Politecnico di Milano

Seventh Assignment

Space Systems Engineering and Operations

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Contents

1 Configuration

1.1 Space Segment

The MESSENGER mission was launched on 3 August 2004 on a Delta II launch vehicle form Cape Canaveral. The size of the spacecraft is dictated by the need to keep the mass low enough to be inserted into a energetic enough orbit, while the dimensions are limited by the fairing size [1].
The very first configuration that is considered is the "*folded*" configuration. The spacecraft is folded to fit inside the launcher's fairing and will need to withstand the harshest mechanical environment during launch. The mechanical environment can be described across several loads that must be considered for a preliminary design:

- Static and dynamic loads
- Payload fairing separation
- Stage separation
- Random vibrations
- Acoustic loads
- Shocks

The spacecraft is protected from atmospheric heating and loads by the fairing which also has some constraints over the payload. In fact, the whole structure of the "*folded*" payload shall be contained within the fairing dynamic envelope which protects the spacecraft from launch vibrations with sufficient margin.
In figure 1 is displayed the folded spacecraft, during the integration with the fairing. Below it the third stage kick motor is visible.
The spacecraft is attached to the rocket by an attach fitting, located at the center of the -Z face (see figure 3), which transfers the aforementioned loads between the two. Numerical accurate simulations and vibration tests were done to ensure the spacecraft's safety along the launch.
The only difference between the folded and unfolded configuration can be found in the solar arrays and the extending boom for the magnetometer, which are retracted to fit inside the allotted space inside the fairing of the Delta-II: the two solar arrays are folded against the body of the spacecraft through a single hinge for each panel, with a positive stop to prevent them form touching the spacecraft's body, while the 3.6 m arm is stowed through two hinges (see figure 2).
Deployment of the solar arrays and the magnetometer boom happens soon after the injection in the escape trajectory to allow for testing and validation of the components.

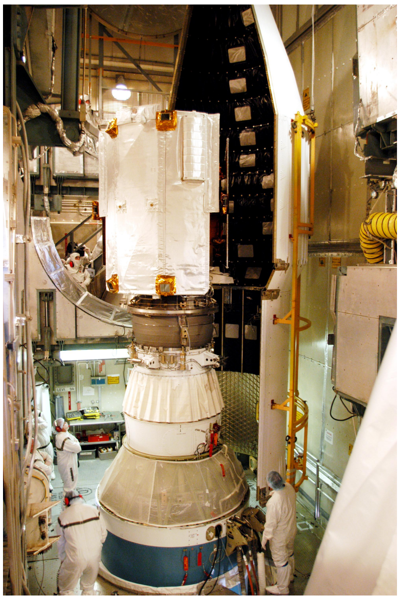

Figure 1: "*folded*" configuration during integration in the fairing

Figure 2: Folded configuration

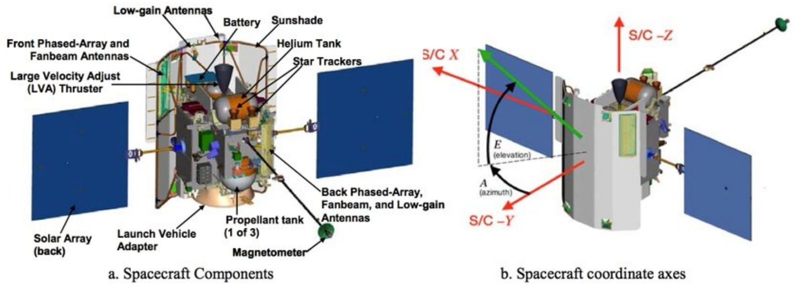

Figure 3: Unfolded configuration

The following sections are based on literature ([2], [3], [4]) and a critical analysis of the spacecraft configuration.

1.1.1 Thermal subsystem

The first design choice was the orientation of the Sunshade towards the sun along the flight to withstand the extreme temperatures. Therefore, the choice of positioning it in the -Y face was the driver for the system configuration.
The two main radiators are mounted flush on the +X and -X faces, the largest available ones, of the body in

a symmetric way to avoid center of mass (CoM) imbalance. This position allows them to have a clear view at empty space, without ever being pointed to the sun.

