Spacecraft Design Lifetime

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A general discussion of issues that drive and limit spacecraft design lifetime is presented. The effects of varying
the spacecraft lifetime requirement on different subsystems are explored, and critical spacecraft mass and life
times are identified. Questionable analyses confirm that the design lifetime requirement is an important
subsystem and significantly affects the spacecraft mass and cost in initial operating capability. The analysis
introduces a formally defined economic metric, the cost per operational hour, that can be directly compared
to the design lifetime requirement. Furthermore, results suggest that other factors should also be taken into
account in specifying the design lifetime, namely, the value of signals resulting from technology advancement
as well as the market the system is competing in case of a commercial satellite.

Nomenclature

A = solar array area, m²
ρ = gravimetric fuel density, kg/m³
Isp = initial depleter of manufactured solar arrays, typically 800
η = solar array degradation, %
M = mass of subsystem, kg
p = power at beginning of life, W
P = power at end of life, W
P₂ = power required during eclipse, W
u₂ = probability that subsystem is operational
β = burnout of eclipse
βₙ = spacecraft design lifetime, years
Mₙ = maximum allowable temperature for the radiator, K
Kₙ = allowable level of the radiator
Vₙ = velocity increment, m/s
α = emissivity of radiator
Tₐ = total mass entrance velocity (110% for 3 and 17% for 6 cells) times array area
fₐ = radiator efficiency, typically 99%
fi = solar array efficiency
βₔ = failure rate of subsystem
σ = complex/deg. 8.85 x 10⁻¹⁹ W/m² K⁴
S = Specific Heat, J/kg K

Introduction

In recent years, several space programs have chosen to increase their spacecraft design lifetime. Over the last decade,
communications satellites, for example, have seen their design lifetime increase on average from 7 to 15 years. This trend is also
observed in the design and development of many high-value aircraft. An increase in average life span of a helicopter delivered today
can exceed 3 years at 20,000 lifetimes.

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In this case, increasing the spacecraft design lifetime was driven by the desire to maximize the return on investment (ROI).
However, extending the design lifetime has several side effects. Prolonging the design life leads to larger and heavier satellites as a result of several factors, such as additional propellant for orbit and on-demand
power generation and storage. This additional mass is in turn increased by the growth of the payload and increased power. Several
additional elements of the spacecraft design life-time increase the lift mass and the satellite design must be increased correspondingly. Before the end of the design lifetime, many components from which the system requirements were derived changed or were modified during the spacecraft's operational life-time. Setting a spacecraft design life-time requirement can therefore be a critical
task for space designers. However, what driven spacecraft design lifetime? How do designers, managers, and stakeholders un请选择合适的回答。
customer require for spacecraft design lifetime? The design lifetime should not necessarily be set to the maximum achievable value. For example, a spacecraft's design lifetime might not be enough to handle the current state of technology. New or enhanced payload capabilities, for example, a better solar array, might be developed and available within a couple of years following deployment. As a result, the need for a new satellite or mission may arise sooner than expected.

The authors in the first paper claimed that the spacecraft's maximum operational lifetime is determined by the maximum storage limit of the various subsystems. The purpose of this study was to investigate whether this is true. The objective was to determine whether the design lifetime requirement is met by the spacecraft subsystems and to investigate the effects of the design lifetime requirement on system performance. The results of this study are presented and discussed for the subsystems and the system as a whole. The implications of the results are also discussed.

Typical Mass Distribution of Satellite Systems

For the typical mass distribution of a satellite system, it is useful to identify the mass contributions of the subsystems to the spacecraft mass. Table 1 shows some typical mass distributions for spacecraft systems. For example, the thermal power subsystems (EPS) can be divided into 10% of the spacecraft mass, with a standard deviation of 10%. The EPS, along with the payload and spacecraft structure, are the major mass contributions and make up approximately 60% of the spacecraft mass.

Satellite Subsystems and Design Lifetime

We now examine how different subsystems interact with the design lifetime requirements. This is a key parameter in designing several subsystems for spacecrafts. For example, the EPS, and indirectly the SES, are influenced by the thermal environment. These influences and implications are explored qualitatively in Table 2. Table 2 represents the direct impact of the design lifetime requirement on the subsystems. The subsystems are listed in Table 1, along with the design lifetime requirement. The EPS is divided into two subsystems: the solar array subsystem and the thermal subsystem. The solar array subsystem is included in the design lifetime requirement. The design lifetime requirement is met if the subsystems' mass contributions are less than 10% of the spacecraft mass. The thermal subsystem is included in the design lifetime requirement. The design lifetime requirement is met if the thermal subsystem's mass contributions are less than 10% of the spacecraft mass.

