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Optimal Orbit Parameters for Power Subsystem of LEO Satellites

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Abstract- The main objective of this paper is to design a Low-Earth-Orbit (LEO) Satellite Power Subsystem based on the variations in orbit parameters. Additionally, finding out the optimal orbit, altitude, and inclination angle; considering both solar array components and energy storage devices.

There are two main challenges in this work. The first challenge is to provide the Electrical Power Subsystem (EPS) with an additional power to compensate for the solar cell's degradation until End-of-Life (EOL) due to the presence of a rarefied atmosphere in LEO. The second challenge arises as a result of eclipses experienced by LEO more frequently and thus more charging/discharging cycles than satellites in Geosynchronous Earth orbit (GEO) leading to more stresses on the LEO batteries, which means higher degradation and lower performance.

Accordingly, the aforementioned challenges were tackled by the simulation and analysis based on the variations of the orbit parameters using Satellite Tool Kit software (STK8.1.1) to result in the optimal orbit, altitude, and inclination angle which provide maximum value for the total generated energy by the solar array during the mission. More importantly, we arrived at the minimum mass and area for solar array simultaneously with minimum values for the batteries': mass, volume, capacity, and total number of cycles, leading to a reduction in the charge/discharge cycles, i.e. lower degradation and higher performance.

Index Terms- Altitude, Beta angle, Inclination angle, LEO Satellite, Orbit parameters.

1 Introduction

Satellite electrical power subsystem plays an important role in its mission performance. To perform the mission successfully, the satellite should be provided with an additional power to compensate for the solar cell's degradation until EOL. The choice of orbit for a LEO remote sensing satellite is governed by the mission objectives and payload operational requirements [1].

Orbit parameters have direct impact on the electrical power subsystem (EPS) design process such as topology selection, solar array type, batteries type and sizing [2]. Also orbit parameters affect the load requirements and cost. The EPS design needs an intensive evaluation study for the influence of the orbit parameters variations to provide a robust design approach against any uncertainties in those parameters.

In order to analyze the approach of sizing the required power source; two main objectives should be followed [3]. The first objective understands the impacts of the orbit parameters variation and their interactions dynamics with the EPS design and operation. The second is evaluation of the importance of their effects.

The case studied has been considered in 3 different orbits at altitudes ranging between 400 km-1200 km, and inclination angle ranging between 40 deg-120 deg, to investigate the impacts of variation in the main orbit parameters e.g. altitude and inclination angle. Then the sizing of power sources has been evaluated via comparing the results of in-orbit simulations of EPS operation. In addition, some indirect impacts of the orbit parameters change are evaluated by analysis and calculation of the interaction between EPS and other subsystems. The results support and show how the sizing and operation of EPS are under the influence of orbit parameters variation via certain factors, such as orbit period, duration and the fraction of eclipse/sunlight phases, received solar irradiance by solar panels, and received thermal fluxes from the Sun.

Kim et al. [4] show statistical data analysis of EPS on-orbit anomalies and failures; it is found that the EPS of LEO satellites experienced much more severe anomaly and failure events than the EPS in GEO satellites; In addition, eclipses experienced by LEO more frequently and thus more charging/discharging cycles than satellites in GEO leading to more stresses on the LEO batteries, which means higher degradation and lower performance. Atomic oxygen (in altitudes below 800 km [5]) that can react with thin organic films, advanced composites، and metalized surfaces, resulting in degraded sensor performance, solar arrays are directly concerned by this type of corrosion and can suffer significant performance degradation.

Jang et al. [6], show that for solar array design, minimum voltage for operation of spacecraft and payload unit, string failure, cell degradation and radiation effects for marginal design is considered. Shadowing that solar cell will go into open circuit when not illuminated. Also, in a series connected string of solar cells, the shadowing of one cell results in the loss of entire string. GaAs triple junction cells which have 27.8% conversion efficiency at BOL (Beginning of Life) were used to meet the power requirement.

The most recent triple-junction solar cells [7], exhibit 30% conversion efficiency under air-mass zero (AM0) conditions. These cells produce more than twice the end-of-life (EOL) power than the best space silicon solar cells ever produced. Higher performance (i.e., 33% efficiency) and thinner multi- junction cells, such as the inverted metamorphic (IMM) solar cell technology presently under development will further enable higher power and lower mass (W/Kg) solar arrays, enabling satellites with greater power and capabilities.

Previous work by Shekoofa and Taherbaneh [3] used STK software in power source sizing in EPS (from 1/4/2003 to 1/4/2004). The operation scenarios are investigated for a life time of 3 years. The solar array contains 4 panels, which are assembled by triple-junction gallium arsenide (GaAs) solar cells with 26% efficiency. Only one nickel-cadmium battery had been used, however we believe that; it exhibits memory effect and low energy density [8-10], so in case of nickel-cadmium the total number of batteries used should have been three [11, 12] in the following manner: the first for operation, the second battery should be added for reconditioning purposes, and the third one in order to consider the possibilities of spacecraft redundant operation, which will add significant mass and cost. Also the number of cycles was (401098.9-329670.32) for altitude range (400-1200) km over a period of 3 years which we believe was wrongly calculated and it is more than 20 times the exact number; the wrong calculation affects three aspects of the battery as follows: the battery voltage at the end of charge, battery capacity at the beginning of life and battery capacity at the end of life. Moreover in calculating the β angle (the angle between the orbit plane and the sun solar vector), the used equation is wrong and the correct equation [13, 14] is

$$\sin(\beta) = \cos(\delta)\,\sin(\mathbf{i})\sin(\Omega - \Omega_S) + \sin(\boldsymbol{\delta})\cos[(\mathbf{i})] \tag{1}$$

Where δ: Sun declination, i: inclination angle, Ω: right ascension of the ascending node, and $\boldsymbol{\Omega_S}$: Sun right ascension.

In this paper four body mounted are selected as a structure for the case studied because the cells are mounted directly on the spacecraft body without using a honeycomb substrate to save mass and gimbals [9]. Also triple junction GaAs with 30% efficiency is selected because it is more efficient and it has been already proven to work on space satellite. Accordingly power at End –of-Life (EOL) will be increased for the same Photovoltaic (PV) area. Moreover, Lithium-ion proved to be the best choice for the case studied, for their high energy density, rapid charge characteristics, low mass, and lack of memory effects [8-10, 15]. Two Lithium - ion (Li-ion) batteries are selected to be more practical considering the possibilities of spacecraft redundant operation [11, 12]. The Satellite Tool Kit (STK) version 8.1 and the Matlab R2008a software have been employed for simulation and analysis.

2 The Case Studied

LEO satellite is assumed to be designed in order to detect, identify, and monitor fires throughout Cairo, and Suez Gulf area. An investigation of this satellite power sources operation in different orbits assuming the altitude ranges of 400, 450, 500,…., 1200 km is done. The calculated equations and the programming on (STK 8.1.1) will be used to get the results in this work by creating 90 scenarios, for 3 years (from 1/1/2011 to 1/1/2014) and 5 years (from 1/1/2011 to 1/1/2016), in order to evaluate and select the optimal orbit, altitude and inclination angle; considering both solar array components and energy storage devices for this satellite.

The satellite EPS operation occurs in 15 circular sun synchronous (SS) orbits (group A), 15 circular inclined (non sun synchronous) orbits with identical inclination in different altitudes (group B), and 15 other circular inclined orbits in the same altitude with different inclinations (group C) will be estimated.

3 Orbit parameters

Tables 1, 2, and 3, show the main orbit parameters. As shown in Group (A, B), orbit period (OP) increases with the increase of the orbit altitude, the total number of Cycles in 3, and 5 years (365 day per year $\times$ 24 hour per day $\times$ 60 minute $\times$ satellite $\frac{\text{life}}{\text{OP}}$) is decreasing with the increase of the orbit altitude. In group (C), (OP) is slightly increasing with the increase of the orbit inclination angle until it reaches 90° and then it slightly decreases with the increase of the inclination angle, the total number of Cycles in 3, and 5 years is decreasing with the increase of the orbit inclination angle until it reaches 90° and then it increases with the increase of the inclination angle.

