Rapid method for spacecraft sizing

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1 INTRODUCTION

The total mass of a satellite has a great influence on the development and launch costs. In order to avoid significant deviations from the initial budget, it is sensible to have a reliable mass estimation at the very beginning of the project, before facing the final programme development.

For designing a spacecraft, the mission must be understood, including the payload's size and characteristics, plus significant system constraints such as orbit, life time, and operations. The design process involves identifying these functions, choosing candidate approaches for each function, and selecting the best solutions.

A spacecraft design is an iterative process. It can be broadly divided into three phases: preliminary design, detailed design and development. First of all, a feasibility study must be carried out to determine whether the mission performance requirements can be met within the mass and size constraints of the launcher. The first step in the actual design is to select a spacecraft configuration, which implies a general arrangement of the subsystems. The mass and power requirements of the subsystems are estimated, based on a preliminary analysis and extrapolation of the existing designs. Once the feasibility of the mission is confirmed and the initial design completed, a detailed design of the subsystems starts with analysis and test carried out at the unit and subsystem level.

The spacecraft design is qualified at the subsystem and system levels by conducting performance, thermal, and vibration tests. After successful completion of the qualification tests, the spacecraft design is finalized and the required number of flight spacecraft are assembled. During these steps it is necessary to confirm that all the estimations of the previous steps are fulfilled, and the initial mass, power and size budgets are not exceeded.

2 EXISTING METHODS FOR SPACECRAFT SIZING

There are few similarities among existing satellites because they are conceived to carry out a specific mission, so that a series fabrication is not common practice as in the rest of the aerospace industry. Thus, a reliable mass estimation is difficult to achieve because it is not easy to establish correlations to obtain estimations based on existing systems. In fact, only when the design process is in an advanced phase is it possible to know the total mass of the satellite by adding the masses of the different elements. The aim of this paper is to help in developing some useful sizing correlations.

As is known, the total satellite mass depends mainly on the payload (mass and power), its lifetime, the requirements imposed by the payload on the spacecraft bus, the orbit and the launch vehicle. Since 1978 attention has been paid to the mass and power estimations for geosynchronous satellites. Most of the authors agree in distinguishing three groups of masses: payload, spacecraft bus and propellant mass; in some cases the mass of the adapter to the launch vehicle is also considered. It is important to point out that it is necessary to define clearly the payload, the objectives, requirements and restrictions of the mission before sizing the spacecraft system.

A straightforward procedure to estimate the total mass starts with an initial identification of the basic elements of each subsystem; then, by using typical values of its masses and adding them, the total mass is obtained (**1**, **2**). A second procedure estimates the subsystem's masses using the satellite total mass as the main datum. In this procedure the power subsystem mass is not considered (**3**). For geosynchronous satellites structural mass has been correlated with the dry mass (**3**). An empirical formula has been proposed as a function of the payload and power subsystem masses which have to be calculated keeping in mind the mission profile (**3**). The aim of the last procedure has been to obtain the mass budget using only the mission objectives. The kind of attitude control (three axis or spin stabilization) must be taken into account before estimating the structural mass (**1**, **3**). This classification seems appropriate because the loads induced by the stabilization devices are very different in each case. Likewise, the propulsion subsystem can be independent or integrated into the attitude control subsystem (**1**).

The specific influence of the mission profile on the propellant (and thus on the propulsion subsystem) must be emphasized. The propellant burnt during the transfer manoeuvre, M_P, is calculated using the initial and final orbits and the satellite mass as follows (**1**, **2**, **3**):

$$M_P = M_i(1 - e^{-\Delta V/I g_0}) \tag{1}$$

where M_i is the mass at the beginning of the manoeuvre, ΔV, the total velocity change, I, the propellant specific impulse, and g_0 the mean sea level acceleration due to gravity. The exponential influence of ΔV is responsible for the large variations of the propellant mass ratios.

Finally, the propellant used in attitude control during the mission can be estimated as a function of the life and the initial mass (**1**) or by studying in detail the manoeuvres (velocity correction and control, control during total velocity change thrusting, spin-up and de-spin, manoeuvering while spinning, cancelling disturbance torques, attitude manoeuvring and limit cycling) (**2**).

A third method existing in the literature correlates the structural mass and the non-structural mass (**4**). This correlation has been obtained from data of 18 satellites, but the satellites considered have very different characteristics (total masses between 147 kg and 1297 kg); consequently the difference between the real mass and the estimated one can be up to 50 per cent.