Table 1: Percentage mass distribution, average and standard deviations (adapted from Ref. 1)

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Percentage</th>
<th>Average</th>
<th>Standard Deviation</th>
</tr>
</thead>
<tbody>
<tr>
<td>Communication</td>
<td>22.4%</td>
<td>22.3%</td>
<td>0.2%</td>
</tr>
<tr>
<td>Navigation</td>
<td>23.4%</td>
<td>23.3%</td>
<td>0.2%</td>
</tr>
<tr>
<td>Remote sensing</td>
<td>21.4%</td>
<td>21.2%</td>
<td>0.2%</td>
</tr>
<tr>
<td>Average</td>
<td>20.6%</td>
<td>20.5%</td>
<td>0.2%</td>
</tr>
</tbody>
</table>

Table 2: Design lifetime influence models

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>T&amp;C</th>
<th>EPS</th>
<th>Thermal</th>
<th>Structure</th>
<th>Propulsion</th>
<th>Propellant</th>
</tr>
</thead>
<tbody>
<tr>
<td>ADCS</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
</tr>
<tr>
<td>T&amp;C</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
</tr>
<tr>
<td>EPS</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
</tr>
<tr>
<td>Thermal</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
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<tr>
<td>Structure</td>
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<td>+</td>
<td>+</td>
<td>+</td>
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</tr>
<tr>
<td>Propulsion</td>
<td>+</td>
<td>+</td>
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<td>+</td>
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</tr>
<tr>
<td>Propellant</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
</tr>
</tbody>
</table>

The design lifetime requirement is met if the subsystems' mass contributions are less than 10% of the spacecraft mass. This is a key parameter in designing several subsystems for spacecrafts. For example, the EPS and indirectly the SES are influenced by the thermal environment. These influences and implications are explored qualitatively in Table 2. The thermal subsystem is divided into two subsystems: the solar array subsystem and the thermal subsystem. The solar array subsystem is included in the design lifetime requirement. The design lifetime requirement is met if the solar array subsystem's mass contributions are less than 10% of the spacecraft mass. The thermal subsystem is included in the design lifetime requirement. The design lifetime requirement is met if the thermal subsystem's mass contributions are less than 10% of the spacecraft mass.
Given a power requirement at EOL, the power output of the solar arrays at EOL scales inversely with life degradation $\Delta P_{array}$ and the solar arrays have to be overdesigned to accommodate this performance degradation. Figure 3 shows the relationship between the $P_{array}$ and the design lifetime for a 4-kW $P_{cell}$ requirement. For example, to deliver 4 kW at the EOL of a 10-year mission, the solar arrays should be designed to provide approximately 5.5 kW in the case of GaAs cells and 6 kW in the case of Si cells at the EOL.

The solar array area required to produce the $P_{array}$ is approximately

$$A_{array} = \frac{P_{array}}{\Delta P_{array} \times \eta}$$  \hspace{1cm} (3)

Given the specific performance of the array in watt per kilogram (or watt per square meter), the mass of a planar array can be directly evaluated. Typical specific performances range between 30 and 50 W/kg. Results for a nominal specific performance of 40 W/kg are presented in Fig. 3. For instance, to deliver 4 kW at the EOL of a 10-year mission, the solar arrays would weigh approximately 130 kg, that is, 22 kg in excess of a solar array delivering the same 4 kW at the EOL of a 5-year mission. This is equivalent to approximately 20% mass penalty for seven extra years of life.

Assumptions:

Spacecraft in Earth orbit undergo between 90 and 5500 eclipse per year. The former figure is typical of a geostationary Earth orbit (GEO) satellite, the latter for a satellite in LEO. During eclipse, electric power is supplied by secondary batteries that are recharged by the solar arrays when the spacecraft emerges into sunlight. In addition, there are some instances where batteries are called upon to
provide peak power in sunlight periods. The existing state-of-the-art and space-qualified batteries (nickel-hydrogen) are heavy and can constitute up to 15% of the dry mass of a typical communications satellite. Current secondary battery technology includes nickel-cadmium, which is a very common space-qualified secondary energy storage system. Nickel-hydrogen batteries are currently the energy storage system of choice for most aerospace applications where high specific energy and long design life are required. Lithium-ion and lithium-carbon batteries are currently under development with expected space qualification for GEO and LEO applications by 2003–2007. Lithium-ion and lithium-carbon batteries would offer significant mass and volume reduction compared to nickel-cadmium and nickel-hydrogen technology, as shown in Fig. 4.

The design lifetime significantly impacts the sizing of secondary batteries. Indeed, the amount of energy available from secondary batteries, the DOD, decreases with the number of cycles of charging and discharging. To be sure, the number of charge/discharge cycles is equal to the number of eclipses a satellite encounters during its design lifetime. Typically a satellite in GEO undergoes two periods of 24 days per year with eclipses lasting no more than 72 min, hence 90 cycles of charging and discharging per year. Satellites in LEO undergo approximately one eclipse per orbit. For a 90 min orbit, this amounts to 16 eclipses per year, or approximately 5000 cycles per year, with a maximum shadowing period of nearly 30 min per orbit. Figure 5 shows the DOD as a function of the number of charge/discharge cycles a battery undergoes, as well as the DOD as a function of the design lifetime of a satellite in GEO.