Table 1 Main orbit parameters (Group A)

Group A				
Altitude [km]	Inc [deg]	OP [11, 12] [min]	#Cycles in 3years	#Cycles in 5years
400	97.03	92.393	17066.78	28444.6
450	97.21	93.415	16880.42	28134.00
500	97.40	94.44	16696.32	27827.20
550	97.59	95.469	16517.91	27529.90
600	97.79	96.502	16339.90	27233.20
650	97.99	97.539	16167.33	26945.60
700	98.19	98.58	15995.13	26658.60
750	98.39	99.624	15828.15	26380.20
800	98.60	100.671	15663.06	26105.10
850	98.82	101.723	15501.38	25835.60
900	99.03	102.778	15343.00	25571.70
950	99.25	103.836	15186.36	25310.60
1000	99.48	104.899	15032.89	25054.80
1100	99.90	107.034	14732.32	24553.90
1200	100.40	109.183	14442.21	24070.30

Table 2 **Main orbit parameters (Group B)**

Group B				
Altitude [km]	Inc [deg]	OP [11, 12] [min]	#Cycles in 3years	#Cycles in 5years
400	70	92.384	17067.89	28446.48
450		93.405	16881.32	28135.54

500		94.432	16697.73	27829.55
550		95.463	16517.39	27528.99
600		96.497	16340.4	27234.01
650		97.536	16166.34	26943.9
700		98.577	15995.62	26659.36
750		99.622	15827.83	26379.72
800		100.669	15663.21	26105.36
850		101.722	15501.07	25835.12
900		102.779	15341.66	25569.43
950		103.838	15185.19	25308.65
1000		104.907	15030.46	25050.76
1100		107.042	14730.67	24551.11
1200		109.193	14440.49	24067.48

Table 3 Main orbit parameters (Group C)

Group C				
Inc [deg]	Altitude [km]	OP [11, 12] [min]	#Cycles in 3years	#Cycles in 5years
40	400	98.40	16024.39	26707.32
45	450	98.43	16019.51	26699.18
50	500	98.46	16014.63	26691.04
55	550	98.49	16009.75	26682.91
60	600	98.52	16004.87	26674.79
65	650	98.55	16000.00	26666.67
70	700	98.57	15996.75	26661.26
75	750	98.59	15993.51	26655.85
80	800	98.60	15991.89	26653.14
85	850	98.61	15990.26	26650.44
90	900	98.61	15990.26	26650.44
95	950	98.60	15991.89	26653.14
100	1000	98.58	15995.13	26658.55
110	1100	98.55	16000.00	26666.67
120	1200	98.50	16008.12	26680.20

3.1 Eclipse Time

Eclipse time and its fraction (eclipse time to period ratio ($T_{\square}$/OP)) affect largely on EPS magnitude and operation. In Figs. 1- 2 the variation of the eclipse time and its fraction are presented for the three groups. Eclipse time and its fraction are both affected by the orbit altitude and the inclination angle.

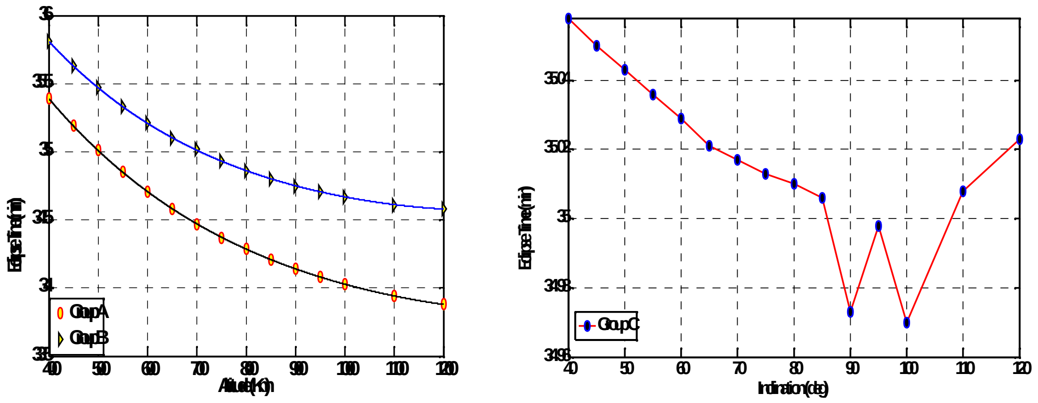

Fig. 1 Average Eclipse time with altitude group A, B (left), and with inclination group C (right)

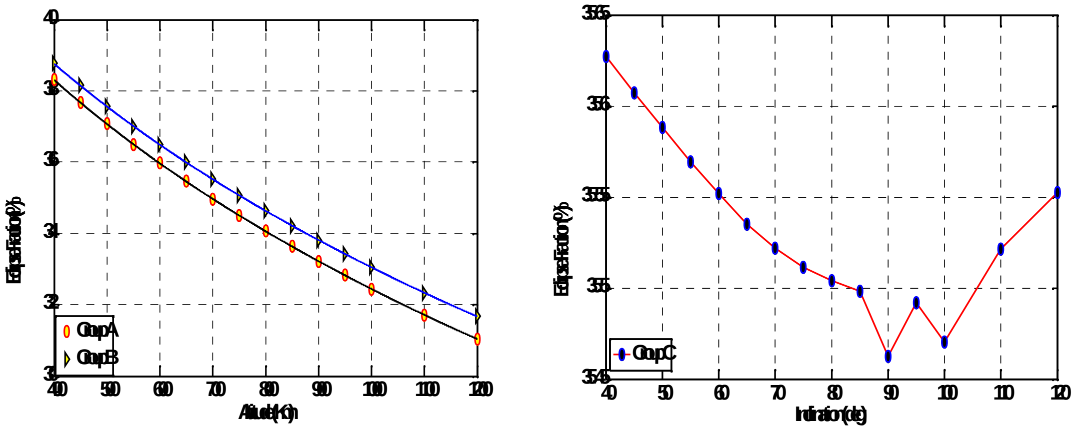

Fig. 2 Average Eclipse fraction with altitude group A, B (left), and with inclination group C (right)

In Fig.3, the trend of eclipse time variation is clearly descending with the altitude increment for a certain inclination, but the variation of eclipse time versus inclination angle is relatively complex, mostly in the area of inclination angle of sun synchronous (SS) orbits, this area should be investigated thoroughly from all aspects to determine whether it is possible to make use of it or not since the eclipse time is very small (13.09 min) compared to other eclipses times which means less PV, batteries requirements. It should be noted that Fig.3 represents the averaged value of eclipse time during a year.

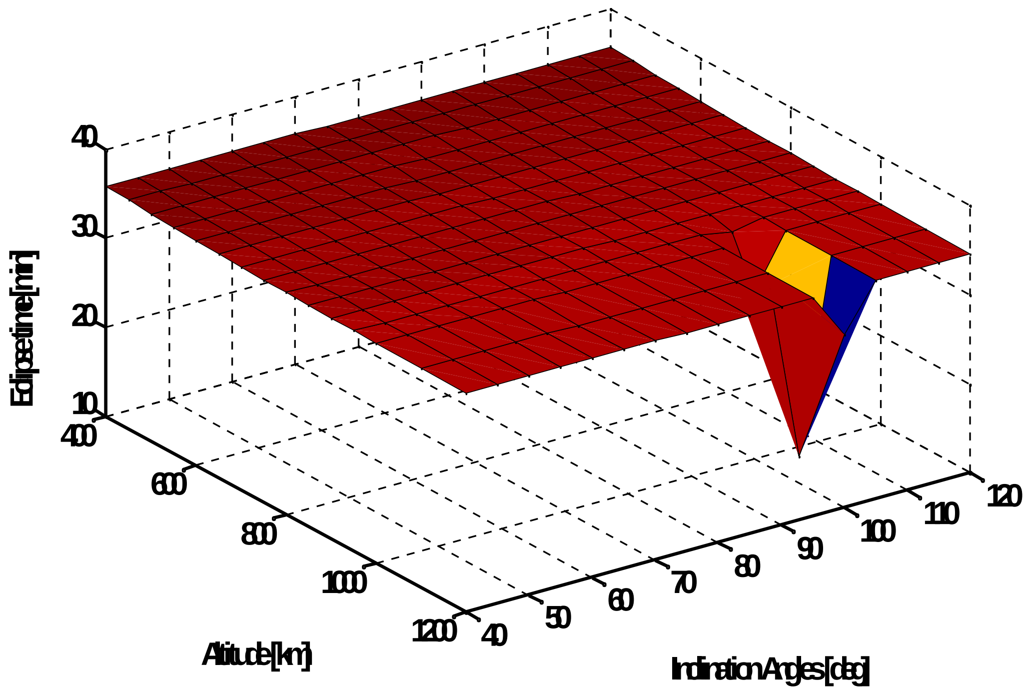

Fig.3 Variation of eclipse time (averaged for one year) versus altitude and inclination

3.2 Beta Angle

Beta angle defined as the angle between the Earth–Sun line and the orbit plane when the spacecraft is close to the sun, it varies between –90° and +90° as shown in

Fig. 4. Beta angle affects the EPS condition and operation. It also affects the thermal control subsystem design. Beta angle in (1) depends on many parameters; such as δ, i, [12, 14], and Ω as depicted in

Fig. 4. The beta angle as shown in Fig. 5. varies seasonally between (i ± 23.45) [9, 12, 16].

