Politecnico di Milano

`Fourth Assignment`

Space Systems Engineering and Operations

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1 Architecture

The MESSENGER spacecraft is a 3-axis stabilized spacecraft that uses reaction wheels and thrusters for attitude control. The combination of reaction wheels and thrusters allowed MESSENGER to achieve precise attitude control throughout its mission. The reaction wheels provided continuous and precise control over the spacecraft's attitude, while the thrusters offered additional maneuvering capabilities when larger attitude changes or orbital adjustments were required.

It's important to note that MESSENGER's attitude control system was a complex and integrated subsystem, incorporating various sensors, algorithms, and control mechanisms to ensure accurate and stable attitude control. The actuators, such as the reaction wheels and thrusters, were key components of this system, working in coordination with the spacecraft's guidance and control algorithms to maintain the desired orientation and stability in space. The set of sensors consists of star trackers, digital Sun sensors, and an inertial measurement unit. Solar panels provide electric power to the spacecraft, and a heat-resistant and reflective sunshield colored in white protects the spacecraft from the extreme thermal conditions encountered close to the Sun at a distance of less than 0.85 AU. The spacecraft body axes and selected component locations are shown in the following figure.

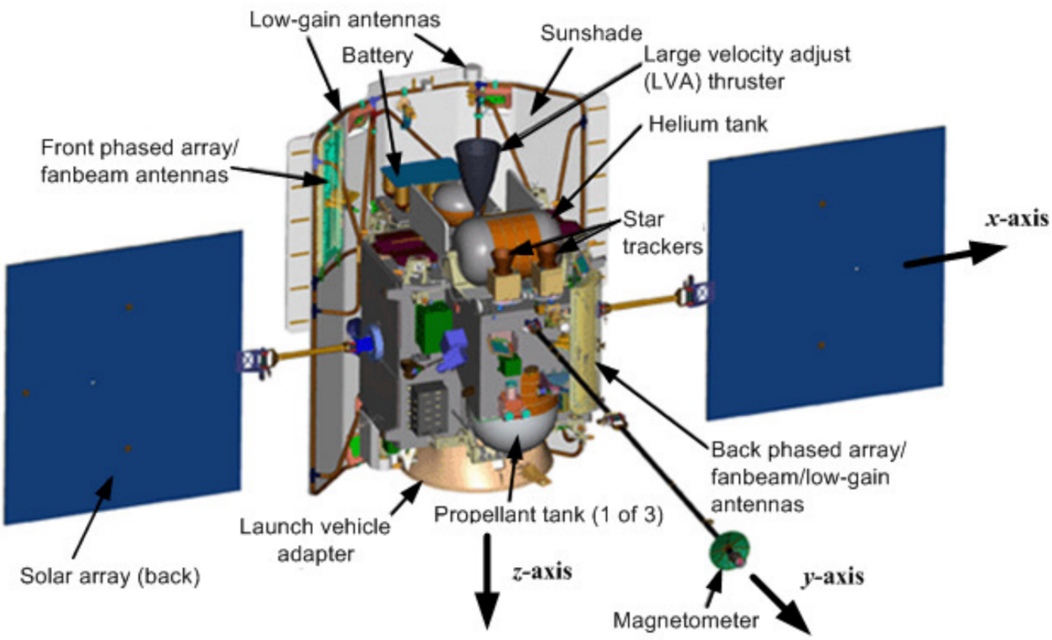


Figure 1: Messenger's Components and body frame

The MESSENGER guidance and control subsystem maintains spacecraft attitude and executes propulsive maneuvers for spacecraft trajectory control. Software algorithms run in the Main Processor (MP) to coordinate data processing and commanding of sensors and actuators to maintain a 3-axis stabilized spacecraft and to implement desired velocity changes.

The system enforces two attitude safety constraints :

- Sun keep-in (SKI) constraint that keeps the sunshade pointed towards the Sun to protect the spacecraft bus from extreme heat and radiation. Maintaining that specific sunshade orientation for the entire orbital mission poses an important constraint to the spacecraft's attitude.
- The hot-pole keep-out constraint protects components on the top deck of the spacecraft from additional thermal extreme temperatures due to re-radiation of sunlight from the surface of Mercury once in orbit.