For instance, a 3-year mission in GEO, the average DOD for a nickel-cadmium battery is approximately 75%, but it drops to 62% for an extended mission of 10 years. How does this impact the sizing of the battery? Battery capacity is quantified as follows:

\[
C = \frac{P_c \times T_c}{(DOD) \times N \times n}
\]

Fig. 3 Solar array (GAL) mass, mass penalty, and percent mass penalty as a function of the design lifetime; reference mission is three years.

Fig. 4 Mass and volume of different types of batteries for a 10-kWh capacity.
The battery capacity scales inversely with the DOD. Therefore, as the number of cycles or the design lifetime increases, the energy available from the batteries during each cycle decreases, that is, the DOD decreases. Consequently, the batteries have to be overdesigned as the design lifetime increases. The mass of batteries can be obtained given the specific energy density of the battery. For nickel-cadmium batteries, the specific energy density ranges between 25 and 30 Wh/kg and for nickel-hydrogen between 40 and 60 Wh/kg (Fig. 6). Lithium-ion and lithium-carbon are expected to reach the 100 Wh/kg level. Figure 7 shows the evolution of battery mass as a function of design lifetime. The power delivered during eclipses is kept constant.

Figure 6 shows the advantage of nickel-hydrogen batteries over nickel-cadmium for high-energy capacity requirements and long mission duration. For smaller capacities, the mass of the nickel-hydrogen batteries, the mass penalty, and the percent mass penalty considering a three-year reference mission as a function of the design lifetime are given in Fig. 7.

The power distribution system (or subsystem) consists of cabling, fault protection, and switches in the form of mechanical or solid-state relays to turn power on and off to the spacecraft loads. Power regulation is required for two main tasks: 1) controlling the solar array power output to prevent battery overcharging and spacecraft heating and 2) regulating the spacecraft power bus voltage (or each load separately).

The solar array output is described by a plot of current (I) vs voltage (V). The I-V curve changes both due to seasonal variation in the array temperature and the solar intensity and due to radiation degradation of the solar cells as already discussed. The array voltage is maximum as the spacecraft comes out of eclipse when the temperature of the cell is minimum; hence, the need to regulate the solar array output. An unregulated bus has a voltage that varies significantly. This is unacceptable for most of the electronic equipment of the payload and the spacecraft (if voltage regulation is not provided separately at each load or equipment). Voltage regulators...
and converters get, therefore, placed either separately at each load or on the spacecraft power bus.\(^1\)

It is difficult to quantify how the mass of the power control unit and the power distribution system scale with the design lifetime. The power control unit as well as the cabling and harness are indirectly affected by the design lifetime because extra power is required at EOL. We use a mass estimate relationship to evaluate the mass (kilogram) of the power control unit (PCU) and the power distribution system (watts):

\[
M_{PCU} = 0.0045 \times P_{BOC}
\]  

(5)

The mass of the power distribution system constitutes a large part of the EPS mass, roughly 10-20%:

\[
M_{PDS} = 0.15 \times M_{EPS}
\]  

(6)

Figure 8 shows a typical mass breakdown of the EPS for a spacecraft in GEO in terms of its components, solar array, batteries, PCU, and power distribution, as a function of the design lifetime:

\[
M_{EPS} = M_{PCU} + M_{MBS} + M_{SD} + M_{PDS}
\]  

(7)

The mass, mass profile, and power mass profile for the electrical power subsystem as a function of the design lifetime are given in Fig. 9. The design lifetime for the reference mission is three years.

**Current**

The preceding sections presented a simple design process for sizing the solar array and the batteries. A limited number of parameters were considered, as well as two mass estimate relationships, to derive typical mass profiles of the EPS as a function of the design lifetime. These parameters included the power at EOL requirement, the
spacecraft orbital parameters to derive the cycle duration for the positioning of the batteries and the solar arrays during the passive phase of the orbit. The solar cell type or the cell energy conversion efficiency, the array-specific performance (watt per square meter or watt per kilogram), and the battery type or its specific energy density. The purpose of this analysis was to highlight and capture the impact of the design lifetime on the sizing of the EPS in a semiquantitative way. In reality, the design process of the solar arrays and the batteries is much more involved. Designers have a plethora of variables to trade and optimize. More elaborate design processes of the EPS are available in literature.  

Thermal Subsystem

A spacecraft contains many components that will function properly only if they are maintained within specified temperature ranges. The thermal design of a spacecraft involves identifying the sources of heat, designing proper heat transfer between all spacecraft elements, and rejecting heat so that different components stay within their operating temperature ranges. 

