, but were used for convenience. Transfer times were first calculated for situations that represented extremes of COMSAT and EP subsystem characteristics. Occultation and startup times increased transfer times, relative to times calculated without those penalties, by a maximum of 7% [18]. To expedite analyses, therefore, most trip times presented were obtained with a 7% increase in SEPSTOP results calculated without consideration of occultation or startup penalties. Although available, the array degradation feature of SEPSPOT was not used, as it did not reflect recent solar array degradation characteristics. Instead, as cited previously, an array mass penalty was added to the BOL mass of the COMSATs.

C. Quasi-Stage System

Some calculations were made in which power equal to $F_P P_O$ was added to the installed COMSAT power levels to evaluate EP transfer times with increased COMSAT power-to-mass ratios. It is important to note that the added power is not available for GEO payloads, as that would require an increase in the COMSAT mass and reduce the overall power-to-mass ratios to near-original values. Consistent with an assumed 15% degradation in the Van Allen belts, the addition of a power of $F_P P_O$ will require a mass of 1.15 $F_P P_O$. Also, the additional power will require more propulsion mass. Unlike the case for integrated EP, it was assumed that a separate thermal rejection system would be required for additional power. Detailed analyses of the thermal subsystem are beyond the scope of the paper, but based on [24], the specific mass of thermal control for the PPU is 31 kg per dissipated kilowatt, P_{diss}.

From the preceding, the quasi-stage concept requires an additional mass beyond that of the integrated EP approach of

$$\Delta M_{\mathrm{EPQ}} \approx \{1.15\alpha_{\mathrm{SA}} + \alpha_{\mathrm{HET}} + 31(1 - \eta_{\mathrm{PPU}})\} F_P P_O \tag{8}$$

The BOL COMSAT mass for the quasi stage is then

$$M_{\mathrm{BOLQ}} \approx \Delta M_{\mathrm{EPI}} + M_{\mathrm{BOL2}} + \Delta M_{\mathrm{EPQ}} \tag{9}$$

The GTO masses, capture fractions, and trip times are then calculated as described for the integrated EP approach.

III. Results and Discussion

Figure 6 shows the reductions in GTO mass requirements for Kourou and KSC launches resulting from the use of integrated EP, rather than chemical apogee propulsion with an I_{sp} of 320 s. The benefits with three values of I_{sp} of EP are shown versus the mass required in the respective GTOs if chemical apogee propulsion were used. Each datum was derived from a specific COMSAT launched in 2005–2006, with some excluded for clarity, and the lightest and heaviest COMSATs are represented by the extreme values of GTO masses. As an example, use of EP enabled reductions of KSC GTO masses of about 900 to 1300 kg at 1000 and 2100 s I_{sp}, respectively, for a COMSAT that required a GTO mass of about 4000 kg with chemical apogee propulsion. Overall, for EP I_{sp} from 1000 to 2100 s, respectively, integrated EP allowed fractional GTO mass reductions of around 0.2 to 0.3 at Kourou and 0.25 to 0.35 at KSC. The lower leverage of EP at Kourou is due to the reduced Delta V for GEO insertions that result from the lower inclination of the launch site. The expected decrease in launcher requirements to deliver COMSATs from Kourou relative to KSC is also reflected in Fig. 6. The data of Fig. 6 may also be used to contrast launcher requirements for specific COMSAT opportunities with both chemical and EP transfers to GEO. Although derived from specific COMSATs, the data of Fig. 6 are general and may also be used to evaluate the impact of integrated EP on dual-launch scenarios. However, in-space propulsion effects on dual launches are dependent on details of the spacecraft support systems in launch fairings and were not treated herein.

Figure 7 shows the capture fraction of launchers versus GTO drop-mass capability for launches from Kourou and KSC, respectively, with chemical and integrated electric propulsion. The capture fraction is defined as the fraction of the 32 COMSATs launched in 2005–2006 that could have been delivered with the specified launch site, GTO delivery capability, and transfer propulsion characteristics. It is seen from both launch sites that integrated EP provided significant gains in capture fraction of recently launched COMSATs for launcher GTO delivery capabilities between about 2500 and 5500 kg. Capture of the entire recent COMSAT market with chemical apogee propulsion required Kourou and KSC launcher GTO masses of about 6500 and 7500 kg, respectively. Integrated EP enabled capture of the full set of COMSATs with Kourou and KSC GTO delivery capabilities, respectively, of about 4400 and 5000 kg. These data show that for launches of single COMSATs, the leverage for integrated EP is greatest for launchers with GTO mass delivery capabilities less than 4500 to 5500 kg. This may offer an opportunity for significant cost reductions for COMSAT missions. However,

