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System Analysis and Design of the Geostationary Earth Orbit All-Electric Communication Satellites

Parsa Abbasrezaee[1,*], Ali Saraaeb[2]

1.Sapienza University of Rome – Aerospace Engineering School – Rome – Italy. 2.Khaje Nasir Toosi University of Technology – Aerospace department – Tehran – Iran.
*Corresponding author: parsa.abbasrezaee@gmail.com

ABSTRACT

With the help of gathered data and formulas extracted from a previous conference paper, the all-electric geostationary Earth orbit (GEO) communication satellite statistical design was conducted and further studied with analytic hierarchy process (AHP) and technique for order of preference by similarity to ideal solution (TOPSIS) methods. Moreover, with the help of previously determined system parameters, the orbital ascension, orbital maintenance and deorbiting specifications, calculations and simulations were persuaded. Furthermore, a parametric subsystem design was conducted to test the methods reliability and prove the feasibility of such approach. The parametric subsystem design was used for electrical power subsystem (EPS), attitude determination and control system (ADCS), electric propulsion, telemetry, tracking and control (TT&C) in conceptual subsystem design level, which highly relies on the satellite type and other specifications, were concluded in this paper; other subsystem designs were not of a significant difference to hybrid and chemical satellites. Eventually, the verification of the mentioned subsystems has been evaluated by contrasting the results with the *Space mission engineering: the new SMAD*, and subsystem design book reference.

Keywords: All-electric; GEO; AHP and TOPSIS Method; Maintenance; Deorbiting; Parametric.

INTRODUCTION

From the previous conference paper, the contrast between all-electric geostationary Earth orbit (GEO) communication and other hybrid and chemical satellite design has shown that using all-electric satellite design has many advantages. Generally, all-electric satellite designs have a 50% total mass reduction associated with an explicit decrease of satellite fuel mass from 45% or 50% to 15% of the satellites total mass, more than 50% reduction in launcher cost and a total life span increase up to 20 years (Abbasrezaee *et al.* 2019). However, all-electric satellite designs have its drawbacks in their orbital ascension that can take from a minimum of 3 months up to 6 months to reach GEO depending on the satellite mass and its propulsive thrust, which is very high in comparison to few weeks of orbit ascension for chemical and hybrid satellites. Therefore, causing a long-time exposure to radiation, forcing the use of thicker shielding for sensitive electronic equipment and solar cells during all-electric satellite

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orbit ascension. The system analysis and design conducted in this paper consisted of statistical system design, orbit transfer, orbit maintenance, deorbiting specifications and parametric subsystem design. For a better understanding of the statistical design approach of the GEO all-electric satellite, AHP (analytic hierarchy process) and TOPSIS (technique for order of preference by similarity to ideal solution) methods persuaded to evaluate. Also, in this paper, orbit ascension, orbit maintenance, deorbiting [end of life (EOL)] simulation and calculation and the parametric subsystem design for electrical power subsystem (EPS), attitude determination and control subsystem (ADCS), telemetry, tracking and control subsystem (TT&C), electric propulsion have been calculated for the GEO all-electric satellites since these subsystems are different with the hybrid and chemical ones (Wertz *et al.* 2011; Brow 2002).

STATISTICAL OVERVIEW ON GEO ALL-ELECTRIC SATELLITE

Statistical design and development of other methods than this paper for GEO satellites could be found in literature references (Agrawal 1986; Chetty 1991; Davidoff 1990; Griffin 1991; Martin 1996). All mentioned methods references are based on multistage algorithms and complicated parametric analysis. Besides that, at the first steps of satellite design most of the required inputs are unknown or known with uncertainty. The proposed statistical design approach below helps designers to do their jobs quickly and with the least number of requirements.

Geostationary Earth orbit communication satellites have been investigating the space since 1964. Initially used for the summer Olympics, now more than 2,000 communication satellites have communicated information to various places globally, most of which are located in GEO orbit. However, the technological developments for more than five decades from the first communication satellites until now have led to GEO all-electric communication satellites, which utilize electric propulsion for orbit transfer and their orbit maintenance during the time of mission operations (Abbasrezaee *et al.* 2019).

The first all-electric satellites launched into Geostationary Transfer Orbit (GTO) on September 6, 1999, was the Russian Yamal-100 satellite, which failed to reach the GEO. However, two successful all-electric satellites were launched by the SpaceX Falcon 9 v1.1, on March 2, 2015, one of which was the ABS-3A (1954 kg), and the other one was the Eutelsat 115 West B (2205 kg) (Abbasrezaee *et al.* 2019). It is worth to mention that for this research all the statistical data had gathered from 2015 up until March 2018 as mentioned completely in the first conference paper (Abbasrezaee *et al.* 2019). Most of the all-electric satellites, according to the statistical data gathered, has a range of mass between 1500–3000 kg and they all consume Xenon based fuel type with a total propellant mass in the range of 200–400 kg (Abbasrezaee *et al.* 2019; AST and COMSTAC 2015; FAA 2018). All-electric satellites use less fuel in their orbital ascension and orbital maintenance, which gives the all-electric satellite an advantage in an extended mission operational period that may last from 15 to 20 years. Besides, this affects the overall design reducing its bus volume at an average of 40% (Abbasrezaee *et al.* 2019).

FORMULA AND TEST CASE

In the conceptual system and subsystem satellites design level, critical specifications like the number of transponders for satellites, payload mass, power consumption and electrical propulsion subsystem selection should be considered. From the previous conference paper, the formulas used in the statistical system design extracted to calculate the main conceptual design parameters of a GEO telecommunication all-electric satellite with the single input of transponder number; the formulas are as seen in the linear and exponential form (Abbasrezaee *et al.* 2019). After putting, the first input (X) of the number of transponders in the first formula, the payload mass (Y1) will generate and, by then, all the parameters will develop by the output of the predecessor formula or related predecessor formula. Generally, all the procedures have more than 0.8 accuracy, which indicates the compatibility of this statistical data for the conceptual design level (Table 1) (Abbasrezaee *et al.* 2019). The Y1-Y7 in Table 1 outputs are the main conceptual system design parameters for a GEO all-electric telecommunication satellite.

Table 1. All-electric satellite formula for calculating the conceptual level design parameters.