As in the preceding sections, we are concerned with the thermal subsystem’s mass scales with the design lifetime. It is useful to keep in mind in the following discussion that the thermal subsystem as a whole on average for only 6% of a spacecraft’s dry mass (see Table 1). A spacecraft’s thermal design is highly dependent on the mission class and the attitude stabilization type. Assuming a configuration of the thermal subsystem has been selected for a reference mission (selection of passive vs. active thermal control, thermal coating and insulating layer insulation, heat pipes, loops, radiators, electrical heaters, etc.), should the subsystem be redesigned if the spacecraft design lifetime varies? If so, how does its mass scale with the design lifetime? 

To answer the preceding questions, we first need to look into the different sources of heat that affect a spacecraft. These include solar radiation, Earth albedo and scattered radiation, and equipment power dissipation (electrical, antennae, and wiring). Although the first two are not affected by the design lifetime, it was shown earlier that the power requirement at BOL increases as the design lifetime increases due to solar array degradation. This power source (see Fig. 2) must be handled by the thermal subsystem. Therefore it is reasonable to assume that the thermal subsystem additional mass varies as a function of the difference between Pmax at BOL and Pmax at BOL: 

\[ \Delta M_{\text{thermal}} = f(P_{\text{max}}, \Delta P_{\text{max}}, \ldots) \]  

Radar systems are sized for the hottest conditions. The best balance equation can be written as follows: 

\[ \frac{c_2 T_{\text{max}}}{k} n_{\text{ant}} A = n A I_s \sin(\theta) + P \]  

The area of the radiator, and consequently its mass, is proportional to the power dissipation: 

\[ A = P f_c \left( \frac{c_2 T_{\text{max}}^2}{k} - \sigma I_s \sin(\theta) \right) \]  

Another effect has to be considered in sizing the radiator surface: the degradation of the thermal properties of its surface. Typically for an optical solar reflector (OSR) covering the radiator panel of a spacecraft in GEO, the solar absorptance and emissivity vary as shown in Table 3. The radiator area has to be sized for the worst case. When the OSR emissivity is assumed to be a fraction of the electric power delivered by the solar panels, the radiator surface can be estimated as follows: 

\[ A = \max \left( \frac{k \times P_{\text{BOL}}}{\sigma_{\text{BOL}} n_{\text{ant}} T_{\text{max}}^2 - \sigma_{\text{EOL}} n_{\text{ant}} T_{\text{max}}^2} \right) \]  

It is not clear which term dominates in this relationship. The variation in the solar absorptance of a spacecraft in GEO (Table 3), the electric power delivered by the solar panels, and the attitude stability of the spacecraft. In the preceding example, the first term drives the sizing of the radiator’s area. To first order, we will consider the mass of the radiator to be proportional to Pmax. 

\[ M_{\text{rad}} = k \times P_{\text{BOL}} \]  

Table 2: Thermal properties of OSR at BOL and EOL after seven years in GEO (adapted from Ref. 2)

<table>
<thead>
<tr>
<th>Period</th>
<th>Solar absorptance</th>
<th>Solar emissivity</th>
</tr>
</thead>
<tbody>
<tr>
<td>BOL</td>
<td>0.80</td>
<td>0.85</td>
</tr>
<tr>
<td>EOL</td>
<td>0.85</td>
<td>0.85</td>
</tr>
</tbody>
</table>

Fig. 9: EPS mass, mass penalty, and percent mass penalty as a function of design lifetime: nickel-hydrogen batteries and GaAs cells, reference mission three years, satellite in GEO.
We will also assume that the mass of the thermal subsystem mass scales with the mass of the radiator:

\[ M_{\text{thermal}} = k_3 \times P_{\text{EOL}} \]

For an active thermal control subsystem, values of \( k_3 \) that reflect realistic thermal control subsystem mass range between 0.020 and 0.035 kg/W. See Fig. 10.

The preceding discussion represents a first attempt at quantifying the effects of the design lifetime on the thermal subsystem. Although it is clear that the thermal subsystem scales with the design lifetime due to the excess power at BOL and the degradation of the thermal insulation optical properties, nevertheless, it is difficult to quantify these effects in a reasonably accurate way without taking into account a multitude of parameters regarding the spacecraft configuration, the type of thermal control, etc., as well as particular details about the mission. Such details are beyond the scope of this study.

**Telemetry, Tracking, and Control Subsystem**

The telemetry, tracking, and control (TT&C) subsystem interfaces between the spacecraft and the ground segment. This subsystem provides the hardware required for the reception, processing, storing, multiplexing, and transmission of satellite telemetry data. The command and data handling subsystem (C&DH), often subsumed under the TT&C subsystem, performs two categories of function: It receives, validates, decodes, and distributes commands to other spacecraft subsystems and gathers, processes, and formats spacecraft housekeeping data for downlink or use by the onboard computer.