3 PROPOSED METHOD

Estimating the mass of a satellite is necessary early in programme planning because both the cost of the satellite and of the launch are dependent on that mass. In fact, it is only possible to do this accurately by the systematic addition of the masses of the individual components. This procedure, however, is practical only after the detailed design has been finished and all the characteristics of the satellite established. At this point, spacecraft manufacturers can calculate the on-orbit mass typically with a tolerance of a few per cent. Unfortunately, the setting of the principal system and spacecraft characteristics, which in turn permits manufacturers to do careful mass and power estimating, is itself dependent on the economics of the system and therefore on the aforementioned masses and power.

The aim of this section is to present a set of correlations which constitute a new rapid method for spacecraft sizing to be used at the beginning of the project. The mass estimations obtained can be helpful for getting a more accurate cost budget. This method does not use a new philosophy but unifies the criteria mentioned in the previous section. The section begins with a classification scheme of the diverse masses that can be distinguished in a spacecraft. Afterwards, using this classification, data from a selected group of satellites are presented in two tables. Finally, by the means of correlations between data from these tables, the proposed method is developed.

Figure 1 depicts a block diagram of the proposed methodology. Once the mission objectives are fixed, the first step is to select the preliminary spacecraft configuration, and the launcher vehicle. At this point, the spacecraft's masses are proposed to be distributed according to Table 1. Then, the correlations presented in this section allows us to refine the previously estimated masses, using the total mass as the basic datum.

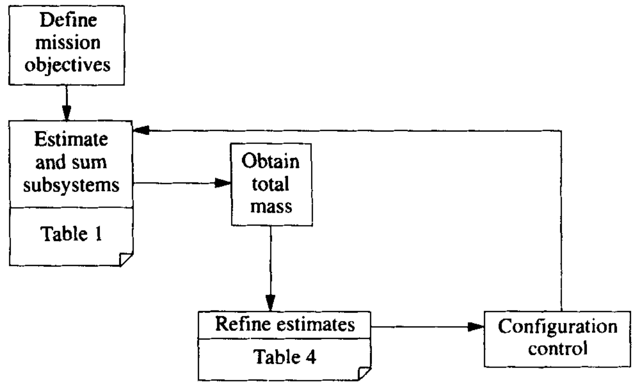

Fig. 1 Methodology block diagram

If during this process large deviations are noted, the initial mass distribution must be revised.

In order to size the satellite there is a distinction between the payload mass, M_{PL}, and the spacecraft bus mass, M_{SB}, that sum up to give the dry mass, M_{DRY}, by adding a margin associated with those elements that do not belong to a specific subsystem. Table 1 shows how the spacecraft bus mass has been divided into the subsystems masses. If the spacecraft carries propellant itself, M_{PROP}, the propellant mass must be added to the dry mass to give the beginning of life mass, M_{BOL}. The mass injected in orbit, M_{INJ}, is the sum of the beginning of life mass, M_{BOL}, and the mass of the kick stage, M_{kick}. Finally, the total mass to be considered for the launcher, M_T, requires us to keep in mind the adapter system mass, M_{adaptr}.

Table 2 depicts a complete mass distribution for the selected satellites [data have been collected from references (**1**, **2**, **4**, **5**) and (S A Hispasat, 1993, personal communication)]. Due to its importance, the kind of propulsion system (that is, orbit correction system integrated into the guidance, navigation and control system or not) and the type of stabilization used are indicated. Table 3 has been elaborated using directly the subsystems mass budget of several satellites appearing in Appendix A of reference (**1**).

From the analysis performed with data from Tables 2 and 3 (the last one for spacecraft subsystem sizing only), linear regressions between the different masses have been obtained.

The first step in the feasibility study of a mission is the launcher identification, as was mentioned in the

Table 1 Mass budget

Payload	M_{PL}
Spacecraft bus	M_{SB}
Propulsion	$M_{PROPULSION}$
Guidance, navigation and control	M_{GNC}
Communications	M_C
Command and data handling	M_{CDH}
Thermal	M_{TH}
Power	M_{POW}
Structure and mechanisms	M_{SM}
Margin	M_{Mar}
Dry mass	$M_{DRY} = M_{Mar} + M_{PL} + M_{SB}$
Propellant	M_{PROP}
Beginning of life mass	$M_{BOL} = M_{PROP} + M_{DRY}$
Kick stage	M_{kick}
Injected mass	$M_{INJ} = M_{kick} + M_{BOL}$
Adapter	M_{adaptr}
Total mass	$M_T = M_{adaptr} + M_{INJ}$

Table 2 Mass distribution for selected satellites (kg)