Formula	Formula accuracy	Input and output
$Y1 = 65.651 \times e^{0.0409\ x}$	0.9325	Input (X): number of transponders Output (Y1): payload mass (kg)
$Y2 = 1411 \times e^{0.0008\ Y1}$	0.9806	Input (Y1): payload mass (kg) Output (Y2): satellite total mass (kg)
$Y3 = 3.6147 \times e^{0.0007\ Y1}$	0.927	Input (Y1): payload mass (kg) Output (Y3): electrical power consumption of payload (kw)
$Y4 = 0.0034\ Y2 + 0.9759$	0.9887	Input (Y2): satellite total mass (kg) Output (Y4): electrical power consumption of satellite (kw)
$Y5 = 5.069\ e^{0.0006\ Y2}$	0.7466	Input (Y2): satellite total mass (kg) Output (Y5): satellite volume (m^3)
$Y6 = 0.012\ Y2 + 56.833$	0.9882	Input (Y2): satellite total mass (kg) Output (Y6): satellite cost (M$)
$Y7 = 11.921 e^{0.0005\ Y2}$	0.9371	Input (Y2): satellite total mass (kg) Output (Y7): launcher cost (M$)

Source: Abbasrezaee *et al.* (2019).

Consequently, for testing this statistical approach, a test case was conducted (Table 2) (Abbasrezaee *et al.* 2019). Generally, this method is divided into two parts, a part that has relied solely on formulas generated for interpolating system parameters, while the other part used data gathered to assign system parameters (Abbasrezaee *et al.* 2019).

Table 2. Results of formula calculation and assumption for GEO all-electric satellite test case.

Calculations	Statistical information	Explanation
Payload mass = 337.09 kg	40 transponders for 1500–2000 kg all electric satellites	Calculated from formula extraction
Satellite total mass = 1847.75 kg	-	Calculated from formula extraction
Payload electrical power = 4.6 kw	4.5 kw payload power	-
Satellite electrical power = 7.26 kw	8 kw satellite power	-
Satellite total volume = 15.36 m^3	12–18 m^3	3.8 × 2 × 2 m
Satellite total cost = US$ 79 mi	US$ 70–90 mi	-
Launchers cost = US$ 30.029 mi	US$ 30–35 mi half volume of Falcon 9 launcher	US$ 61.2 mi for 8000 kg payload of Falcon 9 launcher to GTO
0.165 m N thrust and 300 kg fuel	XIPS-25 propulsion	-
300 kg fuel + 337 kg payload mass +1250 kg bus mass = 1887 kg	BSS-702 SP bus	950–1250 kg bus mass of BSS-702 SP

Source: Abbasrezaee *et al.* (2019).

ANALYTIC HIERARCHY PROCESS TOPSIS TEST CASE EVALUATION

Rather than prescribing a “correct” decision, the AHP helps decision makers find one that best suits their goal and understand the problem. It provides a comprehensive and rational framework for structuring a decision problem, representing and quantifying its elements, relating those elements to overall goals and evaluating alternative solutions. Users of the AHP first decompose their decision problem into a hierarchy of more easily comprehended sub-problems, each of which can be analyzed independently. The hierarchy element can relate to any aspect of the decision problem tangible or intangible, carefully measured, or roughly estimated, well or poorly understood anything that applies to the decision at hand. Once the hierarchy is built, the decision-makers systematically evaluate its various elements by comparing them to each other two at a time, to impact an element above them. In making the comparisons, the decision-makers can use concrete data about the elements, but they typically use their judgement about the elements relative meaning and importance. It is the essence of the AHP that human judgments, and not just the underlying information, can be used in performing the evaluations. The AHP converts these evaluations to numerical values that can be processed and compared over the entire problem range. A numerical weight or priority is derived for each element of the hierarchy, allowing diverse and often incommensurable elements to be compared to one another rationally and consistently. This capability distinguishes AHP from other decision-making techniques. In the final step of the process, numerical priorities are calculated for each of the decision alternatives. These numbers represent the alternatives relative ability to achieve the decision goal, so they allow a straightforward consideration of the various action courses (Wikipedia 2020).

Test case evaluation conducted using AHP and TOPSIS methods (Barichard *et al.* 2009; Goepel 2019). Analytic hierarchy process method was conducted to generate a suitable statistical weight based on priorities given. Besides, the TOPSIS method is applied to categorize data based on their similarity to the ideal solution (Barichard *et al.* 2009; Goepel 2019). Analytic hierarchy process hierarchy is used since the system parameters dictate the decision-making process that will affect the function of subsystem design (Fig. 1). Analytic hierarchy process decision matrix for all-electric satellite is shown in Table 3.

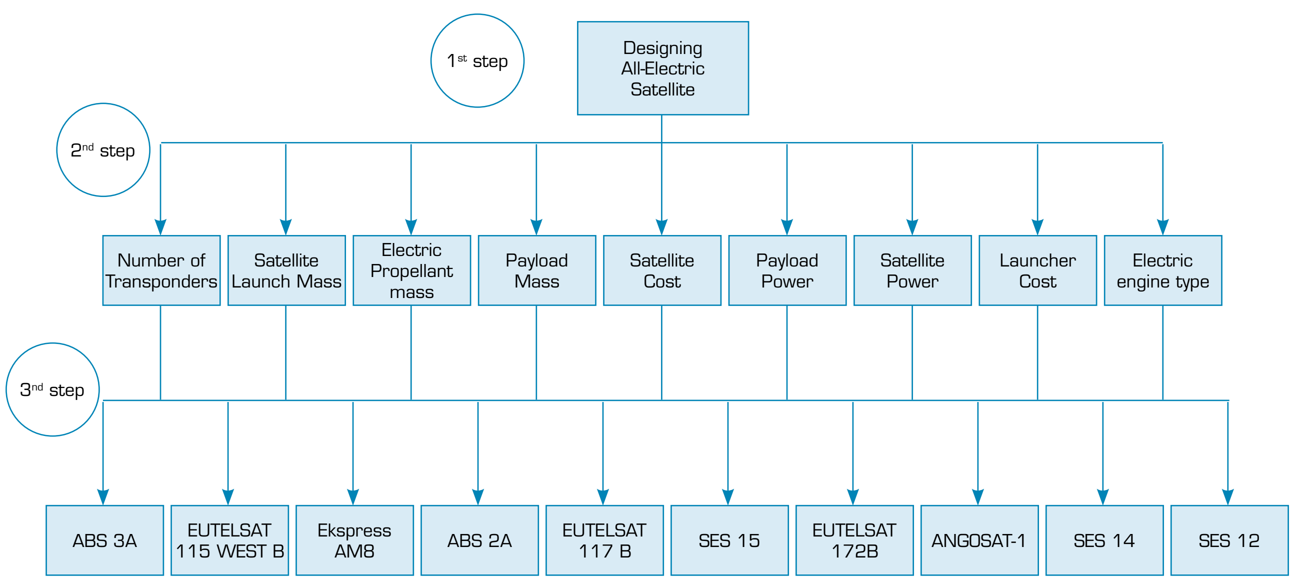

Figure 1. Analytic hierarchy process method hierarchy structure for all-electric satellite.