As in the preceding sections, we are interested in how the TT&C and the C&DH subsystems' mass scale with the design lifetime. Table 1 shows that these two subsystems account on average for 5% of a satellite's dry mass. As with the thermal subsystem, these are minor contributors to the spacecraft mass. The TT&C design is driven by the following requirements: data rates for command and telemetry, data volume and storage type, uplink and downlink frequencies, bandwidth, receive and transmit power, antenna beamwidth, and antenna characteristics. Selection criteria for TT&C include performance (bit error rate, noise figure, etc.), compatibility with other existing systems, for example, the Tracking and Data Relay Satellite System, as well as technology risk.

These requirements, as well as the selection criteria, do not depend on the spacecraft design lifetime. Therefore, it is reasonable to assume that, to a first order, the mass of the TT&C does not depend on the design lifetime requirement. The same is true for the C&DH subsystem.

This argument breaks down, however, if we consider the effect of radiation on the onboard electronics and the need to provide additional shielding as the design lifetime increases. Furthermore, the reliability required of the C&DH will affect the subsystem mass as the design lifetime increases. Redundant components will be needed to maintain the same level of reliability for an extended lifetime, hence, increasing the amount of hardware and, consequently, the mass of the subsystem.

We will consider that the mass of the C&DH as well as the TT&C subsystems scale with the level of redundancy \( n \). This approach is further elaborated in the following sections.

**Reliability/Redundancy Issues**

This section is, in a large extent, based on Refs. 7 and 8.

The question we seek to answer or gain insight into is as follows: How does the spacecraft mass scale with design lifetime and mission reliability? Design lifetime is the intended operational time of the spacecraft on-orbit. Mission reliability is defined as the probability that the space system will function without a failure that impairs the mission, over a specified period of time or amount of usage. The elementary expression for the reliability of a single product is

\[ R = e^{-\lambda t} \]  

(13)

For a spacecraft composed of \( n \) nonredundant elements all equally essential to the spacecraft operation, the overall series reliability is

\[ R_s = \prod_{i=1}^{n} R_i = \exp\left( -\sum_{i=1}^{n} \lambda_i t \right) \]  

(14)

For \( n \) parallel or redundant elements, the overall parallel reliability is

\[ R_p = 1 - \prod_{i=1}^{n} (1 - R_i) \]  

(15)

Where the reliability of the elements is the same, Eq. (15) simplifies to...
We will focus on the first two components of the propellant budget.  
Two parameters that vary with the design lifetime affect the propellant budget: the satellite initial mass $M_0$ and the $\Delta V_{sat}$ required for station keeping over the mission duration.  
For instance, given a $\Delta V_{sat}$ requirement for orbit transfer from geosynchronous transfer orbit to GEO, the mission mass needed to provide this velocity increment is a linear function of $M_0$, as illustrated in Eq. (18).  
Because $M_0$ varies as a function of the design lifetime $T_{des}$ (Sec. 10), one does $M_{sat}(\psi)$:
$$
M_{prop}(\psi) = \psi(\Delta V_{sat}, f_p) \times M_0(T_{des}, \ldots)
$$  
where $\psi$ is given in Eq. (18).  
For example, for $\Delta V_{sat} = 1500$ m/s and $f_p \approx 0.3$, one has $\psi \approx 2$.  
In other words, the propellant required to perform the orbit transfer accounts for 40% of the spacecraft mass.  
The $\Delta V_{sat}$ required for station keeping can be estimated as follows:
$$\Delta V_{sat} = T_{sat} \times \Delta V_{p}$$  
The $\Delta V_{sat}$ yearly for station keeping is a function of the orbit altitude, the solar cycle (minimum or maximum), which in turns alters the atmospheric density, hence the drag encountered by satellites in LEO, or the longitude of station keeping for a satellite in GEO.  
Typically for a satellite in GEO, $\Delta V_{p} = 50$ m/s. Finally, the propellant mass required to provide $\Delta V_{sat}$ is given by:
$$
M_{sat} = M_0 \left[ 1 - \exp \left( \frac{\Delta V_{sat} + \Delta V_{p} \times T_{sat} + \Delta V_{col} + \Delta V_{mag}}{g \times f_p} \right) \right]
$$  
Station keeping is performed using a separate propulsion system from the orbit insertion system.  
In fact, Hall effect thrusters are used for station keeping because of their high specific impulse ($I_p \approx 1500$–$3000$ m/s).  
In this case, for a spacecraft in GEO, the mass of propellant required for station keeping per year accounts for 0.2–0.4% of the spacecraft mass at GEO, as opposed to 1.5–3% using the more traditional chemical propulsion system.  
Propulsion

The propulsion module subsystem consists of the tanks to hold the propellant, the pipes and pressure-regulating equipment, and the
As in the preceding sections, we are interested in how the propulsion subsystem's mass scales with the design lifetime, keeping in mind that the propulsion subsystem accounts for about 4% of a spacecraft's dry mass (see Table 1):

\[ M_{\text{propulsion}} = M_{\text{dry mass}} + M_{\text{propellant}} + M_{\text{masses}} \]  

(27)

As the design lifetime increases, the propellant budget increases. Consequently, the volume and mass of the tank necessary to hold the propellant increase. It is reasonable to assume that the other contributors to the propulsion subsystem mass remain unaffected by an increase in the design lifetime.