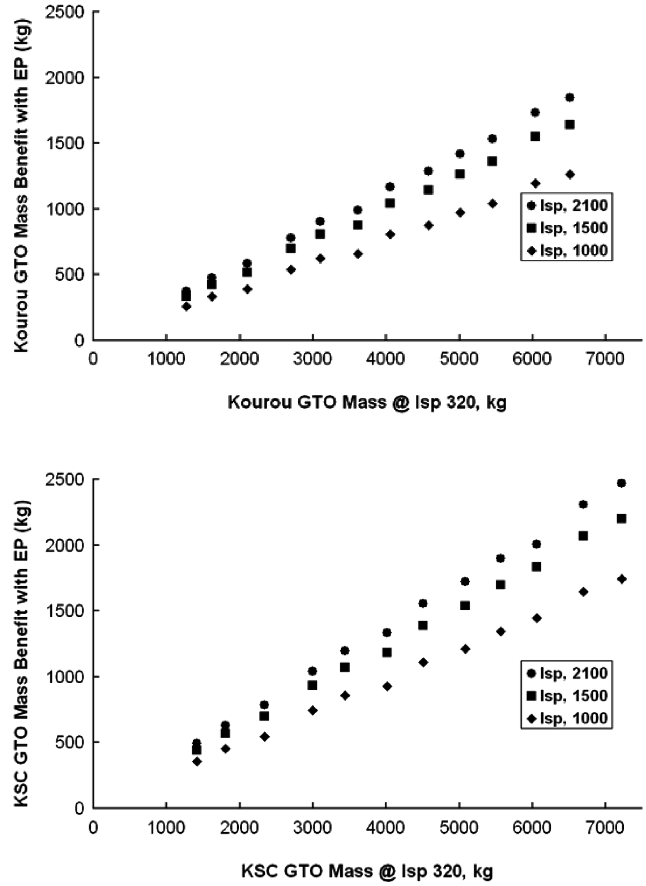

Fig. 6 Mass benefits with integrated EP for Kourou (top) and KSC (bottom).

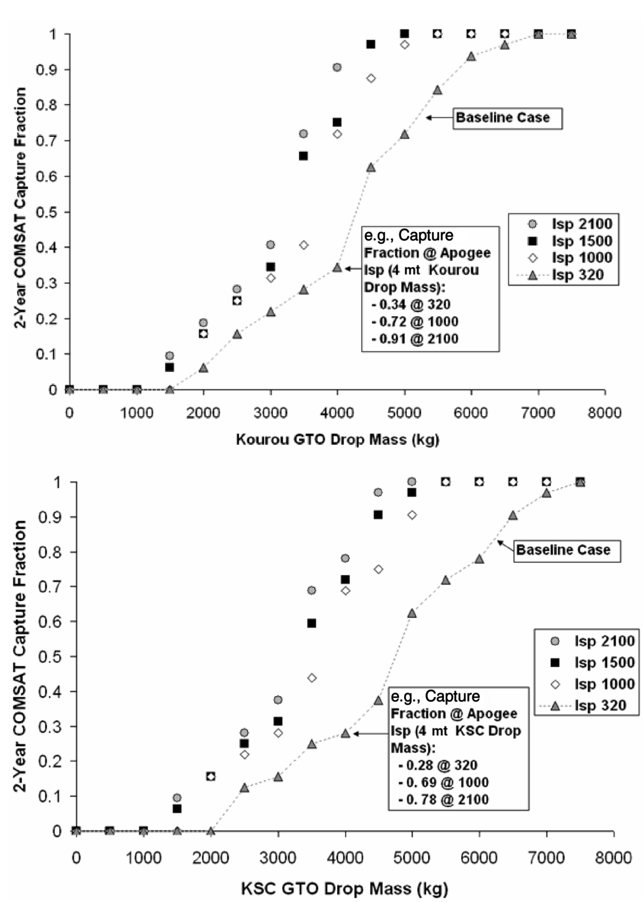

Fig. 7 COMSAT mission capture with integrated EP for Kourou (top) and KSC (bottom).