1.1.2 Power subsystem

The Solar Arrays are in the classic symmetrical configuration to assure symmetry in weight distribution and SRP torque effect. The faces selected for the design are +X/-X, with poles that extend the panels outside the shade of the sunshield.
The battery is located on the -Y face of the spacecraft, closest to the sunshield, to keep it in a location where it could operate for the whole mission in a range between $-10°C$ and $0°C$, with only minor excursions above this range. This position has the advantage of being shielded form Mercury's albedo and IR emissions by the spacecraft body for most of the orbits.
The power system electronics (PSE) and power distribution unit (PDU), solar array junction box (SAJB) as well as the two solar array drive assemblies (SADAs), are located near the base of the poles of the solar panels in order to minimize wiring and in a way that balances out the mass distribution; this also allows the heat generated by the high power electronic components to be easily transferred to the radiators, located nearby.

1.1.3 Propulsion subsystem

In order to avoid ejecting exhaust gasses into the sunshade, the solar panels or the scientific instruments, the only direction to place the powerful 667 N LVA thruster and the four 22 N thrusters is along the Z axis. The choice fell onto the -Z face as the +Z is occupied by the launch vehicle adapter in a position aligned with the CoM of the spacecraft in order to minimize the torque induced by firing the engines, and thus the need for correction burns.
The rest of the 12 other monopropellant 4 N thrusters are placed in the +X/-X and +Y/-Y directions to be able to fully control the attitude of the spacecraft along each axis, each direction and with an almost zero net force. Their location maximizes the lever arm while reducing exhaust impingment on other spacecraft components.
The three main propellant tanks are located at the very center of the structure, oriented along the Z axis and placed side by side in the Y direction. The single oxidizer tank in in the middle, flanked by the two fuel tanks. This configuration allows for low CoM excursions independently of the level of propellant in the tanks, and is close to the center of the spacecraft to reduce its inertia moments.
The auxiliary fuel tank is located besides the battery pack, on the -Z face on its -Y edge.
The pressurizing helium tank is located on the -Z face close to its +Y edge to balance out the mass of the battery and the auxiliary fuel tank.
The whole propulsion system is nearly perfectly symmetrical with respect to the Y-Z plane.

1.1.4 Telecommunications subsystem

The main aspects of this mission under the TTMTC viewpoint, were the quantity of data to be sent to Earth and the distances reached by the spacecraft. Another peculiar aspect of the design not to be underestimated was the fact that, going inwards in the solar system, the position of the Earth is not fixed in a particular direction: the planet could actually be located in every direction with respect to the spacecraft. Also, it is needed to account for the constraints added by thermal management, mainly connected to the need to always orient the spacecraft in a way to have the shade in the Sun direction.
Hence, the design choices were driven towards a configuration capable of uplink and downlink with Earth without the necessity to orient neither the spacecraft body nor the antennas in a particular direction.
The two identical lightweight, High-gain, Waveguide-Based Phased Array Antennas (PAAs) and two Medium-Gain Fanbeam Antennas (MGA), are conveniently mounted on opposite edges of the spacecraft, in particular the first couple comprehending one PAA and one MGA, is mounted on the front side of the shade corresponding to -X and -Y axis, instead the other one correspond to +X and +Y axis. The four hemispherical Low-Gain Antennas (LGA) provide coverage in all directions, without the necessity of rotating the spacecraft. Three are mounted on the sunshade, one on the face looking in the -Y direction, the other two on the edges of the sunshade, looking in the directions +Z and -Z; the fourth one is located in the +Y face, looking in the +Y direction.
This design grants not only pointing capabilities by rotating around the spacecraft-Sun line, but also a balanced mass distribution.
The two solid-state power amplifiers are located near each phased array - fanbeam antennas couple to minimize losses.