When the spherical tank of thickness \( c \) and radius \( r \) is assumed, the mass of the tank and the mass of its propellant are:

\[ M_{\text{tank}} = \rho(4/3 \pi r^3) \times c \]

where \( \rho \) is the mass density of the tank material.

Consequently, the mass of the propellant can be approximated as:

\[ M_{\text{propellant}} = \rho(4/3 \pi r^3) \times (3/2 \pi r^3) \]

(24)

Finally, we can relate the propulsion subsystem's mass to the propellant mass, which varies as a function of the design lifetime, with the following functional relationship:

\[ M_{\text{subsystem}} = a + b \times M_{\text{propellant}} \]

(26)

where \( a \) and \( b \) are constants and depend on the particular design of the propulsion subsystem. They do not vary with the design lifetime (Fig. 13).

### Attitude Determination and Control Subsystem

The attitude determination and control subsystem (ADCS) measures and controls the spacecraft's angular orientation. This subsystem is essential for spacecraft in desired orientations during different mission phases despite diurnal (or seasonal) variations (e.g., due to solar radiation, etc.), and is also used to orient the spacecraft to point the payload in different directions (e.g., maneuvering). Its mass accounts for about 2% of a satellite's dry mass (see Table 1).

The issue of concerns in this section is how the ADCS scales with the spacecraft design lifetime. The selection and sizing of the ADCS is driven by requirements on accuracy and range of angular motion both in terms of decommutation and control. For a three-axis stabilized spacecraft, the torque capability or control authority of reaction and momentum wheels is determined by the magnitude of the disturbance torques and the elements of the spacecraft inertia matrix:

\[ T_{\text{torque}} = J \omega \times F \]

(27)

For a mass \( M \) with an orthogonal coordinate system \((x, y, z)\) located at its center of mass, the moment of inertia about the \( z \) axis, for instance, is given by:

\[ I_z = \int \left( x^2 + y^2 \right) \, dM \]

(28)

As the design lifetime increases, the spacecraft mass increases (EPS, thermal subsystem, propellant, etc.); hence, the elements of its inertia matrix increase. Consequently, the ADCS has to be redesigned for a larger torque capability. How can we relate the torque capability of a wheel to its mass? Answering this question would provide an insight into the relationship between the ADCS mass and the spacecraft design lifetime. This step unfortunately is not straightforward.

In the absence of a physically based rationale for relating the ADCS mass to the design lifetime, we will use a substitute mass estimate relationship provided in Table 1 to evaluate the mass of a three-axis ADCS:

\[ M_{\text{ADCS}} = 0.06 \times M_{\text{subsystem}} \]

(29)

### Structures

The function of the spacecraft structure is to provide mechanical support to all subsystems within the framework of the spacecraft configuration. It also satisfies the subsystem requirements, such as alignment of antennas, actuators, sensors, etc., and the system requirements for launch vehicle interfaces and integration. The spacecraft structure is a major contributor to the spacecraft's dry mass and accounts for 21% of its dry mass (see Table 1). As in the preceding sections, we are interested in how the spacecraft structure mass scales with the design lifetime. To address this question, we start by examining the sources of structural requirements. Structures must endure mechanical loads in different environments, from manufacturing, to launch and normal operations. The environments from which the structural requirements are derived are listed in Table 4. Notice how these structural requirements can clearly relate the spacecraft design lifetime to the structural requirements and,

![Fig. 12 Typical propulsion subsystem mass as a function of propellant mass: a = 4 and b = 0.3.](image)
Table 4 Sources of structural requirements by mission phase (adapted from Ref. 1)

<table>
<thead>
<tr>
<th>Environment/Phase</th>
<th>Source of Requirements</th>
</tr>
</thead>
<tbody>
<tr>
<td>Manufacturing and assembly</td>
<td>Handling fixtures, stress induced by welding, etc.</td>
</tr>
<tr>
<td>Transport and handling</td>
<td>Crane or dollies, reaction, wind, air transport environments</td>
</tr>
<tr>
<td>Testing</td>
<td>Vibrations and acoustic tests, test fixture reaction loads</td>
</tr>
<tr>
<td>Postlaunch</td>
<td>Handling during stacking sequence, preflight checks</td>
</tr>
<tr>
<td>Launch</td>
<td>Steady-state booster accelerations, acoustic noise, transient loads during launch</td>
</tr>
</tbody>
</table>

Fig. 17 Spacecraft total mass, mass penalty, and percent mass penalty as a function of the design lifetime: spacecraft in GEO, mission reliability 95%, three-axis stabilized, GeO-2, Ni-H batteries, reference mission three years.