By utilizing an online AHP priority calculator on the website, the nine parameters of all-electric satellites have been prioritized in that calculator. Eventually, the numbers illustrate the critical factor of the number of transponders to payload mass, which means the number of transponders is five times more important to payload mass in systematic view or six times more than satellites mass, seven times more than electrical fuel mass, six times more than satellite cost, seven times more than payload power, seven times more than satellite power, five times more than launcher cost and eight times more than engine type (Goepel 2019) (Table 3).

All parameters have their factor number against the other parameters, which gave by human judgment. Therefore, the number of transponders have the most important rank and engine type with the list of important factors to the number of transponders. In Table 3, the number of the transponder to payload mass is 5, and payload mass to the number of transponders is 0.2. Moreover, the AHP priority calculator gives a consistency ratio (CR) number that shows all the relations factors between parameters are correct if they would be under 10 (Table 4).

Table 3. Analytic hierarchy process decision matrix table showing importance factor between all-electric satellite parameters.

Decision Matrix	1 Number of transponders	2 Payload mass	3 Satellite mass	4 Electric fuel mass	5 Satellite cost	6 Payload power	7 Satellite power	8 Launcher cost	9 Engine type
1 Number of transponders		5	6	7	6	7	7	5	8
2 Payload mass	0/2		3	5	5	7	7	5	8
3 Satellite mass	0/17	0/33		5	4	7	7	4	8
4 Electric fuel mass	0/14	0/2	0/2		1	2	2	1	3
5 Satellite cost	0/17	0/2	0/25	1		1	1	2	3
6 Payload power	0/14	0/14	0/14	0/5	1		1	1	2
7 Satellite power	0/14	0/14	0/14	0/5	1	1		1	2
8 Launcher cost	0/2	0/2	0/25	1	0/5	1	1		3
9 Engine type	0/12	0/12	0/12	0/33	0/33	0/5	0/5	0/33	

The decision matrix value is assigned based on a simple approach, which gives the highest priority to the number of transponders. Therefore, the higher number of transponders that a satellite has, the more services it can provide. Then, criteria related to the mass of the satellite are given a second priority to the number of transponders, since mass is a critical factor affecting all designing parameters. In conclusion, the decision matrix was a balance between the satellite mass and its services. Then, the AHP pairwise comparison applied to the decision matrix to deduce every criterion (Table 4).

Table 4. Analytic hierarchy process comparison priority percentages and ranks results for decision matrix parameters.

	Category	Priority	Rank
1	Number of transponders	40.1%	1
2	Payload mass	21.6%	2
3	Satellite mass	15.8%	3
4	Electric fuel mass	4.8%	4
5	Satellite cost	4.7%	5
6	Payload power	3.4%	7
7	Satellite power	3.4%	7
8	Launcher cost	4.2%	6
9	Engine type	2.0%	9
CR%	6.2%		

These results show that transponder and mass properties had 82.3% of the total statistical significance. Consistency ratio was equal to 6.2%, which is generally an acceptable value for statistical weight. The consequences resulted from the AHP pairwise comparison are essential to apply the TOPSIS method and for the ideal solution the assumptions below were presumed, noting that engine type has been converted to quantitative value from one to four from best to worst case, respectively, which is based on the previous statistical analysis of engine performance (Table 5) (Abbasrezaee *et al.* 2019).

Table 5. Ideal best/worst value assumptions for decision matrix parameters.

	Number of transponders	Payload mass	Satellite mass	Electric fuel mass	Satellite cost	Payload power	Satellite power	Launcher cost	Engine type
V+ (ideal best value)	Max	Min	Min	Min	Min	Min	Min	Min	Min
V- (ideal worst value)	Min	Max	Max	Max	Max	Max	Max	Max	Max

The TOPSIS was conducted only for all-electric satellite data gathered from the previous paper, in which statistical performance was calculated and, based on it, every satellite ranked including the test case (Table 6). Satellites rank differ because of all priorities and satellites, which have low total mass, a high number of transponders and low payload mass. Most of the satellite data are in the conference paper, such as satellite total mass, number of transponders, launcher types, etc. Table 6, the ANGOSAT-1, test case satellite and EUTELSAT 117 B are the best in rank with AHP and TOPSIS evaluation, since these satellites have 40–60 number of transponders, which is the highest factor to choose. Besides, the payload mass and satellite total mass is low for these satellites, which has the second and third factor in evaluations. The test case satellite has 40 transponders and 1887 kg satellite total mass, 337 kg payload mass, lower than the average 2000 kg satellite total mass and below 400 kg of average payload mass. ANGOSAT-1 has 1647 kg total mass with near 30 transponders; however, the EUTELSAT 172 B has 3350 kg total mass with 45 transponders, which the total mass factor leads this option to be in 10 ranks.

Table 6. The TOPSIS method performance and rank results for selected all-electric satellite.

Name of satellite	Performance (%)	Rank
ABS 3A	91.64183676	5
EUTELSAT 115 WEST B	88.50431894	8
Ekspress AM8	89.01029446	7
ABS 2A	90.9337616	6
EUTELSAT 117 B	91.74572388	4
SES 15	83.84680972	9
EUTELSAT 172B	54.32213346	10
ANGOSAT-1	93.60404619	2
SES 14	23.85183676	11
SES 12	9.898662583	12
Test case	93.47590506	3
V+ (ideal best value)	99.99999949	1
V- (ideal worst value)	5.4293E-07	13

From the results shown in the table, it is possible to conclude, based on statistical performance values, that the test case parameters chosen satisfy the criteria given enough and have an excellent performance of 93.47590506 %. However, the test case did rank directly the best to ideal ANGOSAT-1 was better and ranked the second to the excellent best value with a small difference in performance that can be ignored. The method conducted is realistic and accurately balanced and tuned towards the goal intended, which is having the best balance between the satellite mass and services.

ORBIT TRANSFER, ORBIT MAINTENANCE AND DEORBITING

In the simulation, all-electric satellite total mass in statistical data offers a clear suggestion to divide all-electric satellites into four satellite mass categories 1500, 2000, 2500 and 3000 kg. This division eased comparison between these new sub mass categories, which simplified the calculation of main satellite characteristics, such as orbit transfer, orbital maintenance, deorbiting fuel mass consumption, orbit transfer time and thrust needed with different launchers and different electric propulsion subsystem characteristics (Abbasrezaee *et al.* 2019). Geostationary Earth orbit characteristics were considered in the simulations, such as orbit inclination (i) 0° (with 0.01 accuracy), deorbiting (e) 0° (with 0.001 accuracy), and semi-major axis (a) 42160 km (with 0.01 accuracy). Four launchers were chosen from data sampled based on their high percentage of launcher use in statistic paper (Abbasrezaee *et al.* 2019). Normally, the electric propulsions subsystem consists of two pair of thrusters mounted on an adjustable 3-axis arm (Luebberstedt *et al.* 2018). The 3-axis arm is advantageous since it provides better thermal control, less electromagnetic interference (EMI) and better compatibility with assembly integration and test (AIT) for propulsion wiring and propulsion fuel pumping (Luebberstedt *et al.* 2018; Naclerio *et al.* 2012; Senbely *et al.* 2017). Earth to GTO and GEO, illustrated in Fig. 2, shows launchers mission from Earth to the beginning of the GTO (Proton-M-Brize-M launcher) and satellites, which start from the GTO to GEO (Fig. 2 and Table 7) (Zak 2021).

