# Design Log

## Requirements

### Orbiter

Pointing direction:

* For communications the orbiter must be nadir pointing, except during probe deployment. Max deviation from nadir = 42.4deg **Driving requirement**
* Earth-pointing comms will have own gimbal mechanism for increased accuracy and to allow independent pointing direction
* The EPS must therefore be equipped with a gimbal mechanism to allow pointing towards the earth throughout the orbit.

Pointing Accuracy:

* EPS: max 23 degree incidence angle. Will be outfitted with gimbal mechanism, so not a driving req for AOCS
* Telecomms: For high-gain antenna 0.1-0.5 degree accuracy minimum. HGA will utilize separate gimbal for higher accuracy. **Driving requirement**

Maneuver Requirements:

* No need for high-rate maneuvers. Nominal rates will suffice (0.05 deg/s – 0.5 deg/s)
* Range – all angles must be accessible, as the probes must be deployed from various attachment points
* Jitter – unknown at this stage from other sub-systems
* Settling Time – unknown at this stage from other sub-systems

### Probe

Pointing direction:

* Comms will utilize and omni-directional antenna. No ACS requirements
* EPS will rely on internal batteries
* Propulsion – retrograde during de-orbit burn
* Thermal – prograde during atmospheric entry
* EDL – trajectory guidance if using a lifting trajectory
* De-tumble after deployment

Pointing accuracy:

* Exact pointing accuracies unknown at this design stage
* Dependent
* Large deadbands to save fuel [MSL entry]

Maneuver requirements:

* 180 degree flip maneuver during after de-orbit burn, prior to atmospheric re-entry
* Maneuver rate TBD
* Lift/bank control (likely if lifting trajectory is chosen)

## ACS Type

### Orbiter

* Primarily dependent on TT&C pointing requirements and accuracy

Orbit control type will be (SMAD Table 11-4):

* Three-axis, zero-momentum (3 wheels + thrusters)
* High pointing accuracy
* Combination of thrusters and reaction wheels
* Thrusters used for slewing and momentum dumping
* RW for high accuracy pointing
* Control Moment Gyros (CMG) likely not needed, TBD in hardware selection phase

### Probe

* If lifting trajectory is used, spin-stabilized will not be applicable.
* Spin stabilization also unlikely due to the need to execute a 180deg flip maneuver between de-orbit burn and atmospheric re-entry
* Three-axis control. Unknown landing ellipse requirement, therefore it will work for lifting if necessary. The powered descent phase needs to be 3-axis anyways, so this wil reduce number of systems on board

Control type will be (referring to SMAD table 11-4):

* Three-axis, zero momentum (thruster only)
* Pointing accuracy 0.1 – 5 deg
* High rates possible

## Disturbance Torques

### Orbiter (TODO: get MRO data)

Solar

|  |  |
| --- | --- |
| *Inputs* | * Surface area of s/c * Center of solar pressure (assumed geometric center) * Center of gravity of s/c * Reflectance factor q * Angle of incidence with the sun |
| *Outputs* | Worst-case torque caused by solar radiation |
| *Iteration Choices* | * Assumed reflectance factor of q = 0.6. Based on SMAD * Incidence angle assumed to be worst case, i = 0 * Cp and Cg are co-incident at this phase, producing no torques |
| *Result* | **Negligible at this phase** |

Gravity Gradient

|  |  |
| --- | --- |
| *Inputs* | * Orbital radius * Moments of inertia about x, y, z (Ix, Iy, Iz) * Max deviation of **LONGEST AXIS** from local vertical (depends on orientation in orbit, discuss with Filippo)   + 23.5 degrees as per EPS reqs. However, this is misleading as the gg will pull it towards nadir, and this may only have an effect if comms requires the sc to be held at this angle from nadir |
| *Outputs* | Maximum torque caused by gravity gradient |
| *Iteration Choices* | * S/C will be nadir pointing, and will thus gravity gradient will work as an advantage (IF slender shape points aligned along z-axis * This design phase, s/c is a symmetrical cube Cp and Cg are co-incident at this phase, producing no torques |
| *Result* | **Negligible at this phase** |

Aerodynamic Drag

|  |  |
| --- | --- |
| *Inputs* | * Atmospheric density * Drag coefficient * Surface area * S/c velocity * Center of aerodynamic pressure * Center of gravity |
| *Outputs* | Maximum torque generated by aerodynamic drag |
| *Iteration Choices* | * Placement of appendages is unknown, cannot determine resultant Cp (assuming geometric center to be center of pressure\_ * Current design: CG and CP both located in center |
| *Result* | **Negligible at this phase** |

Magnetic

|  |  |
| --- | --- |
| *Inputs* | * Radius from planet dipole to s/c * Magnetic moment of planet * Residual dipole of vehicle |
| *Outputs* | Maximum torque caused by magnetic fields |
| *Iteration Choices* | * At Earth GEO, negligible. If magnetic field of mars at the design orbit is less than this, then this disturbance can also be neglected [book:Spacecraft-FCS-design, p1] * Earth orbit: mag only of significance in orbits below 1500km. (32,000 nT) * Mars magnetic field is max 1500 nT [<https://www.planetary.org/blogs/emily-lakdawalla/2008/1710.html>, retrieved 5/6/2020] |
| *Result* | **Negligible at this phase** |

Because the geometry of the spacecraft is unknown at this phase, it is not possible to calculate disturbance torques. It will be assumed that the slewing torque to execute maneuvers such as probe deployment will be significantly greater than the disturbances, and the reaction wheels will be sized using these torque requirements.