1.1.5 ADCS subsystem

Four Sun sensors are mounted in the four corners of the sunshield, on the inside face, with the heads of the sensor poking through the shield and with direct line of sight to the Sun. The curvature of the shield allows for differential measures. This is the only body-fixed location available for these sensors, as the only other part of the spacecraft that receives sunlight in the inner cruise phase are the solar arrays, but they can rotate.
Two further Sun sensors are located on the center and +X,-Z corner of the +Y face of the spacecraft, to allow for Sun visibility during the outer cruise.
Two Star trackers are positioned side by side, symmetric with respect to the center line, on the +Y edge of the -Z face, pointed towards the -Z direction. This location was chosen as the Sun will never shine directly in the sensing elements of the star trackers during nominal operations, which may lead to unrecoverable damage.
The inertial measurement unit (IMU), its four gyroscopes and four accelerometers are located under the -X face, close to the spacecraft's CoM to aid precision.
The four reaction wheels are placed in a pyramid configuration. They are located on the -Y face, inside the sunshield, in two pairs that flank the fuel tank: this location was chosen for symmetry reasons with respect to the Y-Z plane and at an X location coincident with the CoM. An added benefit is that not all the torque is applied at the same point on the structure.

1.1.6 Scientific Instruments

The magnetometer MAG needs to be protected by the sunshade but also needs to be far from the possible electromagnetic fields generated close to the spacecraft to operate correctly. Therefore, the only possible direction respecting those requirements is +Y, at the end of a 3.6 m boom (see figure 4).
A suite of five "Nadir" pointing instruments (MDIS, MASCS, XRS, MLA, GRS) all require pointing in the direction of Mercury's surface to operate, hence they were all located on the same face and pointed in the same direction that allows the spacecraft to orient itself in a position that would be suitable for all of them at the same time. The location chosen is on the +Z face, to offset the weight of the LVA, battery and tanks on the opposite face. They are located inside the launch vehicle adapter, which itself is aligned with the CoM, to aid symmetric mass distribution, low inertia and a more efficient use of the space (see figure 5).
The NS doesn't have FOV limitations, and as such is mounted on the -X,-Z corner of the +Y face due to proximity to the data processing unit (DPU).
The EPPS instrument is mounted on the side of the spacecraft, near the top deck (-Z face), and close to the edge of the shade from the sunshield: this allows the instrument to measure ions from Mercury's surface, its magnetosphere, and from the solar wind based on spacecraft orientation.
The FIPS allows for near hemispherical coverage, so positioning is less critical for this instrument. The location chosen was the -X face.

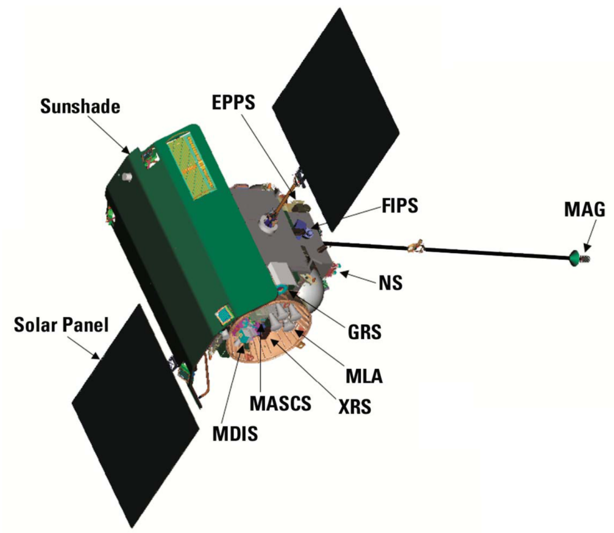

Figure 4: Instrument location

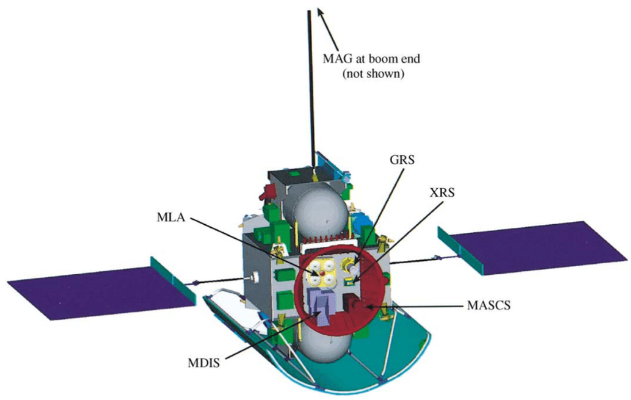

Figure 5: Focus on instruments on the +Z face

2 On-Board Data Handling

The On Board Data Handling subsystem can be split in different segments: the flight computer architecture, the GNC flight software, and the on-board file storage system; each of them will be analyzed in the following section. The subsystem design has been strongly affected by the mission profile that involves both deep space and near-Sun environment: this imposes constraints including the limited on-board storage, low bandwidth, long light-time delays and intermittent connectivity with Earth [5].