1.1 Sensors

The sensor set consists of star trackers, an inertial measurement unit, and sun sensors. Inertial attitude reference is provided by two autonomous star trackers from Galileo Avionica, both of which are mounted on the top deck looking out along the –Z axis. They are placed in that specific location in order to avoid the Sun in its Field Of View. They have a total mass of 6.37 kg, including baffles. Each uses a maximum of 12.3

W in the normal tracking mode. The trackers do image processing to identify star patterns internally, and the attitude solution in the form of quaternion and rate is output to the flight software.

MESSENGER also carries a set of Adcole Digital Sun sensors (DSS) to provide Sun-relative attitude knowledge if there is a failure in the primary attitude sensors. Two separate Sun sensor systems consist of a DSS electronics box connected to three DSS heads, two of which are located on opposite corners of the sunshade and one on the back of the spacecraft.

	Name	FOV **[º]**	**Number**
Star Tracker	Galileo Avionica A-STR	± 16.4	2
Sun Sensor	Adcole Maryland Aerospace	± 64	3

Table 1: *Reaction Wheels*

Spacecraft rotation rates and translational accelerations are provided by a Northrop-Grumman Scalable Space Inertial Reference Unit IMU with four hemispherical resonance gyroscopes and Honeywell QA-3000 accelerometers. The IMU has two power supply and processor boards providing internal redundancy. Typically, one processor board and all four gyros are powered at all times, while the four accelerometers are powered only when performing a trajectory correction maneuver (TCM). Spacecraft attitude is estimated by the MP software using a Kalman filter algorithm to combine star tracker and gyro measurements. A simpler filter is used to estimate accumulated velocity change from accelerometer measurements when executing TCMs.

This set of sensors ensures system's constancy: the Sun sensors alone could not perform attitude determination during the eclipse phases, while the IMU suffers from accumulated errors over time. Star sensors provide fine attitude determination, but are expensive in terms of operations on board for data processing. During nominal operations, attitude determination is achieved through one star tracker, and four gyroscopes. Using one star tracker and four gyroscopes can provide redundancy and help improve the overall reliability of the attitude determination system. The star tracker can provide accurate attitude measurements based on star observations, while the gyroscopes can provide continuous rate information. By combining the measurements from both sensors, it is possible to estimate the spacecraft's attitude with higher accuracy and robustness. Additional sensors such as sun sensors and additional star trackers can be used to enhance the accuracy and reliability of the attitude determination system and of course, they are essential in case of failure.

1.2 Actuators

The actuator suite consists of reaction wheels and thrusters.

	Name	T_{max} **[Nm]**	h_{max} **[Nms]**	**Number**
Reaction Wheels	Teldix RSI 7-75/601	0.075	7.5	4

Table 2: *Reaction Wheels*

The 4 reaction wheels are set in a *pyramid* configuration. This design enables redundancy to the spacecraft to be fully controllable even with the failure of one of the wheels. During nominal operations, attitude control is achieved through the combination of the four reaction wheels.

Thrusters	**Thrust Max [N]**	I_{sp} **[s]**	**Type**	**Number**
LVA	667	316	Bi-Propellant	1
C	22	230	Monopropellant	4
A,B,S,P	4.4	220	Monopropellant	12

Table 3: *Thrusters*

The set of 17 thrusters includes three different thruster types, with respect to the force they shall provide to the spacecraft. The **LVA** (Large Velocity Adjuster) thruster is the bipropellant portion of the system and provides a thrust level of 667 N and a specific impulse of 316 s, in the -Z direction; it is the most powerful engine of the spacecraft, grating major Δv changes. Four monopropellant thrusters disposed in the -Z side (identified as **C**-thrusters) provide a force of 22 N and a specific impulse of 230 s each. Finally, twelve monopropellant engines grant a force of 4.4 N and a specific impulse of 220 s each. More specifically, eight thrusters (**A** in the +X side and **B** in the -X side) are disposed in double canted sets of four for redundant attitude control on the three axis, in modules together with the C-thrusters, even though they yield force in different directions. The remaining four motors (**S** in the +Y side and **P** in the -Y side) are meant to provide velocity changes in the sunward direction (S-thrusters) or in the away-from-the-sun direction (P-thrusters). The spatial configuration of the A, B, S, and P motors ensures that no maneuver requires a null command, limiting the waste of propellant. Furthermore, it is possible to split the required thrust between opposing thrusters on opposite sides of the spacecraft in order to cancel out any residual force. That allows to have a finer attitude control and to use the minimum number of burns to complete a specific task.