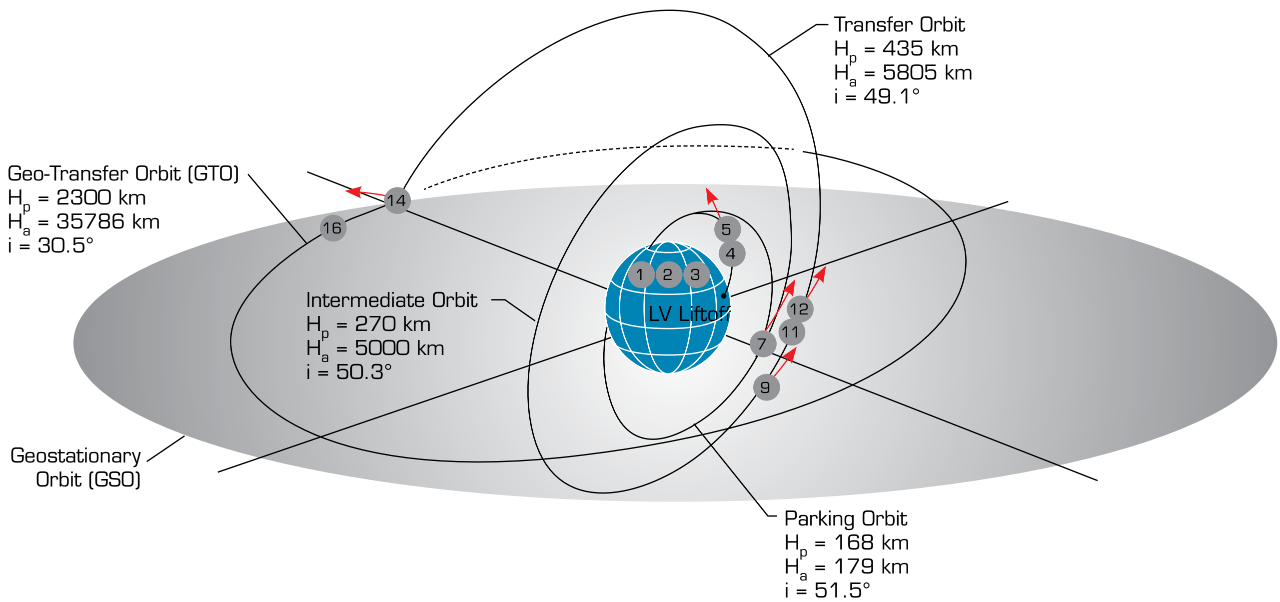

Figure 2. Proton-M-Brize-M launcher GTO to GEO mission. Source: Zak (2021).

Table 7. Sampled GTO launcher specifications from references for simulations.

Specification	Soyuz-fregat	Proton-M-Brize-M	Falcon 9 v,11	Ariane 5 ECA
Inclination (i)	6	30.5	28.5	6
Altitude of perigee (ap)	250	2300	185	250
Altitude of apogee (aa)	35950	35786	35786	35943

Source: AST and COMSTAC (2015) and FAA (2018).

For GTO simulation, the Soyuz-Fregat launcher was chosen, since it has a low GTO inclination release and proper orbit specification. This issue will decrease fuel mass consumption for orbit ascension of the satellite (Tables 8–12) (FAA 2018; Lugtu 1990).

The System Tool Kit (STK) software is used to calculate the fuel mass consumption and time for three main modes of orbit ascension (GTO), orbit maintenance (GEO) and deorbiting. One pair arm thruster force vector is towards the direction the satellite should go in orbit ascension and deorbiting mode. Still, this vector will be towards the satellite center of mass during the orbit maintenance mode in GEO orbit (Luebberstedt *et al.* 2018; Naclerio *et al.* 2012). For comparison between different propulsion subsystem performances of different satellite mass categories, Soyuz specification presumed as launcher specification with four different satellite propulsion subsystem specifications (Table 8). This use of different propulsions showed different variations in fuel mass consumption of orbit ascension and time (per days, months) mentioned in Table 8; these results were simulated for all the assumed mass categories (1500 to 3000 kg).

Table 8. Sampled electric propulsion specification for simulations.

Propulsion Specification	Thrust (N)	ISP (s)	Efficiency (%)	Type of thruster	Name of company	Country
XIPS-25 (Goebel *et al.* 2009)	0.165	3550	71	ION	L-3 Communication	USA
SPT-140D (Kim 2012)	0.29	1770	68	Hall-effect	OKB Fakel	Russia
PPS 1350 (Safran 2021)	0.089	1700	50	Hall-effect	Snecma (Safran Aircraft Engine)	French
BHT20K (Leporini *et al.* 2016)	1.08	2750	72	Hall-effect	Busek	USA

Source: Goebel *et al.* (2009), Kim (2012), Safran (2021) and Leporini *et al.* (2016).

Table 9. Orbit transfer GTO to GEO results of SPT-140D propulsion calculated with STK.

Weight of all electric satellites(kg)	Total xenon fuel (kg)	Fuel consumption for orbit transfer (kg)	Time (day-month)
1500	250	143.98	99.74–3.32
2000	300	189.105	131–4.366
2500	350	234.36	162.36–5.41
3000	400	279.59	193.69–6.456

Table 10. Orbit transfer GTO to GEO results of XIPS-25 propulsion calculated with STK.

Weight of all electric satellites(kg)	Total xenon fuel (kg)	Fuel consumption for orbit transfer (kg)	Time (day-month)
1500	200	71.238	173.96–5.8
2000	250	71.238	232.67–7.75
2500	300	108.41	283.25–9.44
3000	300	129.72	338.88–11.28

Table 11. Orbit transfer GTO to GEO results PPS 1350 propulsion calculated with STK.

Weight of all electric satellites(kg)	Total xenon fuel (kg)	Fuel consumption for orbit transfer (kg)	Time (day-month)
1500	250	149.67	324.49–10.82
2000	300	196.706	426.46–14.21
2500	350	243.72	528.37–17.61
3000	400	290	630.42–21

Table 12. Orbit transfer GTO to GEO results BHT20K propulsion Calculated with STK.