	Hispasat	IUE	FlSatCom	Heao-B	SPOT	OTS	LandSat	Intelsat VI	Intelsat V	DomSat	TVSat
Payload	280.19	117	222	1468	734	56.5	299	635	234	97	252
Spacecraft Bus	693.92	293	618	1414.5	662	270.8	398	932	522	323	716
Propulsion	98.26	45	29.5	25.5		27		120	96	41	123
GNC	47.76	45	57.7	124	106	38.2	21	70	73	23	52
Communications	44.71	46	26.3	71.6	10	21.7	200	80	28	27	29
Thermal	56.34	11	14.5	35.4	18	18.7	14	52	26	18	85
Power	179.23	61	243.4	301.9	158	75		330	142	124	249
Solar Arrays	98.61	23	92.6	77.1		29.7	71				
Structure	169.01	62	154	779	370	60.5	92	280	157	90	178
Margin	81.29	41		148		44.6	63	122	55	22	60
Dry mass	1055.4	451	840	3030.5	1396	371.9	1564	1689	811	442	1028
Propellant	242.3	19	83	138	238	45.2	77	538	197	105	1287
B.O.L. mass	1297.7	470	923	3168.5	1634	417.1	1641	2227	1008	547	2315
Kick stage	857.8	199	855		178	433		1449	861	493	
Injected mass	2155.5	669	1778	3168.5	1812	850.1	1641	3676	1869	1040	2315
Adapter			19.5	18.1							
Total mass	2155.5	669	1797.5	3186.6	1812	850.1	1641	3676	1869	1040	2315
Characteristics											
Orbit (km)	Geosyn.	—	Geosyn.	540	832	Geosyn.	700	Geosyn.	Geosyn.	Geosyn.	Geosyn.
Mission	Comm.	Scientif.	M. & Com.	Scientif.	Scientif.	Scientif.	Scientif.	Comm.	Comm.	Comm.	Comm.
Stabilization	3 axis	3 axis	3 axis	3 axis	3 axis	3 axis	3 axis	Spin	3 axis	Spin	3 axis
Propulsion System	Combi.	Classic	Classic	Classic	Combi.	Classic		Classic	Classic	Classic	

Comm. = communications; Scientif. = scientific; M. & Com. = military and communications; Combi. = liquid bi-prop.; Classic = solid propulsion.

Introduction. An initial estimation of the total mass can be obtained keeping in mind the payload capacity of the launch vehicle, the type of mission and orbit, as is illustrated in (1). Using this datum, the value of the dry mass can be estimated. Equation (2) has a high degree of correlation for geosynchronous spacecraft, as the data from seven such spacecraft, detailed in Table 2, show in Fig. 2. It is not suggested that equation (2) applies as accurately to all other types of spacecraft. In particular the contribution to fuel load for the differing missions will have a significant impact upon the total mass. Having noted this, however, it does appear that there is a fairly good correlation for IUE and LandSat. The correlation for Heao-B and SPOT is poor. The application of equation (2) needs therefore to be handled with care, but can be used with confidence for geosynchronous spacecraft, and possibly as a first approximation for other types.

$$M_{DRY} = 11 + 0.47 M_T \qquad (2)$$

For geosynchronous spacecraft studied so far, plus IUE and LandSat, once the dry mass is known, the spacecraft bus mass can be estimated, which yields an adequate estimate for the payload mass. At this point,

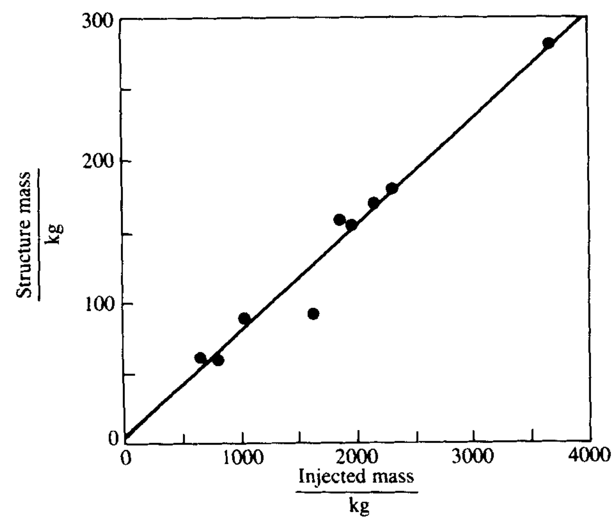

Fig. 2 Structure and mechanisms subsystem mass versus injected mass

Table 3 Satellites considered to complete the subsystem sizing. (Masses are expressed in kg)