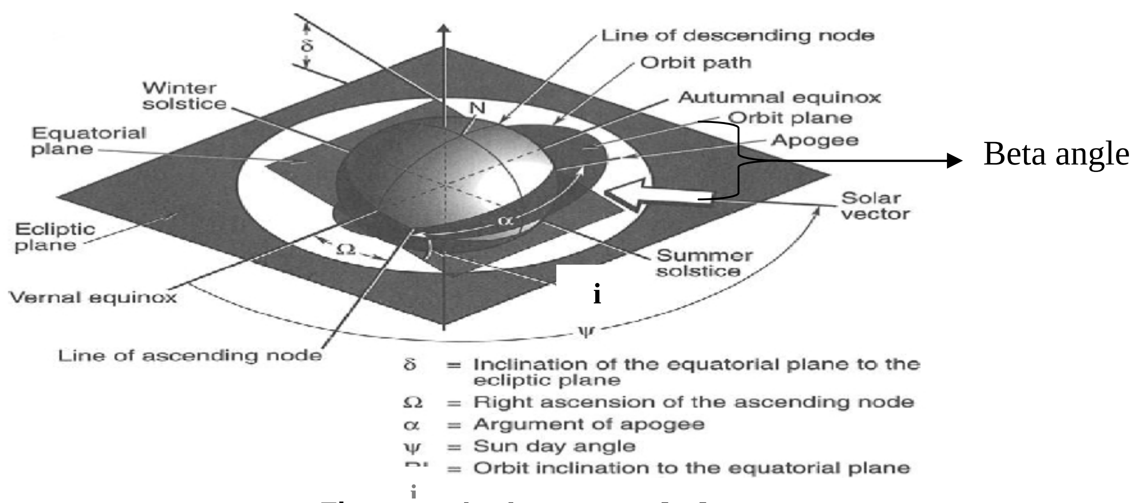

Fig. 4 Orbital parameters [14]

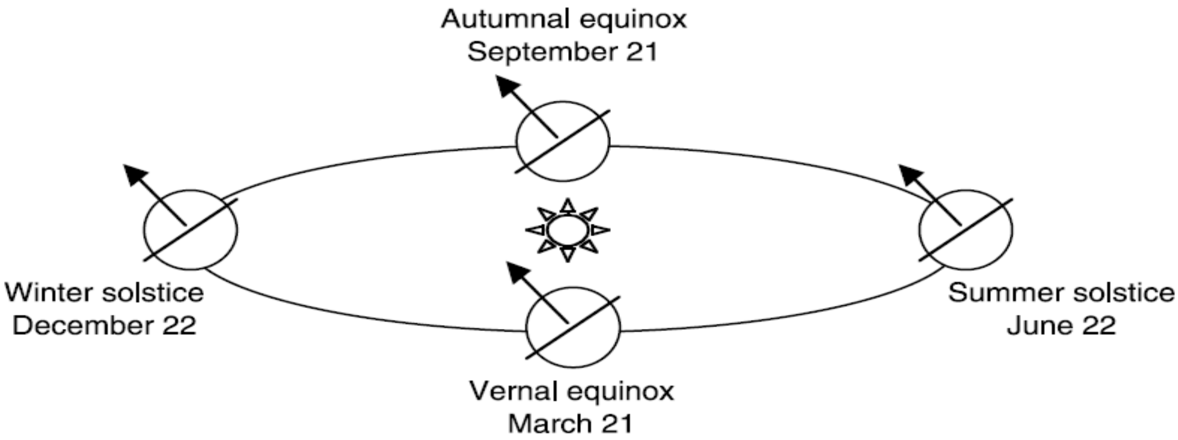

Fig. 5 Satellite in Earth's orbit with seasonal variations [9]

Although there is no clear relationship between beta and altitude, beta angle varies significantly with inclination angle, so it must be considered precisely in the evaluation of the EPS design and operation. On the other hand eclipse time and its fraction are directly in relation with beta angle as stated in (2), (3), and (4).

$$T_E = \frac{1}{2} + \frac{1}{\pi}\sin^{-1}\left[\frac{\sqrt{1-\left(R_{Earth}/R_{orbit}\right)^2}}{\cos(\beta)}\right], \text{ if } |\beta| < \beta^* \quad [9] \qquad (2)$$

$$f_E = \frac{1}{\pi}\cos^{-1}\left[\frac{\sqrt{(h^2 + 2R_{Earth}H)}}{(R_{Earth} + H)\cos(\beta)}\right], \quad \text{if} \quad |\beta| < \beta^* \quad [13, 14] \qquad (3)$$

Also,

$$f_E = \frac{1}{\pi}\cos^{-1}\left[\frac{\sqrt{1-\left(R_{Earth}/R_{orbit}\right)^2}}{\cos(\beta)}\right], \quad \text{if} \quad |\beta| < \beta^* \quad [9] \qquad (4)$$

Where; $\beta^* = \sin^{-1}(R_{Earth} / R_{orbit})$ [9, 13, 14] (5)

$R_{orbit} = (R_{Earth}) + H$ [9, 12-14] (6)

R_{Earth} is the Earth radius (6378 km), H is orbit altitude, T_E is the eclipse time, f_E is the eclipse fraction and Beta-star β^* is the beta angle at which eclipses begin.

The minimum and maximum values of beta angle are calculated by using the STK software and listed in Table **4.**

Table 4 Minimum and Maximum values for beta angle

Altitude [km]	Group A		Group B		Group C		
	β_{min} [deg]	β_{max} [deg]	β_{min} [deg]	β_{max} [deg]	Inc [deg	β_{min} [deg]	β_{max} [deg]
400	-25.03	-15.13	-88.20	89.03	40	-63.30	63.43
450	-25.09	-15.06	-87.37	88.40	45	-67.52	68.07
500	-25.15	-14.99	-88.81	88.86	50	-73.33	73.16
550	-25.20	-14.91	-88.19	89.84	55	-75.14	75.77
600	-25.26	-14.84	-86.63	89.97	60	-83.37	82.94
650	-25.32	-14.76	-88.30	89.24	65	-85.07	88.12
700	-25.38	-14.68	-89.02	88.37	70	-89.02	88.37
750	-25.45	-14.60	-88.54	89.91	75	-85.82	88.27
800	-25.51	-14.52	-89.77	87.61	80	-85.35	85.47
850	-25.58	-14.43	-88.65	88.48	85	-87.94	85.55
900	-25.64	-14.35	-88.38	89.77	90	-66.56	66.56
950	-25.71	-14.26	-89.18	89.07	95	-85.72	68.84
1000	-25.78	-14.17	-85.87	87.59	100	-84.23	88.02
1100	-25.91	-13.98	-85.43	85.82	110	-89.47	88.02
1200	-26.06	-13.79	-89.66	89.28	120	-81.42	83.02

The most significant effect of the β angle on the power system design comes from the eclipse duration as given by (2). In most satellites with the array always pointed to the sun (sun tracking), the β angle has an insignificant effect on the generation of electric power. However, the thermal control system is impacted by the β angle. Low β may require additional heaters, while high β may require additional cooling [9]. Also, the β angle impacts the solar array temperature, which in turn has a small secondary effect on the power generation. As β angle increases, the eclipse duration decreases, which consequently requires a smaller battery and less charging power during sunlight.

3.3 Solar Irradiance

Sun activities and solar intensity are time variable and they fluctuate throughout the year. In this paper it is common to consider a constant value equal to 1353 W/m^2 for received solar irradiance above Earth atmosphere.

4 The Studied Satellite and Its EPS

The proposed satellite has a cubic structure with 4 body mounted solar panels in its lateral sides. The required average and maximum power generation by EPS is assumed to be constant and equal 100 watts for both light phases and eclipses.

5. Power Sources Sizing and Operation

5.1 Solar Array Specification

The solar array contains four panels; the type of solar cell is (triple-junction gallium arsenide (GaAs) solar cells with 30% efficiency. The degradation rate is approximately 2.66% per year [3] and the inherent degradation equal to I_d =0.77. The solar panels' incidence angle (θ) is assumed to be 45 degrees in the worst case, and each panel alone should be able to generate the required average power. The array performance at BOL and EOL are calculated from (10), and (11) and listed in

P_o is the output power with sun normal to the surface of triple junction Ga As, AP_{BOL} is the

array performance at BOL, AP_{EOL} is the array performance at EOL, A_{sa} is the solar array required

area, and M_{sa} is the solar array required mass [11, 12].

Table 5, for 3 and 5 years satellite life time.

The amount of average power that must be produced by the solar arrays, "Psa" is

$$P_{sa}(3,5\ years) = \left(\left(P_e T_e (3,5\ years)/\eta_e\right) + \left(P_d T_d (3,5\ years)/\eta_d\right)\right) / T_d (3,5\ years) \qquad (7)$$

Where $P_e = 100$ watt and P_d = 100 watt; where P_e is the averaged required power in eclipse phase, and P_d is the averaged required power in daylight phase. η_e , η_d are the efficiencies of the paths from the arrays to the batteries and to the loads, while T_e, T_d are time duration during eclipse and daylight in minute. In this paper η_e and η_d are considered to be 0.65 and 0.85 respectively,

$$P_{BOL}(3,5\ years)\ (Watt) = P_{sa}(3,5\ years) \times (1 + degradation\ rate\ per\ year)^{satellite\ life} \qquad (8)$$

Where P_{BOL} is the power at the beginning of life to provide the EPS with an additional power to compensate for the solar cell's degradation until EOL, and P_{sa} is assumed to be the amount of average power that must be produced by the solar arrays at EOL. Considering,

$$P_O\ (w/m^2) = (\text{Efficiencies for the cells}) \times (\text{solar flux constant; } w/m^2) \qquad (9)$$

$$AP_{BOL}\left(\frac{w}{m^2}\right) = P_O \times I_d \times \cos(\theta) \quad (10)$$

$$AP_{EOL}(3,5\ )years\left(\frac{w}{m^2}\right) = AP_{BOL}(3,5\ )years \times (1 - \text{degradation rate per year})^{\text{satellite life}} \quad (11)$$

$$A_{sa}(3,5\ )years\ (m^2) = \frac{P_{BOL}(3,5\ )years}{AP_{BOL}(3,5\ )years}$$

(12)

$$M\ (3,5\ )years\ (Kg) = \frac{P_{BOL}(3,5\ )years}{25}$$

(13)

P_O is the output power with sun normal to the surface of triple junction Ga As, AP_{BOL} is the array performance at BOL, AP_{EOL} is the array performance at EOL, A_{sa} is the solar array required area, and M is the solar array required mass [11, 12].

Table 5 The array performance at BOL, and EOL

Group A,B,C		
AP_{BOL} (W/m^2)	AP_{EOL} (W/m^2)	
3 ,5 years	3 years	5 years
221	203.83	193.13

From the previous table it is clear that EPS has been provided by an additional power until EOL to compensate for the solar cell's degradation at EOL in satellite.