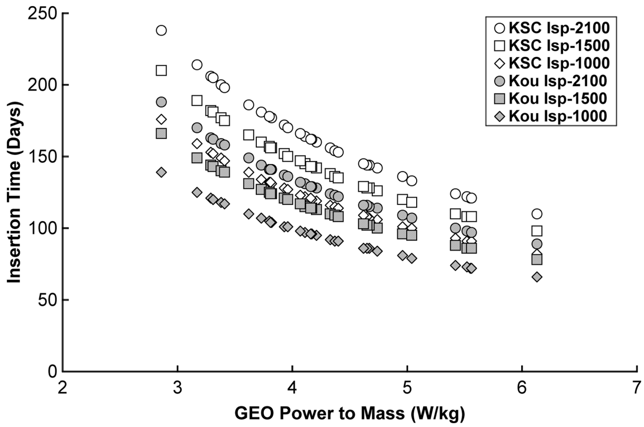

Fig. 8 COMSAT transfer time with integrated electric propulsion.

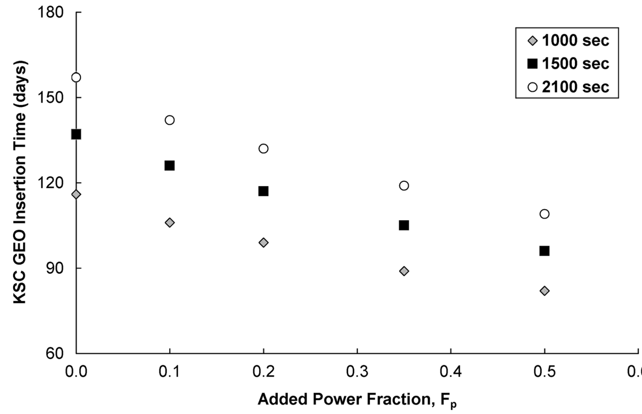

Fig. 9 COMSAT transfer times from KSC with added power fraction.

many relevant costs are generally not public, and so no attempt was made herein to quantify the resource implications of the calculated performances.

Transfer times to GEO with integrated EP are shown in Fig. 8 for the full set of recently launched COMSATs. A specific COMSAT is represented by a value of BOL power-to-mass ratio that ranged from slightly less than 3 to over 6 W/kg for recent COMSATs. Transfer times are shown for values of integrated EP I_{sp} of 1000, 1500, and 2100 s. For a specific launch site and specific impulse (and associated system efficiency) the trip time is very closely specified by the BOL power-to-mass ratio of the COMSAT. Therefore, the trend in average COMSAT power-to-mass ratios shown in Fig. 1 has enabled major decreases in potential COMSAT trip times to GEO with EP. As expected, transfer times at Kourou are shorter than those at KSC, due to the reduced Delta V required for the GTO-to-GEO transfers. Figure 8 also shows that the transfer times become less sensitive to power-to-mass ratio as that parameter increases. That has important implications for the attainment of short transfer times on the order of a month or less, which will be discussed subsequently.

Some reductions in integrated EP trip times from those shown in Fig. 8 should be available with little effort in the near term. First, much-less-conservative assumptions regarding the penalties for occultation are possible, which only limit the launch time of day during some seasons [18]. Startup-time penalties were taken as representative of mercury ion engines, but are negligible for the xenon thrusters assumed herein. The assumed solar array degradation was about twice those experienced on the SMART-1 mission and such penalties may be expect to decrease as new cells become available. Finally, as indicated in Fig. 1, there is a steady upward trend in average power-to-mass ratio of COMSATs. This trend is expected to continue and would naturally provide reduced insertion times. Preliminary estimates of the potential impacts of these pathways were made and insertion-time reductions of about 10% from those of Fig. 8 appeared possible. Additional trip-time reductions with integrated EP would require more complex and invasive or constraining approaches.