2.1 Architecture

2.1.1 Flight computer architecture

MESSENGER is equipped with two sets of flight computers, each of them containing a main processor (MP) operating at 25 MHz and one fault protection processor (FPP) operating at 10 MHz, both of them are *RAD6000* processors with 8 MB of RAM, multiple times space-flight proven. Each couple of MP and FPP, together with other elements, is packaged in an integrated electronics module (IEM) able to provide command and telecommunication functions.
The complete flight software runs in only one active MP performing all nominal operations while both the FPPs monitor the spacecraft safety, the second MP is instead kept unpowered and serves as a backup unit. The following diagram shows the communication links between the flight computers and the GNC hardware components. The databus *MIL-STD-1553* is a military standard federated databus originally developed for avionic systems but widely used also in spacecrafts.

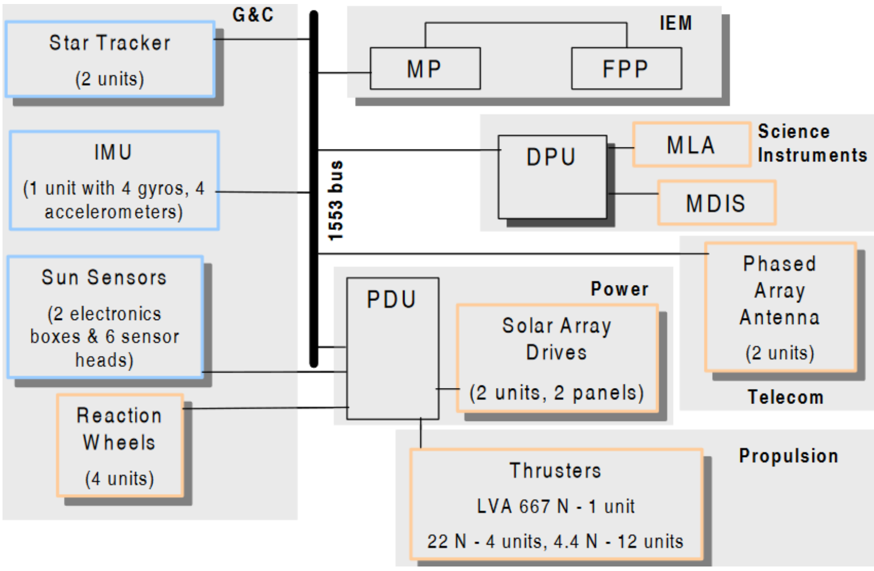

Figure 6: On board computer system diagram

The active MP interfaces with the star trackers, the IMU, a redundant Power Distribution Unit (PDU), two Data Processing Units (DPUs) and the two FFPs directly with the 1553 data bus. The PDU interfaces with the Sun sensors, reaction wheels and the propulsion system. The DPU interfaces with all science instruments processor. Both FPPs can access the Sun sensor data and using the GNC software provide an independent Sun direction computation. The Main processor communicates also via peripheral component interconnected (PCI) to a solid state recorded SSR with 8 *Gb* of RAM [2] [5].

2.1.2 Flight software architecture

MESSENGER on-board software is written in *C* language and is operated under the *VxWorks 5.3.1* real time operating system. The GNC software is composed by six main functional blocks: attitude determination, attitude control, guidance, momentum management, propulsion operations and solar panel control. The GNC Main Processor software functions are split into two main tasks that run at 1 Hz or 50 Hz. Functions necessary for immediate attitude and trajectory control run at 50 Hz and are streamlined to run as efficiently as possible. Attitude determination, solar panel control and all other functions (e.g. phased array antenna steering and MDIS and MLA interfaces) are contained in the 1 Hz tasks [2] [6].