According to (7), (8), (12), and (13), the amount of average power that must be produced by the solar arrays at (EOL) $P_{(EOL)}$, (BOL) "$P_{(BOL)}$", solar array required area "$A_{(sa)}$" , and their overall mass "M" have been calculated, and listed in Figs 6-9.

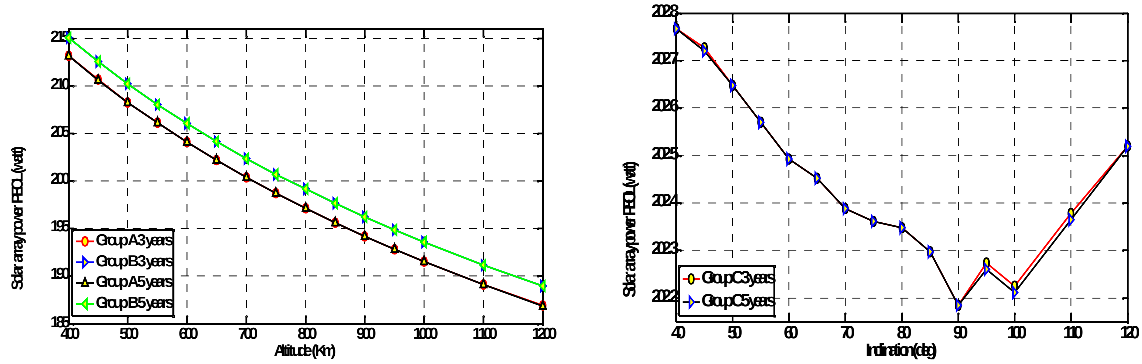

Fig. 6 Solar array power at EOL (P_{EOL}) with altitude group A, B (left), and with inclination group C (right)

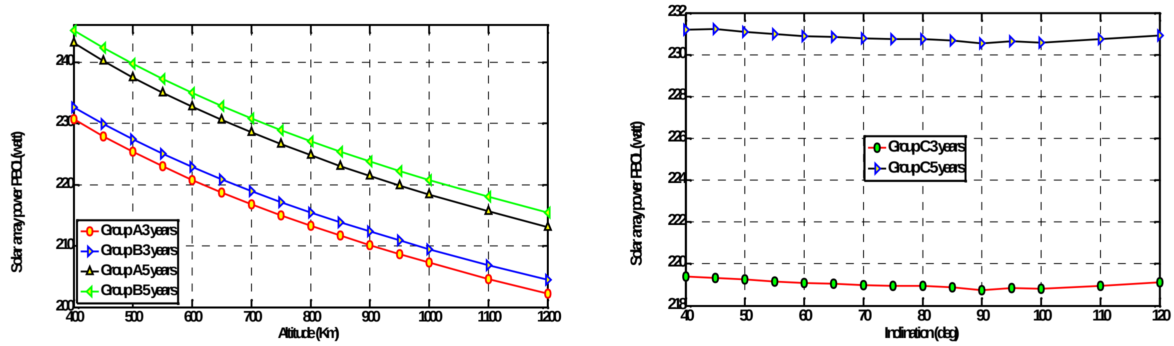

Fig. 7 Solar array power at BOL (P_{BOL}) with altitude group A, B (left), and with inclination group C (right)

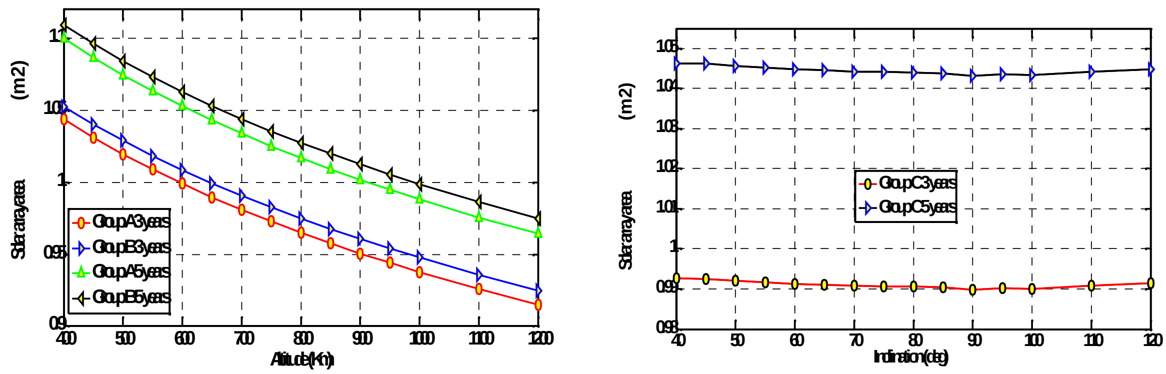

Fig. 8 Solar array area with altitude group A, B (left), and with inclination group C (right)

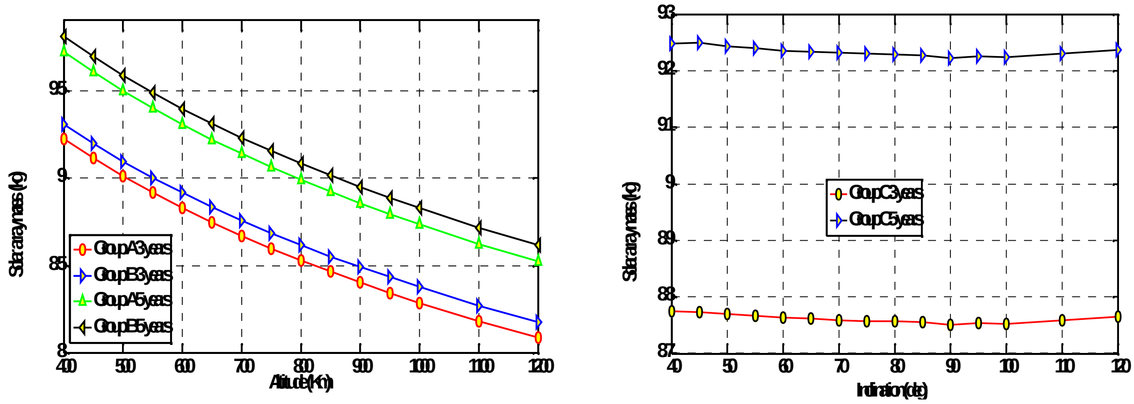

Fig. 9 Solar array mass with altitude group A, B (left), and with inclination group C (right)

Solar array (power at BOL and EOL, area, and mass) are both affected by the orbit altitude and the inclination angle, as shown in Figs 6-9.

5.2 Solar Array operation during the Mission

Solar array duty cycle ($\frac{\textbf{time of producing energy by solar panel}}{\textbf{time which satellite is under Sun projection}}$) should be calculated to determine the total generated energy by the solar arrays in three and five years for each orbit. Then the dependency of the solar array duty cycle and power generation to orbit parameters can be evaluated.

In addition to orbit period and eclipse fraction, there are two more important factors to determine the amount of received solar irradiance by solar array. These factors are the angle of solar incident to the panel, and panel operation coefficient. As mentioned before, the first factor is considered to be 45 deg in the worst case, but the second factor can be formulated in (16) for a given cycle.

The eclipse angle γ_e and the half daylight angle γ_{hd} (e.g. if the orbit has 25% of eclipse then $\gamma_e = \frac{\pi}{2}$ and $\gamma_{hd} = \frac{3}{4}\pi$) as shown in Fig. 10, and can be computed as [17].

$$\gamma_e = \begin{cases} 2\arctan\sqrt{\dfrac{R_{Earth}{}^2 - R_{orbit}{}^2 \cos^2\beta}{R_{orbit}{}^2 - R_{Earth}{}^2}} & \text{if } |\cos\beta| < \dfrac{R_{Earth}}{R_{orbit}} \\ 0 & \text{if } |\cos\beta| \geq \dfrac{R_{Earth}}{R_{orbit}} \end{cases} \tag{14}$$

$$\gamma_{hd} = \pi - \frac{\gamma_e}{2} \tag{15}$$

$$C_a = \frac{1}{2\pi}\int_{-\gamma_{hd}}^{\gamma_{hd}} (p_1 \sin\beta \sin\varphi + p_2 \cos\beta +, p_3 \sin\beta \cos\varphi) d\varphi \tag{16}$$

Where ($p_{□}, p_2, p_3$) is the panel normal vectors, □ is the satellite rotational angle along the orbit.

If it is required to determine the panel operation coefficient during a long period (U_{C_a}) it is possible to calculate the integral of C_a over the range of annual beta angle variation from Table 4, and (17).