A quasi-stage concept was evaluated as a possible evolutionary path to full EP stages, in which all functions (including power) required for EP transfer are provided by the state. As described previously, power was added to the COMSATs to be used for the EP transfer, but the power is not available for normal on-orbit payloads. Added solar array power increases the overall COMSAT power-to-mass ratio, as the specific power of the added power is much higher than that of a full COMSAT, which contains many payload and bus elements with rather low specific powers. A COMSAT with a BOL power-to-mass ratio of about 4 W/kg was selected, as that was approximately the average value for recent COMSATs (Fig. 1). Figure 9 shows the GEO insertion times for that COMSAT as a function of the added power, expressed as a fraction of the original BOL power. An upper bound of added power of half the installed power was arbitrarily selected to illustrate the situation. It is seen that the insertion times dropped slowly with increasing added power. However, Fig. 9 indicates that enablement of insertion times on the order of a month or less requires additional powers significantly greater than those installed on present COMSATs. EP stages typically [9–13] use such an approach and are thereby able to reduce insertion times to values well below those attainable with any integrated EP concept.

IV. Conclusions

Multiple recent and ongoing developments have significantly increased the practicality and expectations for the use of EP for energetic Earth-orbit transfers. These developments include trends of COMSAT power-to-mass ratios and masses, an increasingly competitive launcher situation, major advancements in spacecraft power systems, and the broad and increasing acceptance of EP for multiple propulsion functions. The use of electric propulsion was evaluated for transfers of COMSATs from GTOs to GEOs. Recent communication satellite designs, normal launch vehicle delivery orbits, and integrated electric propulsion subsystems (with input powers less than nominal satellites powers) were assumed to minimize required changes to present and near-term launchers and spacecraft. The capture fraction of recent communication satellites that could have been delivered was evaluated versus launcher delivery capability, launch site, and in-space propulsion characteristics. Electric propulsion significantly increases the capture fraction of single COMSAT launches of launchers with GTO mass delivery capabilities less than about 4500 to 5000 kg and may provide interesting benefits to dual-launch options. Insertion times were found to be accurately specified by the power-to-mass ratio of the satellites and the assumed electric propulsion specific-impulse/efficiency characteristics. Insertion times less than 100 days were found for satellites with high power-to-mass ratios that used high thrust-to-power electric propulsion options. The influence of power for electric propulsion beyond that used for satellite payloads and housekeeping was also assessed, and insertion times less than a month will require powers significantly higher than those presently installed.

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References

[1] Grimwood, T., *UCS Satellite Database* [online database], http://www.ucsusa.org/satellite_database [retrieved 4 Jan. 2007].

[2] Piszczor, M. F., Jr., O'Neill, M. J., Eskenazi, M. I., and Brandhorst, H. W., Jr., "The Stretched Lens Array SquareRigger (SLASR) for Space Power," 4th International Energy Conversion Engineering Conference and Exhibit (IECEC), AIAA Paper 2006-4137, San Diego, CA, June 2006.

[3] Hausgen, P. E., Granata, J. E., Senft, D., Tlomak, P., and Merrill, J., "AFRL Thin Film Solar Cell Development and Upcoming Flight Experiments," 2nd International Energy Conversion Engineering Conference and Exhibit (IECEC), AIAA Paper 2004-5734, Providence, RI, Aug. 2004.

[4] O'Neill, M. J., "1000 W/kg Solar Concentrator Arrays for Far-Term Space Missions," *Space Technology and Applications International Forum (STAIF 2004)*, Vol. 699, American Inst. of Physics, Melville, NY, 8–11 Feb. 2004, pp. 875–882.

[5] Koppel, C. R., and Estublier, D., "The SMART-1 Hall Effect Thruster Around the Moon: In Flight Experience," International Electric Propulsion Conference Paper 2005-119, Oct.–Nov. 2005.

[6] Milligan, D., Gestal, D., Camino, O., Pardo-Voss, P., Estublier, D., and Koppel, C., "SMART-1 Electric Propulsion Operational Experience," International Electric Propulsion Conference Paper 2005-254, 2005.

[7] Camino, O., Alonso, M., Blake, R., Milligan, D., de Bruin, J., Gestal, D., and Ricken, S., "SMART-1: Europe's Lunar Mission Paving the Way for New Cost Effective Ground Operations (RCSGSO)," *6th International Symposium on Reducing the Costs of Spacecraft Ground Systems and Operations, (RCSGSO)* [CD-ROM], ESA SP-601, ESA, Noordwijk, The Netherlands, June 2005, Paper 43.1.

