18.8 Examples

18.8.1 FireSat II

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Continuing with the FireSat II example, we know that the total ΔV required for translational motion is 764 m/s. FireSat II is a 3-axis stabilized spacecraft, and it will carry an extra 4 kg of fuel for ACS as described in Sec. 14.7.2.

In order to provide pure moments about an axis in one direction (with zero net force), we need 2 thrusters. Assuming that each thruster has a fixed thrust direction with respect to the spacecraft, a total of 12 thrusters are needed for stabilizing a spacecraft in 3 axes. Alternatively, reaction wheels can also be used for controlling rotational motion. For example, 4 reaction wheels were used in the Lunar Reconnaissance Orbiter to provide pure rotations in all axes. Two redundant sets of 4 thrusters were used to unload the wheels. However, the 4 thruster configuration yields resultant forces in addition to the desired moments about a given axis. For this example, we will assume that we have 4 thrusters for unloading the wheels and 1 main thruster for primary propulsion.

The ΔV needed for FireSat II is within the range of monopropellant thrusters, but let's compare several options before we decide. From Sec. 14.7, we know the payload mass is 20 kg, but we are carrying a 30% margin so the total payload mass is

$$M_{payload\_max} = 20 \text{ kg} \times 1.3 = 26 \text{ kg} \qquad \text{(18web-2)}$$

From Table 14-18 LEO with propulsion, we calculated in Sec. 14.7 that the payload is 31% of the dry mass, so

$$M_{dry} = M_{payload\_max}/0.31 = 84 \text{ kg} \qquad \text{(18web-3)}$$

Since we know the final mass (M_{dry}), we can use the rocket Eq. (18-19) to calculate how much propellant is required. However, we need to assume a value of I_{sp}. If we choose different types of thrusters for translational motion and for attitude control, then the propellant masses for each type of thruster need to be calculated separately. At this point, though, we don't know what the thrust requirements are for each class of thruster, so let's pick an "average" I_{sp} for the system. Considering monopropellants (Table 18-5), the I_{sp} varies from about 200–235 sec. For this example, let's choose an intermediate value of I_{sp} = 218 sec. Substituting this into the rocket Eq. (18-19), we obtain the propellant mass needed,

$$M_p = M_f\left[e^{\Delta V/\left(I_{sp}g_o\right)} - 1\right] = 84\text{ kg}\left[e^{\Delta V/\left(I_{sp}g_o\right)} - 1\right]$$
$$= 84\text{ kg}\left[e^{764/(218\times 9.81)} - 1\right] = 36.1\text{ kg} \qquad \text{(18web-4)}$$

For the attitude control maneuvers, candidate thrusters are the *MRE-1.0* or the *MRC-111,* which have the added advantage of being flight-proven. From Table 18-5 we see that the mass of these thrusters varies from 0.33 to 1 kg. For estimation purposes, let's assume a mass of 1 kg per thruster. For the primary propulsion thruster, a potential candidate is the *Monarc-445* (1.6 kg), but we need to have more information on the thrust levels required before making a final decision.

To size a bipropellant system (see Table 18-6), let's take an I_{sp} of 291 sec, taken from values for low thrust engines like *5lb Cb* (0.82–0.91 kg) and the *10N Bipropellant thruster* (0.35–0.65 kg). In that case, the rocket equation gives us M_p = 25.8 kg. That is a savings of 10.3 kg in propellants compared to the monopropellant system. We need to compare the masses of the monopropellant and bipropellant thrusters to see if the overall mass savings are enhanced or reduced. Looking at Fig. 18-8 a bipropellant system is more complicated (i.e., it has more components that can fail) and costly than a monopropellant system. Therefore, we need to trade off the price, availability (and the other ilities described in Table 18-1) of the two systems to see if the added mass of the monopropellant system is justified.

Assuming that a monopropellant system is chosen after the different trades, the propellant mass needed to satisfy the ΔV requirements, usually denoted as usable propellant mass, M_{p_usable}, is 36.1 kg plus 4 kg for ACS for a total of 40.1 kg. Not all of the propellant loaded into a tank is usable. As a rule of thumb [Brown, 2002], a 3% margin is applied to the usable propellant to account for propellant trapped in the tank, feed lines, or valves. Also, there is a measurement uncertainty of about 0.5% on propellant loading. Then the total propellant mass loaded into the tank is,

$$M_{p\_loaded} = M_{p\_usable}\left(1.0 + 0.03 + 0.005\right)$$
$$= 1.035 M_{p\_usable} = 1.035\times 40.1 kg = 41.5 kg \qquad \text{(18web-5)}$$

The mass for the propellant tank can be estimated by using the methods described in Sec. 18.5.1. From the propellant mass calculated above and using the density of hydrazine (ρ) at 293 K from Table 18-8, the volume of the propellant loaded is,

$$V_{p\_loaded} = \frac{M_{p\_loaded}}{\rho_p} = \frac{41.5kg}{1.01kg/L} = 41.1L \tag{18web-6}$$