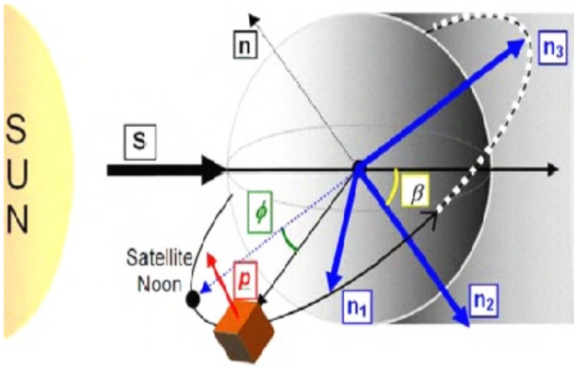

Fig. 10 Illustration of different vectors and angles of satellite attitude [3]

$$U_{C_a} = \frac{1}{\beta_{max-}\beta_{min}} \int_{\beta_{min}}^{\beta_{max}} C_a d\beta \tag{17}$$

To make this integration first integrate

W= $\int C_a$ □β

Using (14)-(16) we can find that

W=

+C

(18)

Where C is constant

To do this integration we have considered that □ is independent of □ in (16). Then we have,

$$U_{C_{max □}}(3.5\ )year = p_2 \times \Bigg[\pi \sin \beta_{max □} - \sin\beta_{max} \times \tan^{-1} \sqrt{-1 + \frac{R_{orbit}{}^2 \sin^2 \beta_{max}}{R_{orbit}{}^2 - R_{Earth}{}^2}} + \sqrt{1 - \frac{R_{orbit}{}^2}{R_{Earth}{}^2}} \times \cosh^{-1} □ \Bigg(\ldots$$

$$U_{C_{min □}}(3.5\ )year = p_2 \times \Bigg[\pi \sin \beta_{min □} - \sin\beta_{min} \times \tan^{-1} \sqrt{-1 + \frac{R_{orbit}{}^2 \sin^2 \beta_{min}}{R_{orbit}{}^2 - R_{Earth}{}^2}} + \sqrt{1 - \frac{R_{orbit}{}^2}{R_{Earth}{}^2}} \times \cosh^{-1} □ \Bigg(\ldots$$

$$U_{C_2}(3,5\ )\text{years} = \frac{U_{C_{max(3,5\ years)}} - U_{C_{min(3,5\ years)}}}{\beta_{max(3,5\ years)} - \beta_{min(3,5\ years)}} \qquad (21)$$

The calculated values of U_{C_2} in satellite noon ($\emptyset$ =0) for panel normal vector p = (0, 1, 0), as depicted in Fig. 11**,** for each orbit. This parameter is found close to 93.83 % for sun synchronous orbits (group A). The average operation time during the mission is calculated from (22) and then presented in Fig. 12. Based on these data, the average total generated energy by the solar array during the mission is calculated from (23) and then presented in Fig. 13. In these calculations, panel temperature is assumed to be 25°C because of an assumed spin rate (refers to the speed it spins on an axis while in flight, measured in rpm) higher than 5 rpm to equalizes the temperature of satellite different sides. It is also assumed that there is one and just one panel which supplies the satellite at any time.

$$\text{Avg opt}(3,5\ )\text{years (hr)} = \frac{U_{C_2}(3,5\ )\text{years} \times T_d(\text{hr})(3,5\ \text{years}) \times \text{satellite life} \times 365\ \text{day} \times 24\ \text{hr}}{\text{OP}(\text{hr})(3,5\ \text{years})}$$

(22)

$$\text{TGE}(3,5\ )\text{years}\ (\text{Kwhr}) = \text{Avg opt}(\text{hr})(3,5\ \text{years}) \times 100\ \text{watt} \qquad (23)$$

Where (Avg opt) is the average operation time during 3, 5 years, and (TGE) is the total generated energy during 3, 5 years.

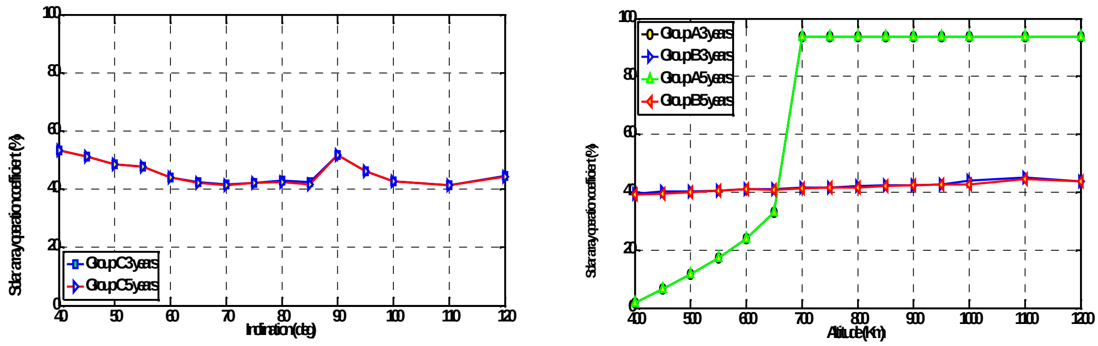

Fig. 11 Solar array operation coefficient with altitude group A, B (left), and with inclination group C (right)

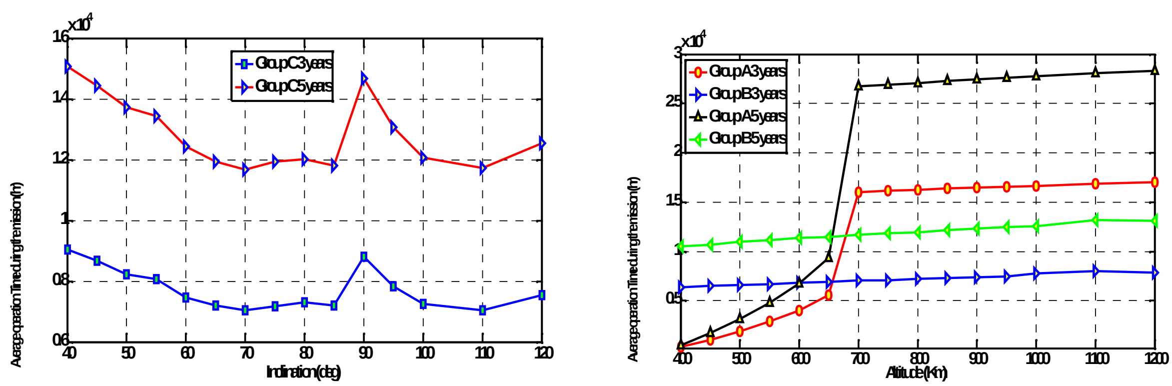

Fig. 12 Average operation time during the mission with altitude group A, B (left), and with inclination group C (right)

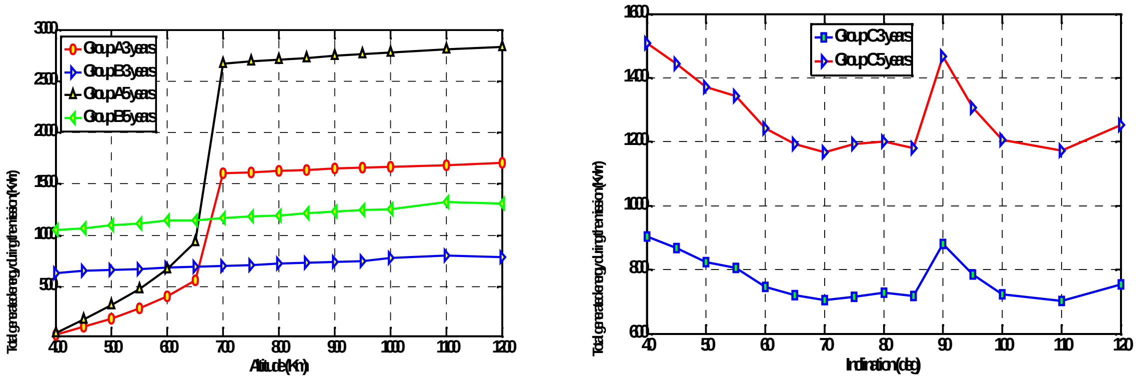

Fig. 13 Total generated energy by solar array during the mission with altitude group A, B (left), and with inclination group C (right)

As shown in Figs. 11-13 group (A) enables maximum value for both of the solar array operation coefficient (U_{C}) which is calculated close to (93.83 %), average operation time during the mission (Avg opt) and total generated energy (TGE) by solar array during the mission at altitudes ranging between (700-1200) km.

5.3 Battery Specifications and Operation

During the mission, Lithium - ion is considered in this paper because it matches with space qualification and has adequate characteristics as listed in

Table 6.

Table 6 Characteristics of [Lithium – ion] Battery

Type of Battery	Depth of discharge (DOD) [%]	Energy Density [Whr/lit]	specific energy density [Whr/kg]
Li-ion	30	150	120

Nominal voltage of 28 volts based on these assumptions and the defined orbit parameters, which should be 28 volts at BOL, normally. This parameter is calculated by assuming a 0.0182 volt decrement per each 1000 cycles for space qualified Battery [3, 18].

The total number of cycles per year is shown in Fig. 14, and is given by (24).