Weight of all electric satellites(kg)	Total xenon fuel (kg)	Fuel consumption for orbit transfer (kg)	Time (day-month)
1500	2o0	91.41	28.53–0.95
2000	250	1120	37.76–1.2
2500	300	150.56	46.99–1.56
3000	350	180	56.21–1.87

Trajectory optimization was not utilized and the trajectory chosen was by default; therefore, by intuition, the amount of fuel consumption and the time to reach GEO can be reduced. The BHT20K propulsion based on the simulated results is the best time from GTO to GEO and in fuel amount consumption. However, the electric power consumption of BHT20K propulsion contradicts since, based on the hypothesis of design of this study, a satellite cannot merely provide this amount of electric power or in best case scenario dedicating such a considerable amount of electrical power for this purpose will decrease the margin of other functionalities. Consequently, the BHT20K propulsion filtered out. As a result, XIPS-25 was the best in all regards and it was selected based on the simulated results among other propulsion subsystems for all-electric satellite design.

Orbit maintenance is essential for all space satellites, especially for GEO communication satellites, because of space satellite environment disturbances, such as solar radiation pressure, third body gravity effect (Sun and Moon gravity) and Earth gravity harmonies. Furthermore, GEO satellites need very high nadir pointing accuracy for their telecommunication antenna reflectors to function appropriately. Geostationary Earth orbit satellites have North/South orbit correction and East/West orbit corrections. North/South orbit correction is utilized for inclination (i) disturbances and East/West orbit corrections are used for eccentricity (e) and longitudinal disturbances corrections. Orbital maintenance was not subjected to any simulation, it was directly calculated based on the formula provided by Nasa Handbook in reference for 20 years of commission for 2000 and 3000 kg all-electric satellite (Lovell and O'Maley 1970; Lugtu 1990). Besides, orbital maintenance was not calculated for all the sub mass categories since relatively the contrasted fuel mass consumption between mass categories is small in amount (1500–2000 and 2500–3000 kg). The results for North/South and East/West orbit corrections have been calculated from reference formula for two all-electric satellite mass categories and the result in Tables 13–15 show that more than 95% of the fuel consumed in orbital maintenance will be consumed in North/South corrections (Lovell and O'Maley 1970; Lugtu 1990).

Table 13. Twenty years North/South orbit correction for all-electric satellites calculated results.

Parameters	2000 kg satellite	3000 kg satellite
Days between each maneuver	4.32	
Total number of maneuvers in 20 years	1691	
Total Δv in 20 years (m/s)	907.245	
Total fuel consumption in 20 years (kg)	71	105.82

Table 14. Twenty years East/West orbit correction for all-electric satellites calculated results.

Parameters	2000 kg satellite	3000 kg satellite
Days between each maneuver	10	
Total number of maneuvers in 20 years	730	
Total Δv in 20 years (m/s)	33.58	
Total fuel consumption in 20 years (kg)	2.6	3.9

Table 15. Sum of 20 years orbit maintenance and correction results from Tables 13 and 14.

Parameters	2000 kg satellite	3000 kg satellite
Total Δv in 20 years (m/s)	940.825	
Total fuel consumption in 20 years (kg)	73.6	109.71

The deorbiting process is essential for eliminating obsolete telecommunication and weather forecasting GEO satellite, where all deorbiting satellites will settle in GEO graved orbit (500 hm above their missions' orbit) (Wikipedia 2021). Therefore, GEO deorbiting simulated in STK for different propulsion subsystem for 2000 and 3000 kg all-electric satellite systems (Tables 16 and 17).

Table 16. Deorbiting 2000 kg GEO telecommunication all-electric satellite with different propulsions STK simulated results.

Parameters	XIPS-25	SPT-140D	BHT20K	PPS1350
Total fuel consumption (kg)	3.02	6.47	4.18	6.76
Time (days)	6.5	4.48	1.21	14.65
ΔV (m/s)	55.68	55.68	55.84	55.84

Table 17. Deorbiting 3000 kg GEO telecommunication all-electric satellite with different propulsions STK simulated results.

Parameters	XIPS-25	SPT-140D	BHT20K	PPS1350
Total fuel consumption (kg)	4.84	9.70	6.25	10.1
Time (days)	11.82	6.72	1.80	21.9
ΔV (m/s)	55.84	55.84	55.84	55.84

The XIPS-25 propulsion is the most efficient propulsion subsystem and consumes the least amount of Xeon fuel. Fuel consumption simulated and calculated for all the main modes associated with the satellite in orbit (orbit ascension, orbit maintenance, and deorbiting). This result for 2000 and 3000 kg presumed system in 277–409 kg fuel mass for PPS 1350 propulsion, 165.7–244.5 kg fuel mass for XIPS-25 propulsion, 270–400 kg fuel mass for SPT-140D propulsion, and 197.8–295.9 kg fuel mass for BHT20K propulsion. All data extracted from the simulation are accurate since the input data used from real propulsions with different specifications.

Subsystem conceptual parametric design

Subsystem parametric design consists of using different formulas for each subsystem calculation, gathering statistical information about various parts used in the same class of satellites and utilizing essential input data from datasheets to choose different subsystems information needed in formulas. All-electric satellites are unique in their electrical propulsion subsystem design, practically the 3-axis arm configuration, which necessitates an efficient enough electrical power supply providing the needed electrical power. Electric power generation and distribution (EPS), ADCS and propulsion, which are the subsystems to have differences in all-electric design with hybrid and chemical ones, were subjected to the parametric design process, which will be represented later in this section. Since this satellite is a communication one, in this case, the design of the TT&C and payload subsystem should also be conduct with the parametric design. Not forgetting that for all-electric satellites, a thicker protective shielding is required for its long exposure to Van Allen radiation during transferring orbit from GTO to GEO to protect solar cells and electronic equipment.

Telemetry and tele-command and communication payload system

Telemetry, tracking and control subsystem transfers and receives telemetry beckons data from the ground station. For GEO telecommunication satellite, multi-communication missions could be considered for GEO telecommunication satellites, such as fix satellite services (FSS), movable satellite services (MSS), broadcast satellite services (BSS), etc. (Mirshams et al. 2015). Therefore, TT&C parametric design starts with selecting payload communications and TT&C related subsystem parts from data available and then it ends with calculating payload mass and electric power consumption. Telemetry, tracking and control subsystem consists of antenna, antenna reflector, traveling-wave tube amplifier (TWTA), multiplexer (input, output), LNA and transponders, where the chosen transponder dictates and puts the main terms in about other parts. Selected communication and TT&C subsystem parts set from Thales Alenia, Tesat Spacecom, RUAG and other companies. The average number of transponders for all-electric satellites were in the range of 40 to 60 transponders, where their KU, C and KA bands were the most used types for all-electric satellite, results and calculation conducted are in Tables 18 and 19. The number of transponders is 65, five more than the average presented, to have adequate design quality for payload mass and power budget.