	Payload	Spa. Bus	Propul.	GNC	Subsystems Comm.	Masses Thermal	Power	Structure	Dry Mass	Propellant	B.O.L.
FLTSatCom 1–5	225.48	624.03	33.47	59.56	25.15	14.87	327.35	163.63	849.60	81.40	930.90
FLTSatCom 6	229.74	641.25	33.36	58.96	26.04	17.33	343.05	162.51	870.90	109.10	980.00
FLTSatCom 7–8	341.74	700.27	34.80	59.18	26.05	22.30	341.22	216.72	1041.90	109.00	1150.90
DSCS II	109.55	366.36	14.10	54.54	33.17	13.18	139.53	111.84	475.90	54.10	530.00
DSCS III	280.48	579.54	35.47	37.73	62.71	48.22	237.73	157.68	867.30	228.60	1095.90
NATO III	70.87	246.11	7.79	20.28	24.06	20.86	111.31	61.81	320.40	25.60	346.10
Intelsat IV	166.45	366.52	16.73	39.48	22.91	27.39	141.14	118.87	532.80	136.40	669.20
TDRSS	384.54	1163.79	108.35	96.60	63.72	43.53	412.72	439.87	1565.70	585.30	2150.90
GPS Blk 1	98.17	382.94	17.30	29.51	27.98	41.68	171.37	95.10	479.10	29.50	508.60
GPS Blk 2, 1	140.87	558.29	23.00	37.82	36.35	68.93	216.51	175.68	699.10	42.30	741.40
GPS Blk 2, 2	197.51	659.56	23.00	45.05	26.60	94.64	252.60	217.67	858.00	60.60	918.60
P80-1	699.83	1094.58	104.48	197.89	88.80	40.05	339.52	323.84	1704.40	36.60	1740.90
DPS 15	780.61	1301.29	47.16	116.53	81.21	10.15	569.75	476.49	2114.90	162.40	2277.30
DMSP 5D-2	243.16	430.52	60.44	25.01	20.04	22.73	174.98	127.32	814.60	19.10	833.60
DMSP 5D-3	308.25	646.29	87.67	29.50	20.45	29.05	293.26	186.36	1012.30	33.10	145.50

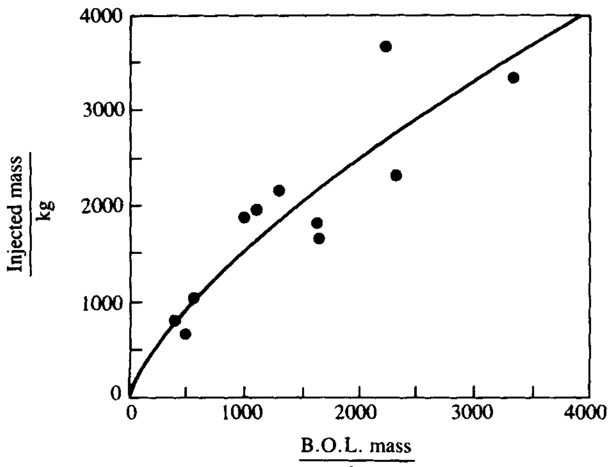

Fig. 3 Correlation between injected and B.O.L. masses

evaluations for the payload, spacecraft bus and dry masses are available. The propellant mass can be calculated taking into account the mission profile, as suggested in reference (1) (see Section 2). The beginning of life mass, M_{BOL}, is the sum of the propellant and dry masses

$$M_{BOL} = M_{DRY} + M_{PROP} \tag{3}$$

which is related to the injected mass, M_{INJ}, with a logarithmic regression shown in Fig. 3 obtained with the data of Table 2. M_{BOL} subtracted from M_{INJ} gives the mass of the kick stage and, in this way, there is a general description of the main masses dependent on the spacecraft mission.

The next step is the sizing of the subsystems' masses, using the previously estimated ones. For the sake of simplicity, the formulae are not going to be presented one by one but forgathered in Table 4. The structure and mechanisms mass can be related to either the injected or dry masses. However, the guidance, navigation and control, thermal and propulsion subsystem masses are preferably calculated from the beginning of life mass. Finally, the power subsystem mass is closely related both to the payload and dry masses.

Figure 2 depicts the relationship between the structure and mechanisms mass and injected mass. This expression has been obtained for the satellites that appear in Table 2 and it is highly accurate ($r = 0.98$).