2.1.3 File storage system

MESSENGER uses an on-board file system for data collection based on an 8 *Gb* Solid State Recorder (SSR). The choice of this sort of system has been driven from two main factors [5]:

- Severely limited downlink capability, because of the combination of available downlink rates, the amount of downlink time and the long round-trip light time. This aspect enforces a constraint on the information stored on-board which needs to be prioritized by importance: in this way it is possible to avoid transmitting in case the bandwidth is not available. A file based system provides simplicity in terms of storing and managing data in terms of priority, making possible to delete the less important data and free space on the SSR;
- Most of science data stored on the SSR are images which come from the Mercury Dual Imaging System (MDIS), compressed and downlinked as individual files by the flight software; this kind of system allows to define variable length images and to compress them using several algorithm options before storing back to the SSR.

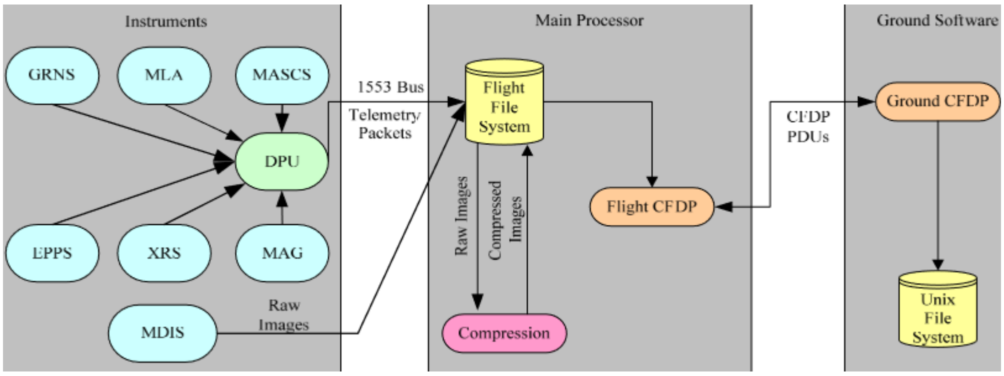

Figure 7: Scientific data storage and transfer structure [6]

The flight software has the capability to convert all the telemetry packets, which consist into data concerning the spacecraft activity and science operations, into files [5]. The same goes also for the images from the MDIS instrument, that are stored via a Direct Memory Access (DMA); at a certain point, the software compresses the images through Integer Wavelet Transform (IWT), performs jailbarring of the data and then subframing. IWT is similar to the JPEG compression, allowing for different levels of compression; jailbarring compression operates by removing columns of data and reducing the image; subframing enables to select specific parts of the image for the compression procedure.
The MP can list all the contents in a directory; once a file is completed and ready for downlinking, it is deposited in one of ten directories, each one ranked by priority as it is displayed in Fig 8:

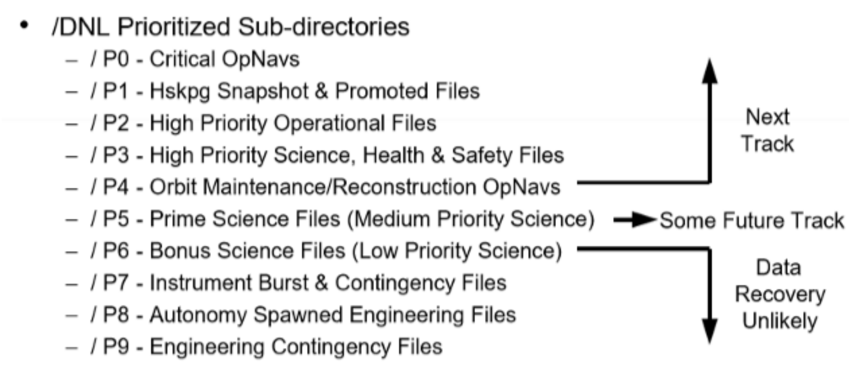

Figure 8: Directory structure

The files can either be downlinked manually or automatically thanks to the SSR's playback capability: it consists in periodically scanning from P0 to P9, determining if a file exists in any of the directories. After that, a playback list is created, with all the files to downlink in it. Priority ranking and scanning's frequency can change depending on the mission operations, and the presence of a top priority level file could also interrupt the process to ensure that it is downlinked as soon as possible, if necessary.