Table 18. Telemetry, tracking and control mass and power calculated results.

Parameters	Numbers	Dimension	Power cons (w)	Mass (kg)	Total power cons (w)	Total mass (kg)
KU telemetry (beacon) transponder	4 (2 redundant)	215 × 162 × 50 mm³	11.5–15.3	1.4	5.6	30.6
LNA	4	41 × 20 × 9 mm³	0.5	0.032	0.128	2
Input multiplexer	1	50 channels	-	0.14 per channel	7	-
Output multiplexer	1	50 channels	-	0.14 per channel	7	-
TWTA	4	282.5 × 75 × 62.5 mm³	30	0.76-3.2	12.8	60
KU antenna	4	-	-	0.5	2	-
Antenna reflector	1	222 mm	0	0.5	0.5	0
Total power consumption (w)	92.6					
Total weight (kg)	33					

Table 19. Telecommunication payload mass and power budget results.

Parameters	Numbers	Dimension	Power cons (w)	Mass (kg)	Total power cons (w)	Total mass (kg)
KU transponder	40	72 × 245 × 165 mm³	20	1.5	800	60
KU antenna	40	-	-	0.5	-	20
C transponder	15	150 × 282 × 195 mm³	44	4	660	60
C antenna	15	-	-	0.7	-	10.5
KA transponder	5	215 × 140 × 175 mm³	40	3	200	15
KA antenna	5	-	-	0.8	-	4
S transponder	5	253 × 175 × 130 mm³	13-30	3	150	15
S antenna	5	-	-	0.8	-	4
LNA	80	41 × 20 × 9 mm³	0.5	0.032	40	2.6
Input multiplexer	5	50 channels	-	0.14 per channel	-	35
Output multiplexer	5	5	-	0.14 per channel	-	35
TWTA	80	80	30	0.76–3.2	2400	160
Antenna reflector	3–6	3–6	0	5	-	15–30
Total power consumption (w)	4190					
Total weight (kg)	451.1					

Electrical power production and distribution subsystem design

Electric power supply and distribution consist of solar cell panels, battery packs, electric power regulators, power distribution units and cables. The latest technological developments in batteries and solar cells should be considered since they significantly affect output design parameters. The scrutinizing of statistical data related to the electrical power subsystem leads the consideration of two modes for all-electric satellites such as 8 kW for 2000 kg and 12 kW for 3000 kg satellites for power production EOL. Besides, batteries and solar cells type considered from the statistical results to be Li-ion (VES 180) from the France SAFT company and 3G30C-Advanced TJ (Ga-Ar) from the AZUR SPACE company, respectively. Most of the formula have been used from the NASA handbooks (Martinez-Sanchez and Pollard 1998; Corey and Pidgeon 2009; Flood and Weinberg 1994; Rauschenbach 1976). The primary required calculations for the electric power subsystem are illustrated in Table 20.

Table 20. Primary calculated results for electric power subsystem.

Parameters	Results for 8 kW EOL	Results for 12 kW EOL
P = orbit period (min)	1436	1436
TE = eclipse period (min)	72	72
Psa = power generated solar array (W)	10,061.4376	15,092.156
PC = power output from one cell (W)	0.728	0.728
NC = number of arrays for Psa	13,821	20,731
AArray = solar area (m^2)	46.35	69.52
Vmp = solar cell voltage after radiation (at EOL) at maximum temperature (voltage)	1.7852	1.7852
NS = number of solar in series	59	59
Imp = maximum current at 80 °C temperature taking into account for the maximum distance loss (A)	0.484	0.484
VCell = solar cell string voltage (voltage)	1.913	1.913
VString = voltage of string (v)	112.8	112.8
Pstring = power of string (w)	48.08	48.08
NP = number of cells in parallel	228	342
ANet = area of solar cells (m^2)	40.5	60.89
Total A = total area with space between cells	45	67.65
MArray = mass of arrays (kg)	21	31.3
CV = capacity of batteries (w/h)	13,333.34	20,000
MBattery = mass of batteries (kg)	81	122
MPGS = mass of arrays, batteries, regulations and cabling (kg)	112	168

For calculations of the electrical power subsystem expenses, the mass of solar cell panels should be considered. Without this consideration, the mean average mass calculated can be meager and unrealistic. However, to start the calculations, solar cell panels hinges, or any other detailed overview of the system was not considered. This resulted in 112 and 168 kg for 8 and 12 kW electric power EOL of the satellite. After that, the number of cells in series and strings have been calculated for 8 and 12 kW EOL satellites. By considering that solar panels consist of many sandwich panels, hinges and opening mechanism. Knowing that almost all GEO satellites solar panels consist of equally two-sided solar wings from the top and bottom of the satellite surface, this configuration is chosen during calculations. The calculation for considering solar panel size is illustrated in Table 21.

Table 21. Calculations for considering solar panels size.

Parameters	8 kw EOL	12 kw EOL
Solar area (m^2)	45	67.65
Number of cells in series	59	59
Number of strings (parallel)	228	342
Dimension of single solar cell (mm^2)	40 × 80	
Max panel dimension (m^2)	2.8 × 3.4	

The total wing area, number of panels, the square panel dimensions and the panel area have been calculated and optimized for 8 and 12 kW EOL power of all-electric satellites (Tables 22, 23).

Table 22. Results of calculations after considering solar panels size.

Parameters	8 kw EOL				12 kw EOL			
Wing area (m^2)	22.5	22.5	22.5	22.5	33.825	33.825	33.825	33.825
Number of panels	2	3	4	5	3	4	5	6
Panel area(m^2)	11.25	7.5	5.62	4.5	11.275	8.456	6.765	5.6
Dimension of square panel (m^2)	5.625	3.75	2.81	2.25	5.63	4.228	3.38	2.8

Table 23. The contrast between different numbers of panels per wings.

Parameters	8 kw EOL	12 kw EOL
Number of panels per wing	3	6
N (cells in series)	59	59
N (cells in parallel)	38	29
A length per wing mm	3114	2376
B width per wing mm	2476	2476
Panel area (m^2)	7.71	5.89
Total solar array area (m^2)	46.26	70.68

The sandwich panels, adhesive and panel coverings types are considered AL5056-50-3, K13C2U and Hexply 954-6 from the statistical analysis. After making a tradeoff between the weight, the number of panels and available panel sizes, the 3-panels form for 8 kW and 6-panels form for 12 kW has been considered for each wing.

The total mass budget for 8 and 12 kW EOL electric power subsystem calculated 283.14 and 439.8 kg, respectively (Table 24).

Table 24. Results of total solar arrays mass budget for two wings.