Depending on the torque required, which is determined by the MOI of the orbiter, it may not be feasible to utilize only reaction wheel. As such, thrusters will need to additionally be utilized to exert enough torque to accelerate the spacecraft sufficiently to execute the maneuver in time. Knowing how quickly the orbiter must be able to slew (such as for deployment of probes) will determine thruster sizing.

For this phase it will be assumed that reaction wheels will be used to maintain pointing of the spacecraft, which is needed in any case for the Nadir pointing of LGA antennas towards Mars.

Max Disturbance Torque Tool:

Starts with lines of variable declarations, as described above as inputs to the disturbance torques.

Then identifies maximum disturbance by comparison.

V&V:

All functions are verified and validated using input data from the FireSat spacecraft. Its orbital parameters are used as inputs as described in SMAD, and the outputs of the numerical model are compared to the given values. This is done using unit tests. This helped identify small errors concerning units (km vs m, rad vs deg). These have subsequently been fixed.

These two steps combined work as verification (where the code is checked for errors) and validation (where a sample dataset is used to validate that the output is relevant to the mission).

### Probe

* Aerodynamic Drag
* Other torques negligible

The probe will need to be inherently stable throughout the entry and descent phase. The addition of active attitude control is dependent on whether the trajectory will be ballistic or lifting. In the case of lifting trajectory, thrusters will be utilized to keep oscillations within the acceptable deadbands.

The sizing of such thrusters will only be done once it is known which trajectory type is used. Then, the required thrust can be computed based on max AoA excursion, and rough MOI estimates of the probe. Again, the geometry is unknown at this phase, and therefore the aerodynamic properties cannot be estimated.

## Orbiter Geometry Model

## 

Based on MRO, assuming dish to be same size as what we need. Body dims estimated to be 1.5x1.5x1.5, which is large enough to accommodate the 4 propellant tanks

## Hardware Selection

Throughout the orbit, the vehicle will experience both cyclic and secular angular momentum. We estimate 80% of the torques to be cyclic, and 20% to be secular. “…the cyclic torques will cause cyclic rates, while secular torques cause gradual divergence.” [SMAD p369]. These cyclic torques will drive the lower limit design of the reaction wheels’ angular momentum, which needs to be large enough to withstand these torques throughout the orbit, without the need for active control. The upper limit of the angular momentum (if any) is defined by the thrusters’ fuel to precess the momentum. Secular torques drive the thruster mass / thrust capability design, since it must desaturate the angular momentum of the reaction wheels. Aerodynamic disturbance torques are secular because the drag vector is constant wrt the spacecraft throughout the orbit. The torque is absorbed in the momentum wheels, which are desaturated once full.

Reaction Wheel Sizing: TORQUE

* Disturbance Rejection: The torque of the reaction wheels must at a minimum be able to counteract the worst-case disturbance torque, with a margin. Typically not a driving factor
* Slew Torque: Reaction wheels on the orbiter will be required to slew the spacecraft during maneuvers. The largest will be after the aerobraking, when it needs to rotate from a max-drag attitude to min-drag attitude. This is a 90deg rotation in 50min window. The thrusters may assist, but preferably RW only because of better pointing accuracy.
* Momentum Storage: Roughly integrated the worst-case disturbance torque over half its period if cyclic. For gravity gradient this is ¼ orbital period. For solar radiation this is ½ orbital period. A simplified equation is used, using the sinusoidal rms rather than a complex integrating function. For aerodynamic disturbances this is not cyclic and will accumulate throughout the orbit. How much depends on orbital altitude. The final circular 200km circular orbit will drive the design, however it is noted that immediately after aerobraking the orbit will be lower and will require more frequent momentum desaturations. Frequency of desaturations is TBD.

### Orbiter

* Three-axis control:
  + Thrusters for momentum desaturation and high-rate slew maneuvers
  + Reaction wheels for controlling resisting disturbances and providing high-accuracy slewing

### Probe

* Likely thruster only (see 2007\_Brugarolas on MSL attitude control) wrt high attitude deadband vs low rate deadbands

## Control Law

### Orbiter

### Probe

* See 2013\_SanMartin\_Dev-of-MSL

## Budgets (Basic sizing based on MRO and MSL, **NO CALCULATIONS**)

### Orbiter (selection based on MRO)

* RCS (not MRO) mass TBD
* RWA 2 – 20kg; 10 – 110W (per wheel) [SMAD]
  + MRO 10kg per RW x 4 wheels
  + MRO mass = 1000kg, our orbiter = 1990kg
  + Estimate 20kg per RW 🡪 **80kg** total
* Sensors:
  + Star trackers 2 – 5kg; 5 – 20 W
  + Sun sensors 0.1 – 2kg; 0 – 3 W
  + IMU 1 – 15kg 10 – 200; W
* TOTAL MASS:

### Probe (based on MSL)

* RCS propellant for MSL:
  + Max 15kg for lander of 900kg (so 1.666%)
  + Assuming our probes of 162kg 🡪 2.6kg propellant to be safe
  + 80% of sys is propellant, 3.24 total
  + 280 kg/m3 cold gas
* Sensors:
  + IMU 1.5kg x2 = 3kg (try to find data about Honeywell IMU)
    - 15W x2 = 30W
  + Landing radar (Terminal Descent Sensor TDS)

### Other Sub-systems

#### EPS:

Orbiter power range: 25 – 333 W

Probe power range: 10 – 200 W (IMU)

#### Thermal:

Orbiter thermal range:

Probe thermal range:

#### CDH:

Commands from earth:

* s/c attitude model [2007\_You]

Commands from orbiter:

* attitude estimation [2007\_You]

Housekeeping telemetry:

* TBD