Total number of cycles (3, 5) (# cycles) =

$$\frac{365\,\text{day} \times 24\ \text{hour} \times 60\,\text{min} \times (\text{satellite life})}{\text{OP}(\text{min})\,(3,5\ \text{years})} \qquad (24)$$

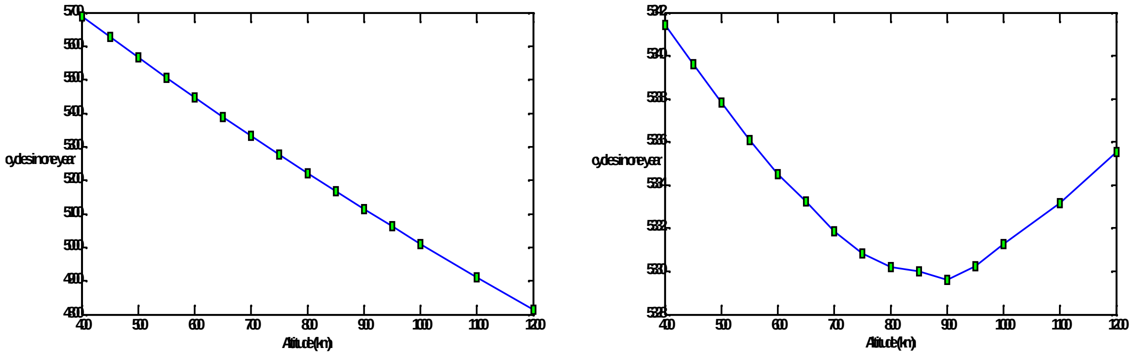

Fig. 14 Total number of cycle in one year with Altitude group A, B (left), and with inclination group C (right)

As shown in Fig. 14**,** it is obvious that the total number of cycle in Group (A, B) is almost identical, moreover any increment in orbit altitude increases the orbit period and decreases the number of cycles in orbit. This leads to reduce the charge/discharge cycles. So we can use the battery with more DOD or use the battery for a longer life with the same DOD; which means lower degradation and higher performance. On the other hand, in Group (C) when the satellite inclination angle increases in a certain altitude, OP is slightly decreasing until it reaches 90° and then it slightly increases with the increase of the inclination angle.

As the worst case, a peak power consumption of 100 watts is assumed. Then the Battery voltage at end of charge (end of eclipse phase) VED, and the required battery capacity C_r for both of (BOL, EOL) are calculated from (25)-(28), and then presented in Figs. 15- 17.

$$C_r\ (AH) = \frac{Pe \times Te(3,5\ years)}{(DOD) \times N \times n \times 60 \times V_{bus}}$$ [3, 11, 12]

(25)

Where P_e is the required power in eclipse phase, T_e is eclipse duration, N is the batteries number, N is taken 2 because the spacecraft needs redundant operation if one unit failed and n is the efficiency of discharge path from battery to load =90% for our study, and V_{bus} is battery bus voltage =28V.

$$C_{rBOL}\ (3,5)\ years\ (AH) = \frac{P_e \times T_e(3,5\ years) \times (1 + degradation\ rate\ per\ year)^{satellite\ life} \times (1 + 0.0182)^{\frac{(\#\ cycles(3,5)years)}{1000}}}{(DOD \times N \times n \times V_{bus} \times 60)}$$

(26)

$$VED\ ^{(3,5)years}\ (Volt) = \frac{(V_{bus})}{(1 + 0.0182)^{\frac{(\#\ cycles(3,5)years)}{1000}}}$$

(27)

$$C_{rEOL}(3,5\ \text{years})(AH) = \frac{P_e \times T_e(3,5\ \text{years})}{DOD\% \times N \times n \times 60 \times VED(3,5)\ \text{years}}$$

(28)

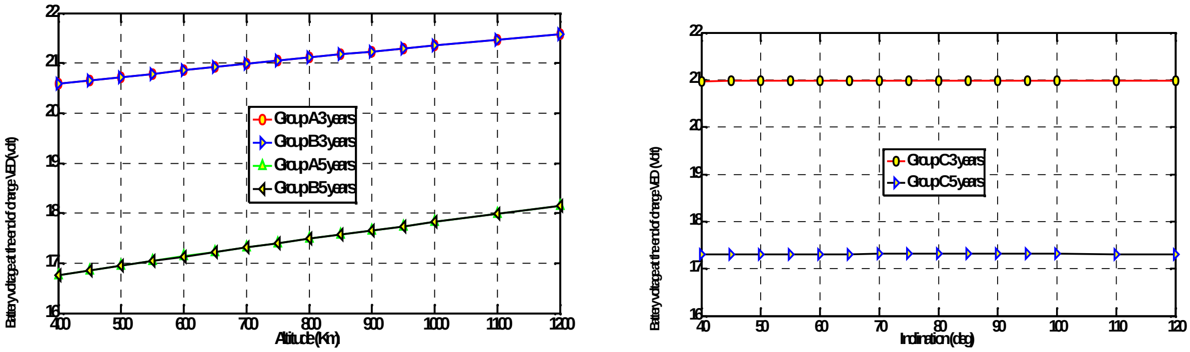

Fig. 15 Battery voltage at the end of charge (VED) with altitude group A, B (left), and with inclination group C (right)

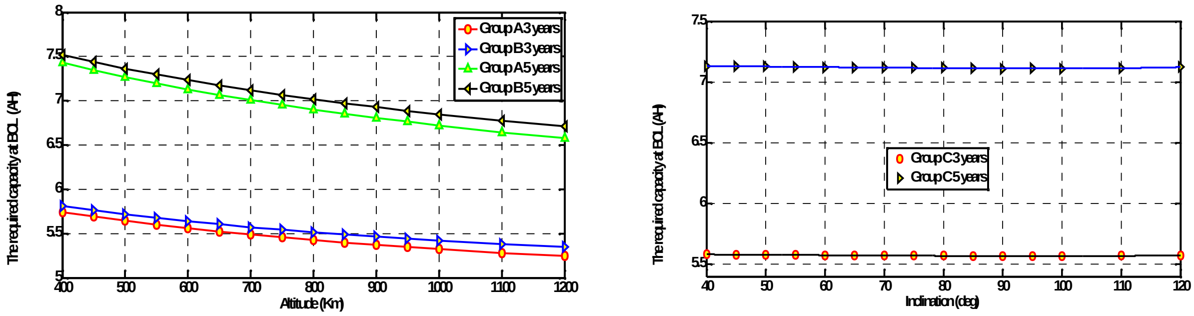

Fig. 16 The required capacity at BOL with altitude group A, B (left), and with inclination group C (right)

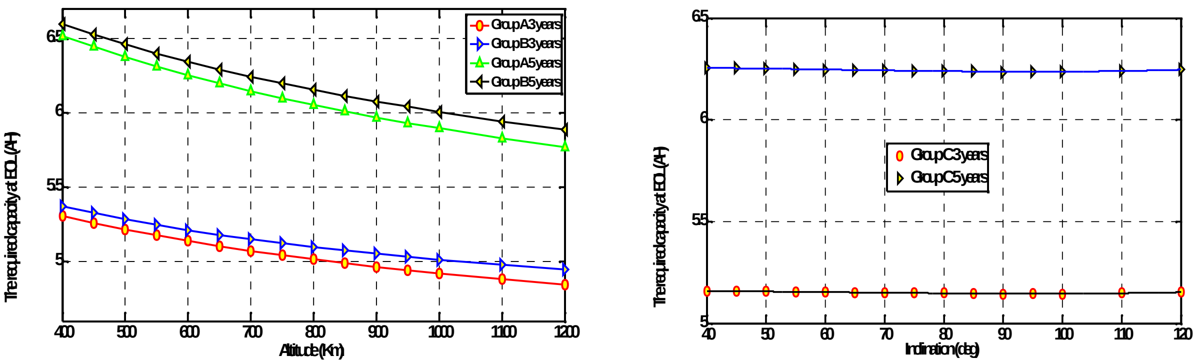

Fig. 17 The required capacity at EOL with altitude group A, B (left), and with inclination group C (right)

According to these data, battery sizing can be calculated from (29), and (30), and then presented in Figs. 18-19.

$$\text{Battery mass}(3,5\ \text{years})\ \text{kg} = \frac{C_{rBOL}(3,5\ \text{years}) \times V_{bus}}{e_b} \qquad (29)$$

Where e_b is the specific energy density =120 Whr/kg, and e_d is the Energy Density =150 Whr/lit

$$\text{Battery volume}(3,5\ )\text{years litre} = \frac{C_{rBOL}\ (3,5\ )\text{years} \times V_{bus}}{e_d} \qquad (30)$$

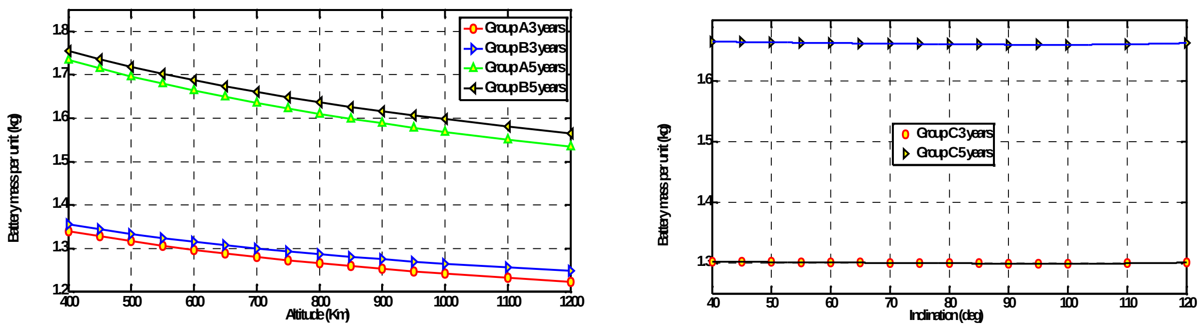

Fig. 18 Battery mass per unit with altitude group A, B (left), and with inclination group C (right)

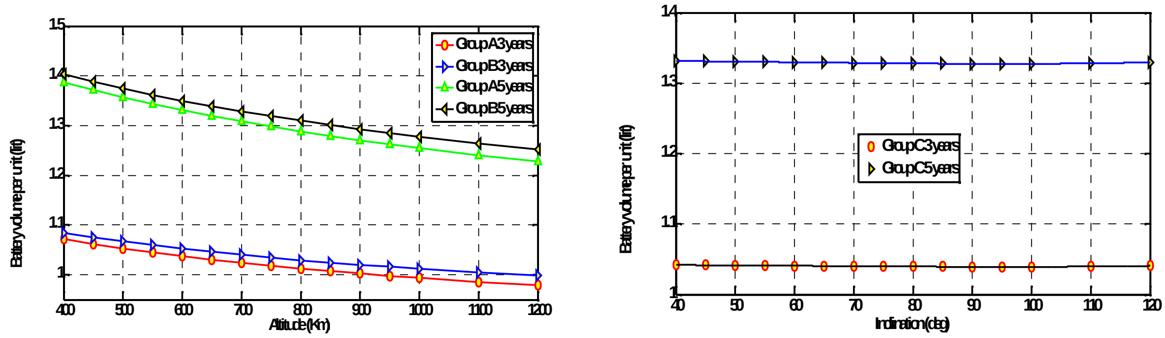

Fig. 19 Battery volume per unit with altitude group A, B (left), and with inclination group C (right)

6. EPS Interaction with other Subsystems

In addition to the operation of power generation or energy storage elements, the other satellite subsystems (EPS loads) operation and performance are affected significantly by the change in orbit parameters.