consequently, to the spacecraft structure mass. It is reasonable to assume that the spacecraft structure scales with the design lifetime because the different sub-systems enclosed within or supported by the structure, as well as the consumables, scale with the design lifetime (EPS, thermal, propulsion, propellant). It is not obvious, however, how the structure mass scales with the design lifetime. The least arbitrary approach is to maintain the mass estimate relationship given in Table 1 relating the spacecraft structure to the satellite dry mass

\[ M_{\text{struct}} = 0.21 \times M_{\text{dry}} \]  

(30)

Spacecraft Mass Profile

The spacecraft mass profile as a function of the design lifetime can now be illustrated by converting the effects of the design lifetime on the different sub-systems as already discussed. The independent variables include orb type and related parameters (eclipse duration, number of batteries charge/discharge cycles, degradation per year for solar arrays, \( \Delta V \), solar cell type and battery type, power at EOL, mission reliability, type of attitude control, and payload mass.

The spacecraft dry mass and total mass (launch mass) are calculated as follows:

\[ M_{\text{dry}} = M_{\text{sys}} + M_{\text{payload}} + n \times M_{\Delta V} + M_{\text{EPS}} + M_{\text{propellant}} + M_{\text{launch}} \]

\[ M_{\text{total}} = M_{\text{dry}} + M_{\text{launch}} \]  

(31)

Figure 13 shows typical spacecraft mass profile, mass penalty, and percent mass penalty as a function of the design lifetime. Note, for instance, that designing a spacecraft for 3 years instead of 15 years results in a mass saving of the order of 40%. Conversely, a mass penalty of 40% is incurred if a fixture is initially designed for 15 years instead of 3 years. The next step is to translate this mass penalty, or mass saving, into a cost penalty, or cost saving. This is undertaken in the next section.

Cost to IOC and Cost per Operational Day

This section is based on a large cost on Ref. 11, as well as on Ref. 12.

In this section, we are interested in isolating the effects of the design lifetime on the spacecraft cost. We will proceed by translating the various mass penalties established earlier into spacecraft cost profiles as a function of the design lifetime. To do so, we are undertaking the qualitative, advantages, and limitations of cost estimates, as well as the various components of a spacecraft cost. The following paragraphs summarize the basic costs modeling. A spacecraft's cost depends on its size, complexity, technology readiness, design lifetime, schedule, as well as other characteristics. Space systems have specific costs (cost-per-unit weight) of the order of $700,000/(kg (Ref. 12)). Specific costs, however, are not sufficient for predicting the real costs of spacecraft. Over the years, several governmental organizations have developed cost estimate relationships (CERs) that relate spacecraft cost or sub-system cost in physical, technical, and performance parameters. The CERs are based on an appropriate historical database of past satellite programs. The basic assumption of parameter cost modeling is that all satellites will cost next time what they cost the last time. CERs include both nonrecurring and recurring costs associated with a space system. Nonrecurring costs are commonly referred to as the procurement, development, etc., and evaluation (SDT&E) costs. These costs include the design, analysis, and test of prototypes and qualification units. Recurring costs include the cost to produce flight units. They are commonly referred to as the theoretical first unit (TFU) costs. This concept represents the cost of the first space-qualified satellite. Typical CERs include the range of the parameters used to
develop the correlations between the subsystems characteristics and their cost, the CER itself, and the associated standard error (SE). An example is given in Table 5.

Launched costs, on the other hand, are derived from published look-up tables. Reference 13 is the guide for launch systems characteristics and costs. Another approach to modeling launch costs is to evaluate an average cost per kilogram to orbit. For instance, the average cost to LEO per kilogram for both United States and European launchers is approximately $10,000. Finally, the cost to IOC is given by

$$\text{cost to IOC} = \text{TFU} + \text{RDT&E} + \text{cost of start} \quad (32)$$

Using the linear extrapolation of the launch costs, the cost to IOC can be plotted as a function of the spacecraft design lifetime. Figure 14 shows the range of uncertainty in the cost estimation of the spacecraft recurring and nonrecurring costs. Figure 15 shows typical spacecraft cost to IOC, cost penalty, and percent cost penalty as a function of the design lifetime. Note, for instance, that designing a spacecraft for 3 years instead of 15 years results in a cost saving of the order of 35%. Conversely, a cost penalty of 35% is incurred if a mission is initially designed for 15 years instead of 3 years. Figure 15 provides an answer to the question we set out to investigate in this paper, namely, how does the design lifetime requirement impact the total system (mass and cost) to IOC? The results confirm that the design lifetime is indeed a key driver of the space system cost and illustrate its particular impact on the various subsystems (EPS, thermal, propulsion, etc.). We can now define the cost per operational day of a spacecraft as follows:

$$\text{cost to IOC} = \frac{\text{cost of IOC}}{\text{design lifetime (days)}} \quad (33)$$

(Fig. 16). This metric corresponds to uniformly amortizing the cost to IOC over the entire intended mission duration (without accounting for the time value of money).