[8] Robinson, C. A., Jr., "Agile Antennas Aid Warriors," *Signal* [online journal], July 2005, http://www.afcea.org/signal/archives/july05.asp.

[9] Byers, D. C., "Upper Stages Utilizing Electric Propulsion," *Joint Army-Navy-NASA-Air Force (JANNAF) Propulsion Meeting*, CPIA-PUB-315-VOL-V, Chemical Propulsion Information Agency, Columbia, MD, 1980, pp. 69–84.

[10] Kaufman, H. R., and Robinson, R. S., "Electric Thruster Performance for Orbit Raising and Maneuvering," 18th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, AIAA Paper 82-1247, Cleveland, OH, June 1982.

[11] Gulczinski, F. S., III, and Shilling, J. H., "Comparison of Orbit Transfer Vehicles Using Mid-Term Power and Propulsion Options," International Electric Propulsion Conference Paper 2003-22, Mar. 2003.

[12] Woodcock, G., "Controllability of Large SEP for Earth Orbit Raising," 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, AIAA Paper 2004-3643, Fort Lauderdale, FL, July 2004.

[13] Kemp, G. E., Dankanich, J. W., Woodcock, G. R., and Wingo, D. R., "Solar Electric Propulsion Upper-Stage for Launch Vehicle Capability Enhancement," *54th Joint Army-Navy-NASA-Air Force (JANNAF) Propulsion Meeting*, JSC CD-49, May 2007.

[14] Jenkin, A., "Representative Mission Trade Studies for Low-Thrust Transfers to Geosynchronous Orbit," AIAA/AAS Astrodynamics Specialist Conference and Exhibit, AIAA Paper 2004-5086, Providence, RI, Aug. 2004.

[15] Kluever, C. A., "Geostationary Orbit Transfers Using Solar Electric Propulsion with Specific Impulse Modulation," *Journal of Spacecraft and Rockets*, Vol. 41, No. 3, May–June 2004, pp. 461–466. doi:10.2514/1.10939

[16] Oh, D., Randolph, T., Kimbrel, S., and Martinez-Sanchez, M. "End-to-End Optimization of Chemical-Electric Orbit Raising Missions," *Journal of Spacecraft and Rockets*, Vol. 41, No. 5, Sept.–Oct. 2004, pp. 831–839. doi:10.2514/1.13096

[17] Oleson, S. R., Myers, R. M., Kluever, C. A., Riehl, J. P., and Curran, F. M., "Advanced Propulsion for Geostationary Orbit Insertion and North-South Station Keeping," *Journal of Spacecraft and Rockets*, Vol. 34, No. 1, Jan.–Feb. 1997, pp. 22–28. doi:10.2514/2.3187

[18] Dankanich, J. W., and Woodcock, G. R., "Electric Propulsion Performance from GEO-Transfer to Geosynchronous Orbits," 30th International Electric Propulsion Conference Paper 2007-287, Sept. 2007.

[19] Shulte, G., "High Performance 400N MMH/NTO Bipropellant Engine for Apogee Boost Maneuvers," 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Los Angeles, AIAA Paper 99-2466, June 1999.

[20] Stechman, C., Woll, P., Fuller, R., and Colette, A., "A High Performance Liquid Rocket Engine for Satellite Main Propulsion," 36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Huntsville, AL, AIAA Paper 2000-3161, July 2000.

[21] De Grys, K., Welander, B., Dimicco, J., Wenzel, S., Kay, B., Khayms, V., and Paisley, J., "4.5 kW Hall Thruster System Qualification Status," 41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, AIAA Paper 2005-3682, Tucson, AZ, July 2005.

[22] Martinez, R., de Grys, K. H., Hoskins, W. A., Welander, B., Beal, B. E., and Haas, J. M., "Development and Characterization of a High Performance Hall Thruster Operating at Specific Impulses from 1,000 to 3,000 Seconds," *54th Joint Army-Navy-NASA-Air Force (JANNAF) Propulsion Meeting*, JSC CD-49, May 2007.

[23] Sackett, Lester L., Malchow, Harvey L., and Edelbaum, Theodore N., "Solar Electric Geocentric Transfer with Attitude Constraints: Analysis," NASA CR-134927, Aug. 1975.

[24] Byers, D. C., Terdan, F. F., and Myers, I. T., "Primary Electric Propulsion for Future Space Missions," NASA TM-79141, May 1979.

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