When the amount, rate and profile of energy consumption vary, the operation and performance of EPS and its elements are affected consequently. So it is necessary to evaluate the influences of orbit parameters on the operation of the subsystems which have interactions with EPS. These interactions could be direct via the electrical connections or they could be indirect through the thermal interfaces.

Two samples of these interactions are considered briefly in this study, and the relevant impacts are calculated for certain subsystems which are in different levels of importance from the viewpoint of interaction with EPS.

1) In telemetry/telecommand subsystem and/or in satellite payloads, transmitters are commonly the major loads of EPS. Satellite access time and pass number determine the amount and regime of power consumption by the transmitters. So their dependencies on orbit parameters and their effects on power sources should be evaluated and calculated for total mission life.

In this paper, a ground station (GS) is chosen in Suez Golf area, access number, average access time per revolution are calculated by using (STK 8.1) for this GS, and then according to these data, the total access time after 3, 5 years can be calculated from (31) and then all data are presented in Figs. 20-22**.**

Total access time (3, 5 years) hr =

$$\frac{\textbf{Access number \#}(3,5\ \textbf{years}) \times \textbf{Average access time per revolution}\ (\text{sec})}{((60 \times 60))(3,5\ \textbf{years})}$$

(31)

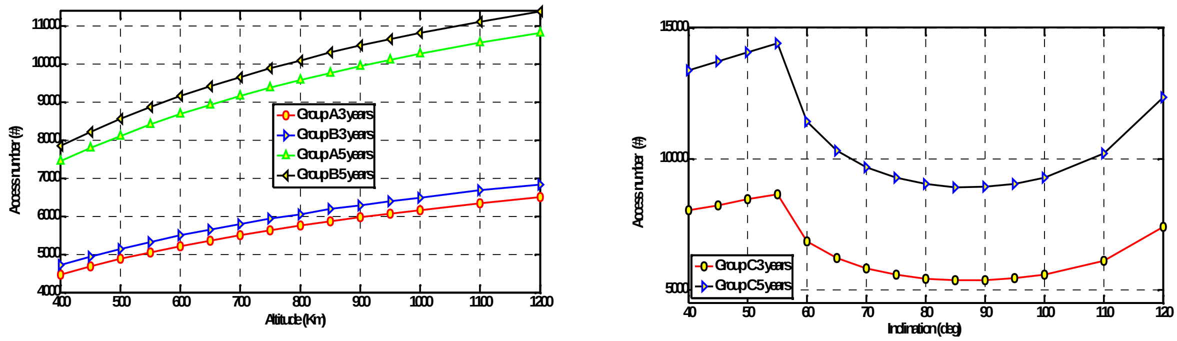

Fig. 19 Access number with altitude group A, B (left), and with inclination group C (right)

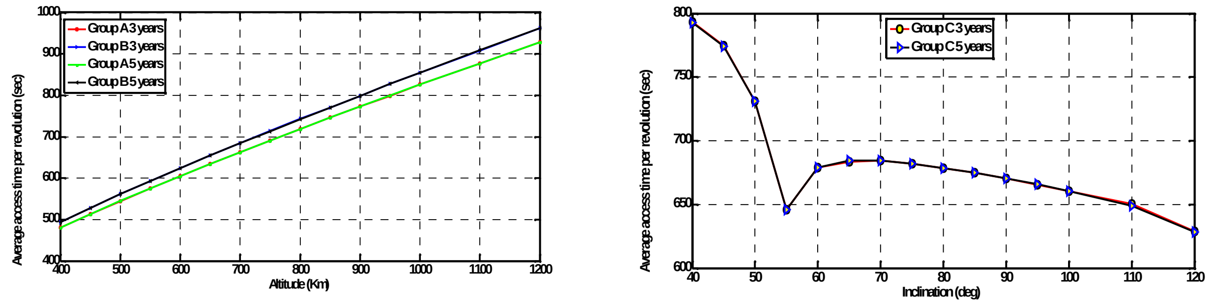

Fig. 20 Average access time per revolution with altitude group A, B (left), and with inclination group C (right)

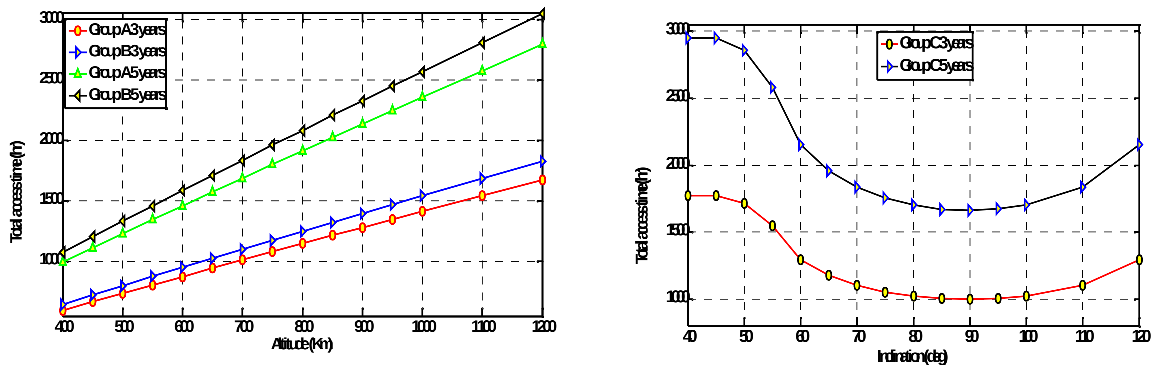

Fig. 21 Total access time with altitude group A, B (left), and with inclination group C (right)

It is assumed that the maximum power consumption (100 watts) happens in each cycle, and if the access times is considered to be equal to the average access time, then the energy consumption, and average DOD% are calculated from (32)-(33) and then presented in Figs 23-24**,** these data can be extracted to show the impacts of orbit parameters on energy consumption and battery operation.

$$\text{Kwhr Transmitters (3, 5 years)} = \frac{\mathbf{P_e\ watt \times Total\ access\ time\ hr\ (3,5\ years)}}{\mathbf{1000}} \tag{32}$$

Where (Kwhr Transmitters) is the Energy consumption by the transmitters

$$\text{Average DOD Transmitters (3, 5 years) \%} = \frac{\mathbf{Kwhr\ Transmitters(3,5\ years) \times 1000 \times 100 \times 60}}{\mathbf{(P_e\ watt \times OP(3,5\ years)min \times Access\ number\ \#(3,5\ years))}} \tag{33}$$

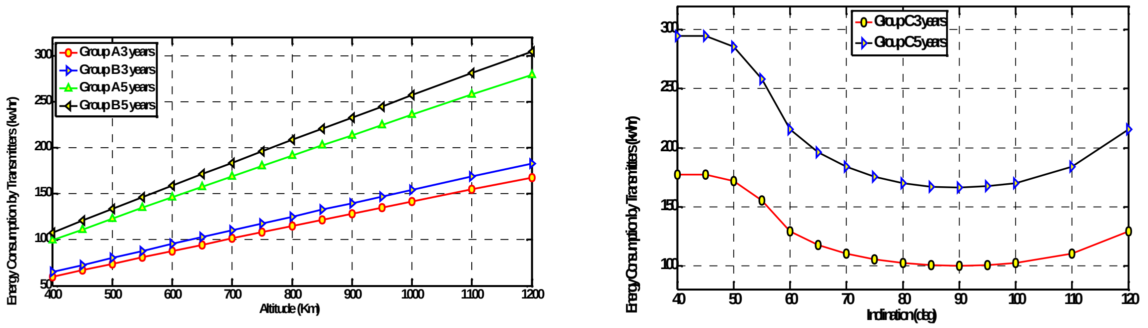

Fig. 22 Energy Consumption by Transmitters with altitude group A, B (left), and with inclination group C (right)

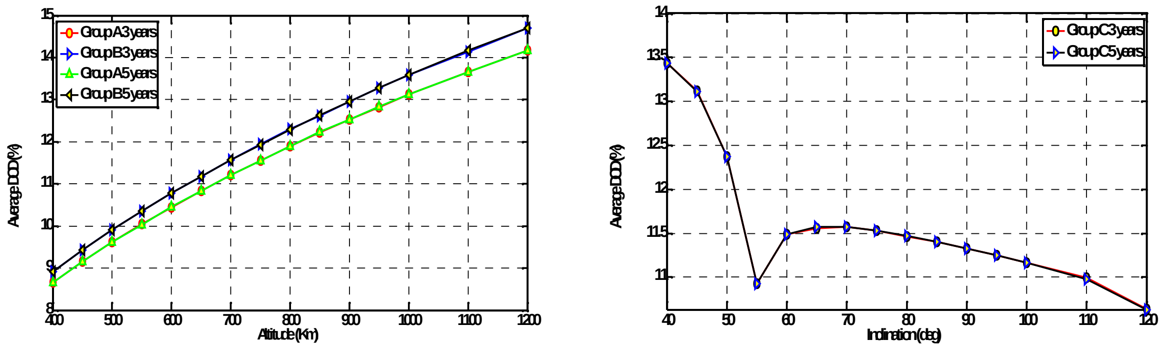

Fig. 23 Average DOD with altitude group A, B (left), and with inclination group C (right**)**

2) As mentioned before, any variation of altitude or beta angle, leads to changes in eclipse parameters. Eclipse duration and fraction have the most important impacts on the satellite thermal condition. On the other hand, thermal behavior and operation of many parts/elements/subsystems strongly depend on thermal conditions.