<table>
<tr><th>Parameters</th><th>Panels weight (kg)</th><th>Number of panels per wing</th><th>Mass of hinges (kg) (2 pcs/panel)</th><th>Mass of HDRM (kg) (4 pcs/panel)</th><th>Total array mass (kg)</th></tr>
<tr><td>8 kw EOL</td><td>39.39</td><td>3</td><td rowspan="2">1.5</td><td rowspan="2">1.2</td><td>283.14</td></tr>
<tr><td>12 kw EOL</td><td>28.85</td><td>6</td><td>439.8</td></tr>
</table>

Propulsion subsystem design

The propulsion subsystem design should be conducted for hybrid and chemical satellites to recognize the thruster number and configuration. However, the all-electric satellite configuration is the 3-axis arm configuration by convention and there is no need to consider other designing options since it is nearly optimal. As shown in the previous section, XIPS-25 propulsion was best in almost all main aspects of consideration (specific impulse [ISP], thrust, weight, electric power consumption) for an electrical propulsion subsystem. All-electric propulsion units consist of xenon tank (XST), fill and drain valve (FDV), pyro valve (PV), xenon filter (XEF), power processing unit (PPU), thruster module assembly (TMA) and Xenon Regulator and Feed System (XRFS). Power consumption varies significantly between orbit raising and orbit maintenance modes. Hence, the orbit-raising mode consumes more electric fuel mass for a longer duration to deliver the satellite. The electrical power would be twice as near as orbit maintenance mode (Table 25) (Naclerio *et al.* 2012; Corey and Pidgeon 2009).

Table 25. All electric satellites propulsion subsystem design parameters.

	Name of parameters	Parameters	Number	Total unit weight (kg)
Thruster	XIPS	25	4	54.8
	Input max power to thrusters (w)	4250		
	Thruster efficiency (%)	71		
	ISP (s)	3550		
	Thrust (mN)	165		
	Xenon mass flow (mg/s)	4.71		
	Weight (kg)	13.7		
PPUs	Power supply name	25 cm	3 (one redundant)	63.9
	Input max power to power supply (w)	4500		
	Efficiency (%)	94		
	Bus voltage (v)	97–103		
	Dimension (mm^3)	21 × 54 × 35		
	Weight (kg)	21.3		
Xenon Assembly	Propellant supply assembly (kg)	23	4	92
	Xenon propellant (kg)	200–450	1	200–450(differ from 2000–3000 kg satellites)
Total electric propulsion weight (kg)			411–661	
Total electric propulsion mass without propellant (kg)			210.7	
Total power consumption (w)			2500 (station-keeping) – 4500 (orbit raising)	

Attitude determination control subsystem design

In GEO communication satellites, ADCS is used during the mission to neutralize the internal and external disturbances, such as the third gravity effect, Earth gravity harmonies, solar radiation pressure and to maintain a high Nadir pointing accuracy for satellite, which is so critical for a functional GEO satellite (Table 26). Reaction wheels and other common satellite elements used, wherein these case all-electric satellites have had an active 3-axis control with 0.005-degree accuracy. Statistical data maintained rise numerous considerations for the number and type of sensors that should be used, so six sun sensors (two redundant), 2-star trackers (one redundant), four gyroscopes (two redundant), two GPS (one redundant), and four reaction wheels (one redundant) were considered. Attitude determination control subsystem design starts with calculating the satellite disturbance, in which significant external disturbances should be considered for reaction wheels momentum calculations. The momentum of reaction wheels should be ten times more than the most significant disturbance to accurately control the satellites, maintaining high nadir pointing accuracy (Wertz *et al.* 2011; Brow 2002). Sensors and reaction wheels were selected based on data gathered from datasheets (Table 27).

Table 26. Geostationary Earth orbit disturbance calculations for reaction wheels momentum determination.

Radiation solar pressure	Enharmonic earth gravity pressure	Reaction wheels momentum needed
1.9012×10^{-5}	4.1853×10^{-7}	4343 N.m.s × 10 = 4.343 N.m.s
The biggest disturbance for GEO orbit		

Table 27. The mass and electrical power consumption of the ADCS subsystem selected parts.

Parameters	Numbers	Company name	Weight (kg)	Electric power consumption (w)	Total weight (kg)	Total power consumption (w)
Reaction wheel (RSI-18-220-45)	4 (1 Redundant)	Rockwell Collins	6.3	20-150	25.2	80-600
GYRO	4 (2 Redundant)	Honey wheel	4.7	32	20	128
Star tracker	2 (1 Redundant)	Hydra-TC	5.3	8	10.6	16
Coarse sun sensor	6 (2 Redundant)	Moog Inc	0.215	-	1.29	-
GPS	2 (1 Redundant)	Garmin	0.332	7.8	0.664	15.6
Total weight (kg)	57.754 (without electric boards and electric propulsion)					
Total power consumption (w)	240–760 (without electric board and electric propulsion)					

The significant difference between the ADCS of all-electric satellites to other designs is the all-electric satellite ADCS. This way, it does not require separated thrusters and the same 3-axis arm propulsion mechanism will be used for orbit maintenance and ADCS modes simultaneously. Therefore, other designs need separate thrusters requiring the additional fuel tank to separate orbital ascension mode from orbital maintenance; that ADCS subsystem is responsible for these maneuvers.

Subsystems verification

Results of conceptual parametric subsystem designs should be contrasted with statistical data or with one of the primary satellite parametric design references. For instance, *Space mission engineering: the new SMAD* book, which is one of the best satellite-designing books, is used to contrast the subsystems budgets with the main power and mass calculations with the percentages of space mission engineering (Wertz *et al.* 2011). All percentages of different subsystems have been calculated, evaluated and illustrated in Tables 28–31. In addition, all contrasting results of percentages for parametric and new SMAD book are less than 15 %, which is very accurate for this level of designing. As mentioned earlier, the propulsion subsystem would be in the ADCS mass and power budget values. Consequently, there is no need to calculate separately for all-electric satellites. For electric power consumption, SMAD budgets are small since this book evaluated with the previous technology of solar cells and batteries (Wertz *et al.* 2011; Brow 2002). The EPS, ADCS, TT&C subsystems verification are illustrated in Tables 28–31.

Table 28. *Space mission engineering: the new SMAD* power and mass budget for GEO satellites.

Parameters	Mass (%)	Power (%)
ADCS	5–10 of total mass	10–30 of total power
EPS	10–30 of total mass	7–30 of total power
TT&C and PAY	18–30 of total mass	15–50 of total power
Propulsion (chemical) without fuel	3–8 of total mass	0–1 of total power

Source: Wertz *et al.* (2011) and Brow (2002).

Table 29. Electrical power subsystem comparison and verifications with design parameters and SMAD budget.