Table 4 Regression coefficients A and B for equation (4)

$\frac{X}{\text{kg}}$	$\frac{Y}{\text{kg}}$	$\frac{A}{\text{kg}}$	B	r
M_T	M_{DRY}	11	0.47	0.98
M_{DRY}	M_{SB}	173	0.46	0.94
M_{SB}	M_{PL}	−91	0.60	0.89
log M_{BOL}*	log M_{INJ}*	0.71 log 11*	0.71	0.92
M_{INJ}	M_{SM}	4	0.075	0.98
M_{DRY}	M_{SM}	−34	0.24	0.96
M_{BOL}	M_{GNC}	3	0.072	0.93
M_{BOL}	M_{TH}	383	0.025	0.78
M_{BOL}	$M_{PROPULSION}$	−4.35	0.06	0.84
M_{DRY}	$M_{PL} + M_{POW}$	−53	0.63	0.98

* Units above indicated are applicable to the variables involved and not to the input and output.

Data from Heao-B and Spot have had to be rejected because they are scientific satellites (astronomical and Earth observation respectively) and their payloads require very stiff structures compared with the other cases. Using all the data available in Tables 2 and 3, the structure and mechanisms subsystem mass has been correlated with the dry mass. The advantage of this expression is the number of satellites involved, 26 against nine for the correlation shown in Fig. 2. Also, a good agreement has been found ($r = 0.96$).

The formula relating the guidance, navigation and control subsystem mass with the beginning of life mass is only appropriate for three-axis stabilized satellites. A general expression for all types of stabilization is difficult to obtain because of the different requirements imposed by the stabilization subsystem. The correlation of Table 4 has been obtained using only the data from the three-axis stabilized satellites of Table 2 with the exception of TV-Sat. Something similar happens in the case of the thermal control subsystem. The devices used in this subsystem are strongly dependent on the orbit, the payload and the other subsystems' design so data are scattered and it is not possible to establish formulae with a good correlation. Although the formula in Table 4 is only adequate for geosynchronous satellites, the correlation coefficient is 0.78. It must be pointed out that, in this case, the satellites considered have been all the ones in Table 3.

Propulsion subsystem configuration has evolved from complex systems to lighter ones with similar performances. The proposed correlation has been computed with data from the 26 satellites considered in this paper. The degree of correlation is 0.84 and, therefore, the estimated mass must be corrected depending on the technology selected.

Power subsystem mass does not show any evident tendency in relation with the principal masses of the spacecraft. However, with the idea suggested in reference (3), a highly accurate correlation ($r = 0.98$) has been obtained between power plus payload masses and the dry mass, as can be seen in Fig. 4. All the selected satellites have been considered in this analysis with the exception of Landsat and Heao-B; Landsat's power subsystem mass was not found and Heao-B showed a great disagreement with the others.

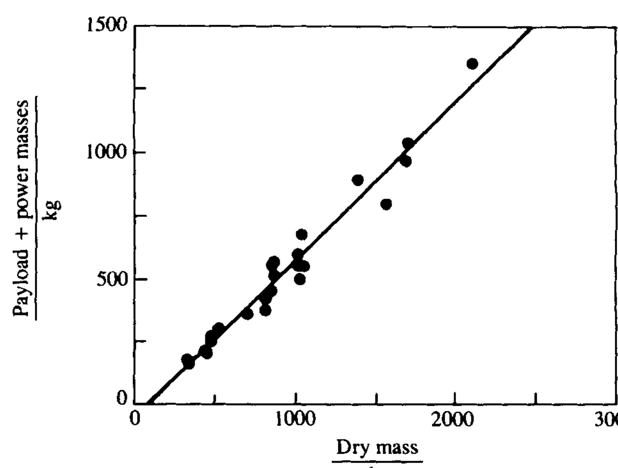

Fig. 4 Correlation between the sum of payload and power subsystem masses with the dry mass

The complete set of correlations can be expressed generically as follows:

$$Y = A + BX \quad (4)$$

The statistically determined regression coefficients, A and B, and the degree of correlation, r, are summed up in Table 4.

5 CONCLUSIONS

A systematic and rapid method for mass sizing of satellites has been presented. This method allows a preliminary estimation of overall spacecraft mass and the mass of each subsystem and to establish an initial mass budget for the spacecraft. Twenty-six satellites have been considered, and all the formulae show a high degree of correlation. The aim of this method is the sizing of the satellite up to subsystem level extending the applicability of the methods found in literature. This method is applicable to large satellites, independently of their mission. Therefore, during the analysis of the data, difficulties have been found when the configurations of the satellites were very different. This inconvenience has been avoided by rejecting some scattered data. One way to improve this work will be to perform a similar analysis, but to divide the data into sets corresponding to different types of missions; although the method will be more complex in its applicability.