From which the volume of the usable propellant is V_{p_usable} = 39.7 L. The next step is to decide whether this system will be blow-down or pressurized. Since there are some commercial off-the-shelf thrusters that could potentially meet our needs, we would contact the manufacturers and get more details on the thrusters to best design the rest of the propulsion system. However, for this example, we will baseline a blow-down system since monopropellant thrusters have a large range of operating pressures and simplicity and low cost are important drivers for FireSat II. As the tank empties in a blow-down system, the pressure, and consequently the thrust, decrease. Therefore, this choice would not be feasible for a spacecraft that requires constant thrust throughout the mission. The blow-down ratio, B, is defined as,

$$B = \frac{V_{ullage\_f}}{V_{ullage\_i}} = \frac{P_{press\_i}}{P_{press\_f}} \tag{18web-7}$$

where V_{ullage_f} is the final ullage volume, V_{ullage_i} is the initial ullage volume, P_{press_i} is the initial pressurant pressure and P_{press_f} is the final pressurant pressure. Let's assume $B=3.5$, which is a typical value for blow-down propellant tanks.

The initial volume of pressurant needed is given by,

$$V_{ullage\_i} = \frac{V_{p\_usable}}{B-1} = \frac{39.7L}{3.5-1} = 15.9L \tag{18web-8}$$

As mentioned in Sec. 18.5.1, it is customary to carry a 20% margin on the volume of the propellant loaded, which would increase the initial volume of the ullage by 20% as well to give us 19.0 L. The final tank volume is 1.2 × (41.1 + 15.9) = 68.3L.

From Figs. 18-9 and 18-10 respectively, the mass of a tank with a volume capacity of 68.3L can range from 5.6 to 8.2 kg for a PMD and a diaphragm tank respectively. Let's take the diaphragm tank as a baseline design since it represents a worst case scenario for the weight, and it can handle a large array of accelerations.

Now that we have the mass of the tank and the propellants, we need to calculate the mass of the pressurant. We don't know the initial operating pressure for the tank, but we can make some educated guesses. For example, the operating range for the *MRE-1.0* is 0.055 to 3.9 MPa and for the *Monarc 445* is 0.5 to 3.1 MPa. To carry a 20% margin over 3.9 MPa, let's assume the initial pressure is 4.7 MPa, the initial temperature is 323 K and that the pressurant is Helium (He). Then the mass of the pressurant is given by

$$\begin{aligned} M_{press} &= V_{ullage\_i} \times \rho_{press\_i} \\ &= 19.0e-3m^3 \times 6.87\ \text{kg}/m^3 = 0.13\ \text{kg} \end{aligned} \tag{18web-9}$$

where ρ_{press_i} is the initial density of the He, obtained from the NIST Chemistry WebBook, [2010]. Had we chosen N_2 to pressurize the tank, the mass would have been 0.93 kg (ρ=48.9 kg/m^3). There is a significant percentage savings in mass by choosing He though both values are small compared to the total mass of the propulsion system.

The last thing to estimate is the mass for the feed system. A detailed estimate would involve creating a schematic like the one shown in Fig. 18-8b and determining the quantity and type of required valves, filters, transducers, along with the length, diameter and material of the tubing used. Without that information, as a rule of thumb for liquid propulsion systems, the mass of the tank and plumbing is about 10% of the total mass of the propulsion system, and it can be higher (up to about 20%) for smaller thrusters. As a conservative guess, let's assume 20%. Then,

$$M_{tank} + M_{feed} = 0.20 \times (M_{tank} + M_{feed} + M_{p\_loaded} + M_{press} + M_{thruster}) \tag{18web-10}$$

$$M_{feed} = 0.25 \times (M_{p\_loaded} + M_{thruster} + M_{press}) - M_{tank} \tag{18web-11}$$

$$= 0.25\ (41.5\ \text{kg} + [4 \times 1\ \text{kg} + 1.6\ \text{kg}] + 0.13\ \text{kg}) - 8.2\ \text{kg} \tag{18web-12}$$

$$= 3.6\ \text{kg} \tag{18web-13}$$

Table 18web-3. Preliminary Propulsion Mass Budget for FireSat II and SCS.

	FireSat II	SCS
Thrusters (kg)	5.6	2.0
Tank (kg)	8.2	2.1
Feed System (kg)	3.6	0.5
Total Dry Mass (kg)	17.4	4.6
Propellant (kg)	41.5	8.2
Pressurant (kg)	0.13	0.028
Total Mass of Propulsion System (kg)	59.03	12.83

The total mass estimate is thus 59.0 kg with propellant and pressurant making up 41.6 kg (71%) of that total. The preliminary mass budget for the propulsion system for FireSat II is shown in Table 18web-3.