Parameters	2000 kg satellite	3000 kg satellite
Mass of parametric design (kg)	395.14	607.8
SMAD mass references (kg)	400	600
Error percentage	1.3%	1.3%

Table 30. Attitude determination and control system with propulsion subsystem and without propellant comparison and verification.

Parameters	Electrical power (W)		Mass (kg)	
	8000 w EOL sat	12000 w EOL sat	2000 kg sat	3000 kg sat
Parametric design budgets	2640	3160	269	269
References budgets	800–2400	1200–3600	100–250	150–300
Error percentage	10%	12.3%	7.6%	10.3%

Table 31. Telemetry, tracking and control and payload for 2000 kg, 8000 W all-electric satellite subsystem verification.

Parameters	Total weight (kg)	Total power consumption (W)
Parametric design budgets 2000 kg all-electric satellite	451.1	4190
References budgets for 2000 kg sat	360–600 (480 average considered)	1200–4000
Error percentage	6%	4.8%

CONCLUSION

Technique for order of preference by similarity to ideal solution and AHP method, which have been conducted, showing that the test case was consistent in the range of all-electric satellite system parameters, and validating such an approach feasibility with 93.47% accuracy. Moreover, it was necessary to disclose the uncertainty about orbit specifications; hence, calculations and simulations were conducted to validate the orbit ascension, maintenance and deorbiting with the presumed statistical propellant mass data. Besides, all-electric parametric subsystem design was still necessary to validate test case design based on subsystem availability, which was revalidated with less than 13% SMAD book percentages. The GEO all-electric satellite system conceptual design approach validated to be operational and opening the doors for detail design.

AUTHORS' CONTRIBUTION

Conceptualization: Abbasrezaee P and Saraaeb A; **Methodology:** Abbasrezaee P and Saraaeb A; **Investigation:** Abbasrezaee P and Saraaeb A; **Writing – Original Draft:** Abbasrezaee P and Saraaeb A; **Writing – Review and Editing:** Abbasrezaee P; **Funding Acquisition:** No funding; **Resources:** Abbasrezaee P; **Supervision:** Abbasrezaee P.

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The statistical data is in the conference RAST 2019 paper, avaialble at: https://ieeexplore.ieee.org/abstract/document/8767854

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REFERENCES

[AST] FAA Commercial Space Transportation, [COMSTAC] Commercial Space Transportation Advisory Committee (2015) 2015 Commercial Space Transportation Forecasts. Washington: FAA.

[FAA] Federal Aviation Administration (2018) The Annual Compendium of Commercial Space Transportation: 2018. Washington: FAA.

Abbasrezaee P, Mirshams M, Seyed-Zamani S (2019) Conceptual GEO Telecommunication All-Electric Satellite Design Based on Statistical Model. 2019 9th International Conference on Recent Advance in Space Technologies. IEEE, Stambul, Turkey.

Agrawal BN (1986) Design of Geosynchronous spacecraft. Upper Saddle River: Pretice Hall.

Barichard V, Gandibleux X, T'Kindt V (2009) Multiobjective Programming and Goal Programming: Theoretical Results and Practical Applications. New York: Springer.

Brown CD (2002) Elements of Spacecraft Design. Reston: AIAA.

Chetty PRK (1991) Satellite Technology and its Applications. New York: TAB books.

Corey RL, Pidgeon DJ (2009) Electric Propulsion at Space Systems/Loral. Paper presented 31th International Electric Propulsion Conference. IECP, Michigan, USA.

Davidoff MR (1990) Satellite experimenter's Handbook. Newington: American Radio Relay League.

Flood DJ, Weinberg I (1994) Advanced solar cells for satellite power systems. NASA Technical Memorandum 106777.

Goebel DM, Polk JE, Sandler I, Mikellides IG, Brophy JR (2009) Evaluation of 25-cm XIPS© Thruster Life for Deep Space Mission Applications. Paper presented 31st International Electric Propulsion Conference. IEPC, Michigan, USA.

Goepel KD (2019) AHP-OS. AHP Priority Calculator. AHP Criteria, Pairwise Comparison [Online]. https://bpmsg.com/ahp/ahp-calc.php

Griffin MD, French JR (1991) Space Vehicle Design. AIAA. https://doi.org/10.2514/4.862403

Kim V (2012) Stationary Plasma Thrusters in Russia: Problems and Perspectives. Tr Mosk Aviats Inst 60.

Leporini A, Giannetti V, Andreussi T, Pedrini D, Rossodivita A, Piragino A, Andrenucci M, Estublier D (2016) Development of a 20 Kw-class hall effect thruster. Paper presented Space Pripulsion 2016. Rome, Italy. http://www.sitael.com/wp-content/uploads/2016/05/3125196_DEVELOPMENT_20_kW-CLASS_HET_leporini.pdf

Lovell RR, O'Malley TA (1970) Station Keeping of High Power Communication Satellites [Online]. Technical Memorandum. Nasa. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19710001770.pdf

Luebberstedt H, Bastante JC, Lau M, Beekmans S, Tata M, Schneider A (2018) Electra- Full Electric Propulsion Satellite Platform for GEO Missions. OHB Electra.

Lugtu SD (1990) Impact of ion propulsion on performance, design, testing and operation of a geosynchronous spacecraft (Master thesis). Monterey: Naval Postgraduate School. In English.

Martin DH (1996) Communication satellites 1958-1995. El Segundo: Aerospace Corp.

Martinez-Sanchez M, Pollard JE (1998) Spacecraft electric propulsion-an overview. J Propuls Power 14(5): 688-699. https://doi.org/10.2514/2.5331

Mirshams M, Zabihian E, Zabihian AR (2015) Statistical design model (SDM) of communication satellites. Paper presented 2015 7th International Conference on Recent Advances in Space Technologies (RAST). IEEE, Istanbul, Turkey. https://doi.org/10.1109/RAST.2015.7208369

Naclerio S, Salvador JS, Such E, Avezuela R, Vara RP (2012) Small Geo Xenon Propellant Supply Assembly Pressure Regulator Panel: Test Results and Comparison with Ecosimpro Predictions. Paper presented 3rd edition of the International Conference on Space Propulsion. Bordeoux, France.

Senbely X, Wartelski M, Doubrère P, Deltour B, Cau P, Rochard F (2017) Design and Development of an Electric Propulsion Deployable Arm for Airbus Eurostar E3000 ComSat Platform. Paper presented 35th International Electric Propulsion Conference. IEPC, Atlanta, USA.

Wertz JR, Everett DF, Puschell JJ (2011) Space Mission Analysis and Design: The New SMAD. New York: Mcgraw-Hill.

Wikipedia (2021) Graveyard orbit [Online]. https://en.wikipedia.org/wiki/Graveyard_orbit

Wikipedia (2020) Analytic hierarchy process [Online]. https://en.wikipedia.org/wiki/Analytic_hierarchy process