2 Pointing Budget for Control

2.1 Control Modes

From a Guidance and Control perspective, flight operations can best be described as the interaction of three spacecraft operational modes with a set of four primary activities. When all systems are performing nominally, the spacecraft is in its "operational" (OP) mode. Demotion to one of two safe modes – safe-hold (SH) or Earth acquisition (EA) - occurs autonomously in response to certain faults or by ground command. Promotion from either of the two safing modes to OP mode can occur only via ground command. The four G&C activities are maintaining spacecraft attitude, managing spacecraft momentum, executing TCMs, and pointing the two solar panels.

OP mode is the normal mode for science observations and engineering activities. All varieties of the four primary G&C activities can be performed in this mode. Spacecraft attitude is altered by command as needed to point antennas at the Earth, point instruments at various science targets, or align thrusters with a target ΔV direction. A wide variety of pointing options are available for pointing in inertial directions, to various celestial bodies, and to locations on one of these bodies. Scan patterns combining periods of fixed rate rotations about specified axes with pauses can be added to the base pointing option. One star tracker and four gyroscopes are used during nominal operations, while the other sensors ensure reliability of the measurements and redundancy.

The spacecraft enters SH mode when a fault of intermediate criticality is detected. Commanded execution of TCMs or momentum dumps is prohibited in this mode. Spacecraft attitude is restricted to the "downlink" attitude, which aligns a specified body axis with the Sun line, in order to satisfy the SKI constraint, and places the Earth line in one of the quadrants of the XY plane covered by one of the two antenna sets, permitting communication with Earth. If the HPKO constraint is enabled, the attitude will be adjusted away from the downlink attitude when necessary to keep the top deck pointed away from the planet's surface. Once the spacecraft passes out of the defined hot-pole region, normal downlink pointing is automatically reestablished. On entry to SH mode, solar panel control is set to the fixed Sun offset mode using a specially designated value for the size of the offset angle. The offset angle value can be changed or one of the other two control modes (panels in a fixed angle in the body frame or sun offset mode with temperature offset) can be invoked by ground command once in SH mode. Despite SH mode being a non-nominal condition, there are no other severe constraints on the sensors, apart from the ones demanded from attitude's requirements.

The spacecraft goes into its lowest level safing mode - EA mode - in response to faults of highest severity. As in SH mode, execution of TCMs or momentum dumps is prohibited, and commands to change to one of the other pointing options are ignored. A specific "Sun-relative rotisserie" attitude is automatically implemented. The rotisserie attitude points a specified spacecraft body axis at the Sun and rotates the spacecraft about this axis at a fixed rate. The nominal EA Sun line is either the +Y or −Y axis, depending on spacecraft range from the Sun, and the rotation rate is 0.0005 rad/s, taking 3.5 hours to complete a single revolution. This rate and the axis can be altered by command. While star tracker data are used if available, the EA attitude can be achieved using only Sun sensor and gyro rate measurements. If both star tracker and Sun sensor data are lost, the system switches to a Sun search routine that performs a series of rotations about each of the body axes until Sun direction information is restored. On entry to EA mode, solar panel control is set to the body-fixed angle mode using a specially designated value for the position angle. The body-fixed angle value can be changed or one of the other two control modes can be invoked by ground command once in EA mode.

In any of the three modes, the G&C system monitors system momentum and will initiate a momentum dump using thrusters when momentum magnitude exceeds limits that could compromise controllability using the wheels [2].

2.2 Pointing budget

When in operational mode the pointing budget is dictated by the accuracy requirements of the scientific instruments on board. The pointing precision needs to be at least 0.1° and the pointing knowledge at least 0.02° considering the most requiring instrument [3]. The requirements are satisfied by using star trackers (accuracy ranging from 0.0003° to 0.01° [4]) together with reaction wheels. In the other two control modes, the pointing accuracy is the one needed to point the antennas at Earth and the Sun shade and solar panels at the Sun . The Sun shade axis has to stay within 12° from the Sun line, the antennas have a 12° wide beam and the solar panels don't need a high precision pointing, so using the star trackers and Sun sensors, together with reaction wheels provides enough accuracy [2].

3 Disturbances

3.1 External Disturbances

Counter acting the the disturbances coming from the environment is crucial in order to pursue the scientific objectives of the mission and to perform the attitude control's operations; a preliminary estimation of the predominant torques acting on the spacecraft during its mission is performed. The most relevant phases are the fly-by performed around Earth, Venus, and Mercury. Since for Venus and Mercury multiple fly-bys are performed, the ones taken into account are those with the closest approach to the surface.