2.2 Reverse Sizing

The first step to size the OBDH system is to list all application functions allocated to the computer. This is done in table (1) were the first part contains the data directly obtained from literature [**Vaughan2005**], while the second one reports the ones deduced from the analysis of the mission architecture.

Device	**Measurement or command**	**Acquisition frequency**
IMU	Gyro integrated angular rate	1 Hz and 50 Hz
	Accelerometer integrated linear velocity	50 Hz
	Diagnostic data	1 Hz
Star tracker	Quaternion and rate	1 Hz
	Rate only	50 Hz
	Diagnostic data	1 Hz
Sun sensor	Head ID and Sun aspect angles	1 Hz
SADA	Reference and potentiometer voltages	1 Hz
	Step commands	1 Hz
Reaction wheel	Tachometer time pulse information	50 Hz
	Wheel torque commands	50 Hz
Thrusters	On/off commands	50 Hz
Latch valves	Open/close commands commands	1 Hz
Heaters	On/off commands	1 Hz
Phased-array antennas	Steering commands	1 Hz

Device	**Measurement or command**	**Acquisition frequency**
Instruments	Pointing and data logging	1 Hz
Attitude control	Complete attitude control software	50 Hz
Attitude determination	Complete attitude determination software	1 Hz
Orbit propagation	Orbit propagation software	1 Hz
Complex ephemerides	Planets and stars ephemerides	1 Hz
Engine control	LVA control software	50 Hz
Pressure sensors	Read pressure transducer data	1 Hz
Uplink	Receive and decode uplink data	1 Hz
Downlink	Encode and transmit downlink data	1 Hz
Operating system	Low-level on board software	50 Hz

Table 1: Acquisition frequencies of the functions allocated to the MP

Notice that in general only a subset of the previous functions will be executed simultaneously for each flight mode. Nevertheless this analysis considers a worst-case scenario thus the sizing condition accounts for all of the aforementioned functions executed at the same time.
Knowing the acquisition frequency for a given function, the throughput is computed *by-similarity*:

$$KIPS = \frac{KIPS_{typ} \cdot f}{f_{typ}} \tag{1}$$

The number of code and data words per function, instead, doesn't need to be adjusted according to the frequency. The total throughput, code and data are obtained by summing the values required by each function.

Component	Number	Code (words)	Data (words)	Throughput (KIPS)
IMU	1	800	500	90
Star tracker	2	2000	15000	200
Sun sensor	2	500	100	1
SADA	2 x 2 *funct*	2000	1000	1
Reaction wheel	4 x 2 *funct*	1000	300	125
Thrusters	16	600	400	30
Latch valves	9	800	1500	30
Heaters	17	800	1500	30
Phased array antennas	2	2000	1000	1
Instruments	7	2000	6500	1
Attitude control	1	24000	4200	300
Attitude determination	2	15000	3500	15
Orbit propagation	1	13000	4000	20
Complex ephemerides	1	3500	2500	8
Engine control	1	1200	1500	50
Pressure sensors	5	800	1500	30
Uplink	1	1000	4000	0.7
Downlink	1	1000	2500	0.3
Software	1	15400	7300	180
Total		163300	170500	3504
Total margined		816500	852500	17520

Table 2: Code words, data words and KIPS per component

It is important to note that the values in table (2) are typical values, that in reality could vary significantly, therefore a margin of 400 % has been included in the calculations.
Transforming data and code words in Kb (considering 32 bit words) it is then possible to calculate the ROM and RAM size:

$$ROM = CODE[Kb] \qquad RAM = CODE[Kb] + DATA[Kb] \tag{2}$$

The results shown in table (3) considers the worst case operative condition that occurs in the orbital phase where all the subsystems are considered simultaneously active except for the main engine control. The ROM is nevertheless sized to account for the complete code while the RAM and throughput are sized according to the aforementioned condition.

	ROM [Mb]	RAM [Mb]	Throughput [MIPS]
Reverse sizing	3.24	6.62	17.27
Real processor	4	8	20

Table 3: Memory and performance sizing results [5] [7]

The *RAD6000* processor chosen for the mission satisfies both memory and CPU needs.

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