Table 6: CER for estimating subsystems TFU and RDT&E costs (adapted from Ref. 7)

<table>
<thead>
<tr>
<th>Component</th>
<th>TFU*</th>
<th>RDT&amp;E*</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight</td>
<td>13.1k</td>
<td>36.5k</td>
</tr>
<tr>
<td>Length</td>
<td>56.4m</td>
<td>31.9m</td>
</tr>
<tr>
<td>Volume</td>
<td>1126m³</td>
<td>44.2%</td>
</tr>
</tbody>
</table>

* Rand year 2000.

Within the interval of the design lifetime considered, the cost per operational day decreases monotonically. In the absence of other metrics, the cost per operational day provides the boundary of the design lifetime and designing spacecraft with increasingly longer lifetimes. It also suggests that a customer is always better off requesting the contractor to provide the maximum design lifetime: 

$$T_{\text{CE}} = T_{\text{max}} \quad (34)$$

This, however, is not necessarily true. Launching spacecraft with increasingly longer design lifetimes increases the risk for the satellite of becoming technically and commercially obsolete before the end of its mission. Thus, in specifying the design lifetime requirements, decision makers have to assess this risk of losing out due to both the obsolescence of their product's technology base as well as the likelihood of changing market trends, or the volatility of the market.

The system is serving, during the system's operational lifetime. These issues will be explored in a subsequent paper.

Limitations

The preceding analysis presents several limitations that degrade the accuracy of the results. First, to isolate and capture the effects of the design lifetime on the spacecraft mass and cost, a limited number of parameters were considered in the analysis, instead of the plethora of variables that subsystems experts typically have to track and optimize. This was done to maintain a manageable size analysis and to avoid drowning the key parameters and effects in background clutter.

The accuracy limitation results from the use of mass estimate relationships, such as in the case of the spacecraft structure. Although it is clear that the spacecraft structure, for instance, scales with the design lifetime by the fact the different subsystems enclosed within or supported by the structure scale with the design lifetime (EPS, thermal, propulsion, etc.). It is not possible to relax the spacecraft structure's mass to the design lifetime without taking into account partial data details about the mission or the spacecraft configuration and layout. In other words, a preliminary design of the spacecraft is required to estimate reasonably the mass of the spacecraft structure. In light of the objectives set forth in the introduction and summarized earlier, such an analysis is beyond the scope of this study. In the absence of quantifiable physical arguments for relaxing a subsystem's mass to the design lifetime, mass estimate relationships were used as the least arbitrary way to proceed with the analysis.

![Fig. 14](image-url) Cost to IOC for a LEO spacecraft as a function of the design lifetime (three SE above and below the nominal CER output, same parameters as in Fig. 13).
The third limitation is due in part to the use of cost estimate relationships and dollars per pound to estimate launch costs. This resulted in smooth or continuous cost profiles instead of the discontinuous profiles that would be obtained in reality because of the performance and cost of existing launch systems, for example. $13 million for less than 1000 lb to LEO on Pegasus XL and $22 million for less than 3000 lb to Taurus. The availability and use of commercial-off-the-shelf hardware, which exists in discrete performance bins and does not necessarily match the customer's needs exactly, will also render discontinuous both the mass and cost profile of a spacecraft as a function of the design lifetime.

Some of the limitations discussed render the task of building generic models relating the spacecraft mass and cost to the design lifetime very challenging. However, in practice, the aforementioned inaccuracies will be attenuated when, during the conceptual design phases of a particular spacecraft, designers evaluate the mass and cost of their particular design at discrete values of the design lifetime, for example, three, five, seven, nine years, all of these being equal (performance and reliability). Thus, more accurate estimates could be obtained for the mass and cost of the spacecraft, or its cost per operational day and help guide the selection of the design lifetime.

Conclusions
This paper explored the impacts of the design lifetime on the spacecraft mass and cost to IOC. It first examined how different subsystems scale with the design lifetime, using physically based
arguments whose possible and mass estimate relationships in orbit instances. The data were then transformed to generate space- craft mass and cost profiles as a function of the design lifetime. Preliminary results confirm that the design lifetime is a key requirement in sizing various subsystems. For instance, a mass and cost penalty of 20-40% is typically incurred when designing a spacecraft for 15 years instead of 3 years, all else being equal. It was also shown that the cost per operational day decreases monotonically with the design lifetime. This finding justifies putting the boundary of the design lifetime and designing spacecraft with increasingly longer lifetimes. It also suggests that a customer is always better off asking the contractor to provide the maximum design lifetime achievable. This, however, may not always be the case. The decision regarding the design lifetime requirement should incorporate external factors such as the availability of the technology embedded in the spacecraft, the relationship between technology obsolescence and market share, and the volatility of the market is serving in the case of a commercial satellite.

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A. C. Tezrhan
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