For example, using heaters for battery is not avoidable sometimes, especially when the satellite goes through the earth eclipse and faces a cold condition. When the inside temperature becomes lower than the permitted level, the battery will be affected significantly, because it is one of the most sensitive elements to the thermal conditions. Though; heaters are needed, they must be supplied by battery, so there is a mutual interaction between them. The evaluation of the impacts of using a 10-watt heater, for the (Lithium – ion) battery of our satellite can be calculated from (34), and (35), and then presented in Figs 25-26.

Kwh (Heaters) (3, 5 years) =

$$\frac{\left(10\ watt \times T_e\ (3,5\ \mathbf{year})\ \min \times (\#\ \mathbf{cycles})(3,5\ \mathbf{years})\right)}{(60 \times 1000)} \qquad (34)$$

Where Kwh (Heaters) is the Energy consumption by the Heaters

Average DOD %(Heaters) (3, 5 years) =

$$\frac{\mathbf{Kwh}(\mathbf{Heaters})(3,5\ \mathbf{years}) \times 60 \times 1000 \times 100}{\mathbf{P_e}\, watt \times \mathbf{OP}(3,5\ \mathbf{years}) \min \times (\#\ \mathbf{cycles})(3,5\ \mathbf{years})} \qquad (35)$$

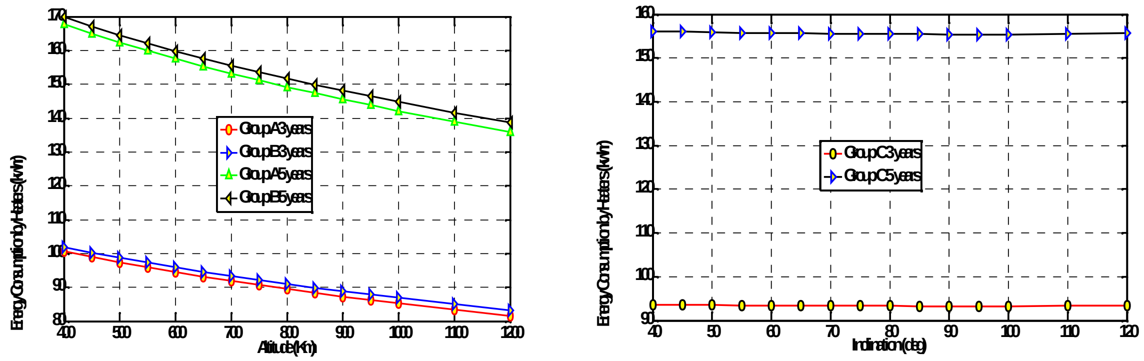

Fig.24 Energy Consumption by the Heaters with altitude group A, B (left), and with inclination group C (right)

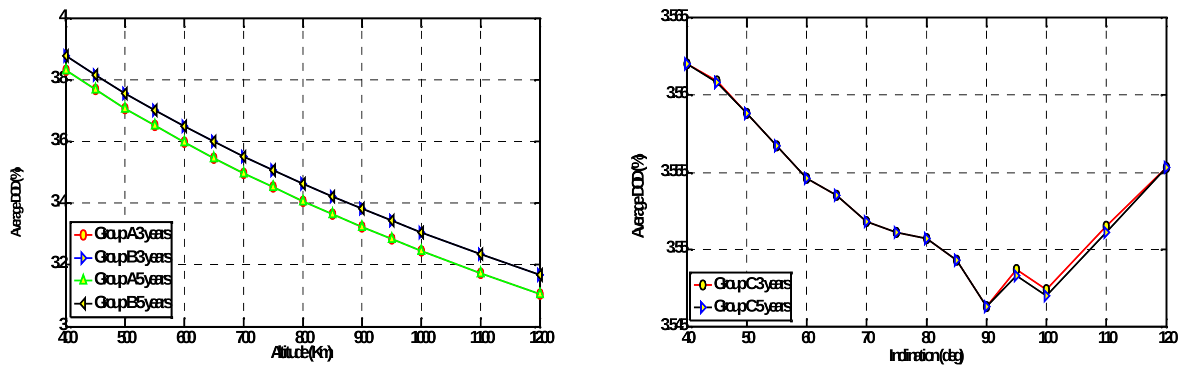

Fig. 25 Average DOD with altitude group A, B (left), and with inclination group C (right)

By adding energy consumption by transmitters (32) and energy consumption by heaters (34) in order to calculate the total energy consumption (TEGC) then comparing the result with the total energy generated by solar array (TGE) (23), and presenting the results in Fig. 26.

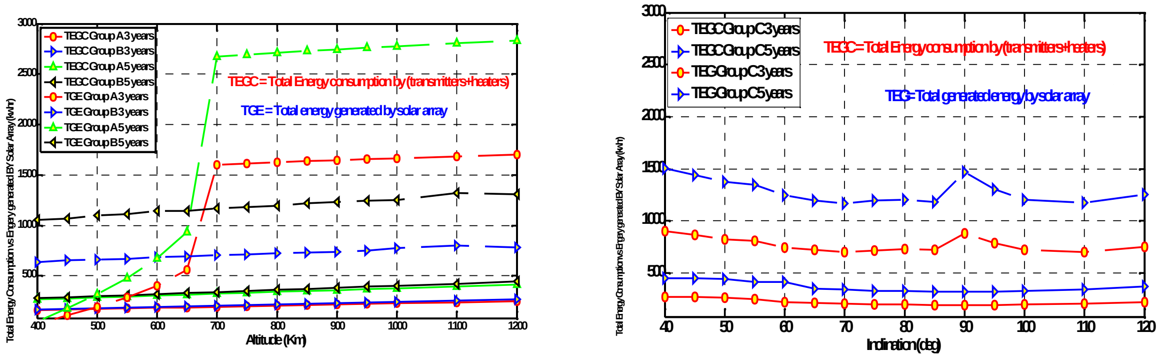

Fig. 26 Total Energy Consumption by (transmitters & heaters) versus Total Energy generated by Solar Array with altitude group A, B (left), and with inclination group C (right).

As shown in Fig. 26, group (A) enables maximum value for the total generated energy (TGE) by solar array during the mission at altitudes ranging between (700-1200) km. Group (B, C) TGE is greater than TEGC for all altitudes and inclination angles of the selected range.

7. CONCLUSIONS

The impacts of orbit parameters change are investigated for EPS, solar array and battery sizing, their operation and performance. This is done by comparing the EPS specification for different altitude in a typical LEO proposed satellite. Indirect effects on the EPS operation due to its interaction with the other subsystems have been analyzed and described. Quantifying these impacts help the EPS designer to define more precise margins, to achieve a better EPS design and performance.

From our research results, it is clear that eclipse time and its fraction that slightly change the battery specifications.

The variation of eclipse time versus inclination angle is relatively complex, mostly in the area of inclination angle of sun synchronous (SS) orbits. In the near future, this area should be investigated thoroughly from all aspects to determine whether it is possible to make use of it or not, since the eclipse time is very small (13.09 min) compared to other eclipses times which means less PV, batteries requirements.

It is concluded that Group (A) enables maximum value for each of the solar array operation coefficient which is calculated close to (93.83 %), average operation time during 3 and 5 years and total generated energy by solar array during the mission at altitudes ranging between (700-1200) km. More importantly, we arrived at the minimum mass and area for solar array simultaneously with minimum values for the batteries': mass, volume, capacity, and total number of cycles. Accordingly it saves the energy consumption by the heaters at the highest altitude of the selected range, this leads to reduce the charge/discharge cycles so it would be possible to use the battery with more DOD or use the battery for a longer life with the same DOD, which means lower degradation and higher performance.

Group (B) at the highest altitude of the selected range and group (C) at the area of inclination angle ranging between 40 deg and 45 deg are the optimum choice from the point of view of the communication designer although this will lead to higher energy consumption while the total generated energy by the solar array at this area will be double the required energy.

To explore an exact mission design successfully, we must remove the walls between the sponsor, space operators, users or customers, and developers so that they all become one team. A good team considers the mission's operations, objectives and requirements as well as the available technology to develop the best possible mission concept at the lowest possible Life-cycle cost.

In future the use of flywheel will help satellites to stay longer periods up to 20 years or more.

8. References

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