	Earth fly-by	Venus fly-by	Mercury fly-by
Gravity gradient [Nm]	$2.10 \cdot 10^{-4}$	$3.85 \cdot 10^{-4}$	$3.43 \cdot 10^{-4}$
Solar radiation pressure [Nm]	$1.50 \cdot 10^{-6}$	$2.88 \cdot 10^{-6}$	$9.87 \cdot 10^{-6}$
Atmospheric drag [Nm]	$\simeq 0$	$\simeq 0$	$\simeq 0$
Magnetic field moment [Nm]	$4.79 \cdot 10^{-4}$	$\simeq 0$	$1.30 \cdot 10^{-5}$

Table 4: Worst-case external disturbance torques

Thanks to the results provided by the NASA's missions MESSENGER and MARINER 10, which provided informations about the intensity and the direction of the magnetic field of Mercury, it is has been possible to provide an estimate of the troque related to its presence. Anyway, studying the magnetic field of Mercury is one of the scientific goals of MESSENGER: before the launch, the only data on Mercury's magnetic field were the ones provided by MARINER 10, which covered a very limited part of the planet and computed a low amount of data. Because of that, the torque induced by the external magnetic field has not been taken into account.

3.2 Internal Disturbances

A possible internal disturbance all along the mission is propellant mass distribution that could change the position of the center of mass. The way this problem is handled is by emptying both fuel tanks simultaneously so that symmetry is maintained. Fluid sloshing could potentially be a relevant disturbance, but it is minimized by adding baffles inside the propellant tanks. All sorts of misalignments and uncertainties around the center of mass are considered by positioning it a few centimeters away from it's real position. Moreover, structural dynamics due to flexibility of some components (especially the appendages) and thermal shocks, given by the rapidly changing distance from the hot surface of Mercury, have to be considered.

4 Reverse Sizing

4.1 Subsystem mass sizing

In Table (5) are shown the masses for the different components of the ADCS subsystem according to the literature.

–	Quantity	Mass [kg]
Reaction wheels	4	15
Star trackers	2	5
Sun sensors	2	2.6
Inertial measurement unit	1	5.6
Total		28.2

Table 5: Mass sizing

The total mass obtained is very similar to the real one of about 27 kg [5].

4.2 Reaction wheels sizing

The reaction wheels provide attitude control to the spacecraft by counteracting the external disturbances and performing slew maneuvers. The previous analysis shows that the amount of external disturbances is considerably high when the spacecraft is in proximity of the Earth, however having taken in consideration both the importance of the orbital phase around Mercury for the accomplishment of the scientific goals of the mission and the time duration of this phase, the sizing of the actuators is estimated taking into account the torques the spacecraft undergoes to in Mercury's environment.

4.2.1 Station keeping

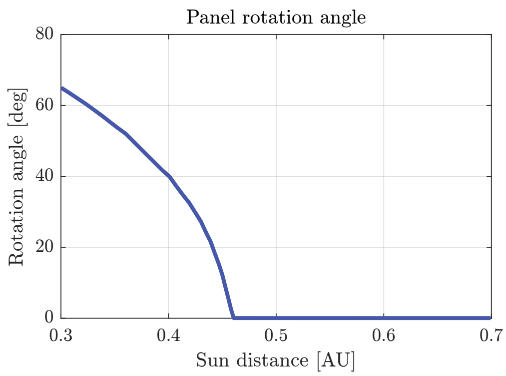
The total torque has been estimated taking in consideration the two main disturbances acting on the spacecraft: the solar radiation pressure and the gravity gradient at a given distance R from the main attractor Mercury.

$$T_{GG} = \frac{3\mu_{pl}}{2R^3}\left(I_{Max} - I_{Min}\right) \cdot sin\left(\theta\right) \quad \text{Gravity gradient} \tag{1}$$

$$T_{SRP} = \frac{F_S}{c}\left(2A_{pn}\left(1+q_{pn}\right)cos\left(\alpha\right)\Delta x_{pn} + A_{ss}\left(1+q_{ss}\right)\Delta x_{ss}\right) \quad \text{Solar Radiation Pressure} \tag{2}$$

where the subscript pn refers to the solar panels and the subscript ss refers to the sunshade. Moreover notice that:

- F_S is the solar energy flux density
- θ is the orbit-averaged deviation of the z-axis from the local vertical
- α is the angle between the solar panel and the sun direction
- Δx is the distance between the center of solar pressure and the spacecraft center of mass
- q is the reflectivity coefficient of the body



Since the orbit of Mercury periapsis is equal to 0.31 AU an its apoapsis is 0.47 AU the value of α has been considered to be changing accordingly.
By considering an averaged value of the Gravity Gradient along the orbit around Mercury during the first part of the scientific phase (i.e. 12 h, 200 $\times$ 15200 km orbit), it is possible to compute the total angular moment absorbed by the spacecraft leading to the following results.

Single RW storage [Nms]	RW number [-]	Computed CMD [-]	Real CMD [-]
7.5	4	64	41

Table 6: Commanded Moment Dump manoeuvres

The number of momentum dumps calculated is higher than the one actually performed by the spacecraft, as MESSENGER's solar panels were cleverly used in order to generate an SRP torque that was capable of

reducing the accumulation of momentum in the reaction wheels.
One further detail that was not modeled in our calculation is the fact that the orbit-averaged effect of the gravity gradient torque is much lower than the mere integral of the gravity gradient along one complete orbit: this is due to the fact that some of the momentum stored during half of the orbit can be used to counteract the torque perceived by the spacecraft during the other half of the orbit.
This computation is therefore to be intended as a worst case scenario, where all of the applied torque is counteracted by the reaction wheels.

4.2.2 Slew manoeuvres

Given the fact that MESSENGER mostly held a inertial sun pointing attitude due to thermal constraints, the main slew maneuvers performed where 3 *Flip maneuvers* and 2 *Flop maneuvers* when crossing the 0.95 AU distance from the Sun, to orient respectively the sun shield *towards* and *away from* the Sun.
Considering a slew manoeuvre, with constant acceleration and breaking, the torque needed by the reaction wheels is:

$$T = 4\theta_{max}\frac{I_{max}}{t_m^2} \tag{3}$$

where:

- $\theta_{max} = 180$ is the maximum slew angle possibly needed around the major inertia axis;
- $I_{max} = 467\ kgm^2$ is the major inertia moment;
- $t_m = \dfrac{\theta_{max}}{\dot{\theta}_{max}}$ is the minimum slew maneouvre time, considering the acceleration as $\dot{\theta}_{max} = 0.5°/s$;

The resulting torque needed for this type of maneuver is $T = 0.045N$, which is under the maximum torque which a reaction wheel can provide.

4.3 Thruster sizing

Reaction Wheels desaturation can be performed passively by changing the solar panels orientation to exploit the solar radiation pressure or actively by firing a set of eight thrusters (A and B sets). Anyway, due to the frequency of scientific observations during the orbital phase around Mercury the passive dumping is no more sufficient; a sequence of commanded moment dumps (CMD)

is scheduled in order to avoid saturation of the reaction wheels [1].
Firstly, the minimum time required to desaturate is computed:

$$t_{min} = \frac{h_{max}}{n_{th}LFmax} \tag{4}$$

where:

- $h_{max} = 7.5\ Nms$ is the maximum angular momentum storable by one wheel;
- $n_{th} = 2$ is the number of thrusters responsible for each one of the four reaction wheels;
- $L = 1.12\ m$ is the arm of the torque;
- $F_{max} = 4.4\ N$ is the maximum force which each thruster can provide

This computation results in $t_{min} = 0.76\ s$, which is compatible with the minimum burning time of the 4.4 N thrusters used for this application. However, the reaction wheels cannot despin fast enough to keep up.
The burn is therefore divided into multiple smaller impulses that take place over a time window of roughly one minute[6].

4.4 Mass budget

It is possible to compute the propellant mass by combining the duration of each momentum dump manoeuvre and the total number of dumps required according with the following equation:

$$m_p = n_{dumps} \cdot \frac{n_{wh}t_bF_{th}}{I_{sp}g_0} \tag{5}$$

where:

- $I_{sp} = 220\ s$ is the specific impulse of the thrusters;
- $n_{dumps} = 181$ is the total number of momentum dumps;
- $n_{wh} = 4$ is the number of reaction wheels;
- $t_b = t_{min}$ is the burning time;
- $F_{th} = F_{max}$ is the force applied by one thruster.

This results in a propellant mass needed of $1.12\ kg$.

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