18.8.2 Supplemental Communications System

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The initial design for the SCS satellite requires thrusters for both primary propulsion and for unloading of the reaction wheels for the 3-axis stabilized spacecraft. These requirements commonly lead to a propulsion configuration involving four total thrusters (or 8 total thrusters if full redundancy is required). All of the thrusters will be aimed approximately 15 deg off the primary axis (z-axis) in the +x, –x, +y, –y directions. All four will fire simultaneously for a ΔV maneuver while fewer than four thrusters fire together when a rotation is required. A single common propellant and tank will be used in the system. The total mission ΔV (including margin and ullage) has been estimated to be 90 m/s which is relatively low for satellite systems. These early requirements already indicate that thruster simplicity will be an important consideration along with thruster performance (specific impulse). Hydrazine monopropellant thrusters are commonly used for the required roles, but cold gas thrusters should also be evaluated because of their inherent simplicity. The two thruster systems can be evaluated based on their ability to provide the required total mission ΔV without consuming too much of the mass budget (total spacecraft mass of 200 kg).

The first step in choosing the propulsion system is to evaluate the amount of required propellant based on Eq. (18-19). Cold gas thrusters with low leak rate propellants typically have I_{sp} of 45 to 73 sec Table (18-4). For the SCS mission, cold gas thrusters would require 23–37 kg of propellant based on the above I_{sp} range and assuming ΔV = 90 m/s. Hydrazine monopropellant thrusters typically have specific impulses between 215 and 235 sec as in Table 18-5. Applying Eq. (18-19), for the above I_{sp} values, the SCS mission hydrazine thrusters would require between 7.7 and 9.0 kg of propellant. It is unlikely that the projected propellant mass savings for the hydrazine system (roughly 20 kg) could be overcome by the dry mass savings of the cold gas system. With the significant flight heritage of hydrazine monopropellant thrusters they are a good first choice for the SCS system.

Sizing of the propulsion system can continue by choosing appropriately sized thrusters with flight heritage to use as examples. The single thruster module of the MRE-1.0 system can meet the thrust requirement of 4.5N and has a mass of 0.5 kg and an I_{sp} of 218 sec shown in Table 18-5. Equation (18-19) indicates that a mission with a total ΔV of 90 m/s would require 8.2 kg of propellant at a specific impulse of 218 sec. The specified total mission ΔV, 90 m/s, includes 30 m/s for margin and ullage which will account for the unusable fraction of the loaded propellant. The SCS system requires four single thruster modules yielding a total thruster mass of 2.0 kg. To finish the mass estimates we must estimate the mass for the propellant tank and feed system. The mass for the propellant tank can be estimated by using the methods described in Sec. 18.5.1. A simple diaphragm blow-down tank is chosen for this example problem because of its simplicity and the ability of monopropellant thrusters to operate over a wide range in pressures. According to Table (18-8), hydrazine has a density of 0.982 g/cm^3 at an assumed high temperature of 323 K. This yields a required propellant volume of 8.35 liters. Assuming a typical blow-down ratio, B, of 3.5, the initial pressurant volume can be estimated from adding the recommended volume margin of 20% to the pressurant and propellant volume yields a recommended tank volume of 14.0 liters.

The curve fit from Fig. 18-10 then yields a propellant tank mass of 2.1 kg. The mass of the pressurant gas can also be estimated if the initial tank pressure and pressurant gas is known. The maximum operating pressure for the MRE-1.0 monopropellant thruster, 3.9MPa, is a reasonable first estimate for the initial tank pressure and helium can be selected because it is the lightest pressurant gas. Still assuming a temperature of 323 K, the mass of the pressurant is given by

$$V_{ullage\_i} = \frac{V_{p\_usable}}{B-1} = \frac{8.35L}{3.5-1} = 3.34L \qquad \text{(18web-14)}$$

where ρ_{press_i} is the initial density of the He, obtained from the NIST Chemistry WebBook [2010].

$$\begin{aligned} M_{press} &= V_{ullage\_i} \times \rho_{press\_i} \\ &= (3.34 \times 1.2)e-3m^3 \times 6.87kg/m^3 = 0.028kg \end{aligned} \qquad \text{(18web-15)}$$

The general rule of thumb for liquid systems is that tank and feed system represent 10% of the total propulsion system mass (propellant + pressurant + tank(s) + thruster + feed system). In the SCS system, as is common with small scale satellites, the tank and feed system could represent a higher fraction. Assuming that the tank and feed system accounts for 20% of the total mass of the propulsion system, that is,

$$\begin{aligned} M_{tank} + M_{feed} = 0.20 \times (&M_{tank} + M_{feed} + M_{p\_loaded} \\ &+ M_{press} + M_{thruster}) \end{aligned} \qquad \text{(18web-16)}$$

then, the tank (2.1 kg) and feed system (0.5 kg) would have a total mass of 2.6 kg yielding a total propulsion system mass of approximately 12.8 kg. The breakdown of the propulsion system mass budget is given in Sec. 18.8.1. The next step in the design process would be to create a system design including all required components which would yield a schematic similar to the one in Fig. 18-8b.