UA Mars Orbiter Spacecraft and Mission Design

# System Engineering

The University of Arizona Mars Orbiter (UA-MO) is a spacecraft that is designed to fulfill the mission requirements set forth by the University of Arizona Lunar and Planetary Laboratory (UA-LPL). The mission must satisfy the science objectives and make use of the science payload listed in the attached System Requirement Document (SRD).

The MRO will use an Atlas V 551 launch vehicle and will leave earth orbit on March 22 2021. It will cruise to Mars using a Type-I trajectory where it will establish a circular polar orbit at an altitude of 400 km on October 8th 2021. The mission is required to last for three years with an option of a three-year extension.

The overall design of the mission can be broken into three parts: launch, cruise, and arrival (science-gathering). The launch phase includes all parts of the mission while the spacecraft is in the launch configuration (attached to the Atlas V 551 launch vehicle). The cruise phase is defined as the portion of the mission in which the spacecraft is in the cruise-configuration (separated from the launch vehicle, but the science payload has not yet been deployed). The arrival phase is defined as the portion of the mission in which the science payload is active and deployed.

The spacecraft can separated into the component subsystems for analysis. The subsystems are as follows: propulsion, attitude control, physical structures, power, thermal, command and data handling, and telecommunications. The mass and power budget for each of the subsystems and overall design is shown in the attached mass equipment list (MEL). The MEL also contains the historical evolution of the spacecraft design.

The Atlas V 551 was chosen due to reasons…

* **Executive Summary** **of overall Design**
* Design Assumptions todo
* Mass and power budget todo
  + MEL
  + Historical evolution of project
* Mass equipment list todo
* Launch Vehicle discussion and adapter todo
* Functional Block diagram and ConOps todo

# Mission Design and Analysis

## Requirements

The spacecraft mission must adhere to the mission requirements set forth in the SRD:

1. The mission shall be a type-I trajectory to Mars with a circular polar orbit insertion.
2. The spacecraft shall arrive at Mars in October 2021.
3. The spacecraft shall maintain an orbit at an altitude of 400 km.

## Assumptions

The following assumptions are made with regard to the mission design.

The first table of values show the list of assumptions for the launch. The second table include the comparable variables for the arrival at Mars. Both tables of values are critical for calculating the required for the mission.

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| --- | --- | --- |
| Variable | Value | Units |
| Date of Launch | March 22, 2021 | N/A |
| Longitude | 181.44 | Degrees |
| Eccentricity | 0.0167 | N/A |
| Semi-Major Axis | 149,598,020 | Km |
| Angle of Inclination | 0 | Degrees |
| Longitude of Perihelion | 102.958 | Degrees |
| Longitude of the Ascending Node | 0 | Degrees |
| Radius | 149,905,909.7 | Km |
| Velocity | 29.89 | Km/sec |
| True Anomaly | 78.48 | Degrees |
| Flight Path Angle | .9348 | Degrees |

Table X.x Assumed Launch Variables

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| --- | --- | --- |
| Variable | Value | Units |
| Date of Arrival | October 8, 2021 | N/A |
| Longitude | 333.22 | Degrees |
| Eccentricity | 0.0934 | N/A |
| Semi-Major Axis | 227,939,133 | Km |
| Angle of Inclination | 1.849 | Degrees |
| Longitude of Perihelion | 336.093 | Degrees |
| Longitude of the Ascending Node | 49.572 | Degrees |
| Radius | 206,671,197 | Km |
| Velocity | 26.94 | Km/sec |
| True Anomaly | 357.128 | Degrees |
| Flight Path Angle | -0.2452 | Degrees |

Table X.x+1 Assumed Arrival Variables

## Transfer Ellipse Design

The trajectory from earth to Mars was determined by using the Patched-Conic Approximation procedure. The Patched Conic Approximation provides an easier alternative to solving the four-body problem (sun, spacecraft, earth and Mars) and does so with sufficient accuracy. There are three portions of the Patched Conic Procedure: the departure phase, the cruise phase, and the arrival phase.

The departure phase consists of plotting a hyperbolic exit from earth while assuming that the sun and the destination planet do not affect the hyperbola. The cruise phase involves only the spacecraft and the sun and uses a transfer ellipse. The final phase has only Mars and the spacecraft and uses an arrival hyperbola. A diagram of the patched conic orbit shows the relationship between the destination planet, the departure planet, and the path between.

The transfer ellipse design process begins with the assertion that the ellipse must contain earth at launch and Mars at arrival. In addition, the time of flight must be equal to the number of days allotted for travel. These two requirements are the rendezvous conditions.

The algorithm for determining the parameters of the transfer ellipse is included in its entirety in Appendix (TODO), but is briefly described below.

First, the Line of Apsides is placed through earth at launch. This means that the true anomaly of earth is set to 180 degrees and the true anomaly of Mars is offset by 180 degrees. Next, the eccentricity of the ellipse is calculated by using the following equation:

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Then, the periapsis radius is calculated using this equation:

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Finally, the semi-major axis is calculated using the previous two values:

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From here, the Kepler equations allow for the time of flight to be calculated. The algorithm included in the appendices shows the final result being TODO. This result was achieved by performing the steps listed above until the time of flight was less than 200 days. If the calculated time of flight was more, the starting Line of Apsides was incremented by one degree and recalculated. Once the time of flight was within the required timeframe, the transfer ellipse design was completed with the resulting eccentricity, periapsis radius, and the semi-major axis being determined.

## Departure Trajectory Design

The trajectory of the spacecraft leaving earth can be defined by two variables: the transfer plane and the velocity at sufficient distance away from earth, As with the design of the transfer ellipse, the exact algorithm used to determine the and C3 for this design is found in the appendix while the essential steps are listed below.

First, the angle of inclination of the transfer plane must be determined. This can be done using the law of sines and the law of cosines and the following diagram :TODO. Further, the and C3 can now be solved for using the following diagram as a guide: TODO.

## Departure Hyperbola Design

The departure hyperbola is the path that the spacecraft takes as it leaves earth and is not yet aligned with the transfer hyperbola. There is a critical point along that path, the injection point, at which the spacecraft achieves enough velocity to leave earth orbit and begin traveling along the transfer hyperbola. The required velocity at the injection point is found by the following formula:

where is the gravitational parameter for earth, and r is the periapsis radius from the transfer ellipse. From here, the parameters for the departure hyperbola can be calculated as follows:

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And

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## Arrival Trajectory Design

The first step in designing the arrival trajectory is very similar to designing the departure hyperbola, and similar equations are used. TODO

# Propulsion Subsystem

## Requirements and Design Assumptions

The driving requirement for the Propulsion subsystem is to provide enough propellant to meet the Isp demands of the mission from the previous section. However, the propulsion subsystem must be able to meet any translational velocity requirement given in the SRD.

## Thruster Selection and Sizing

To aid in the thruster selection and sizing for the mission, a thruster simulator was created in a software environment. The calculations for the thruster parameters such as specific impulse are shown in the attached appendix TODO. The results are summarized here.

## Trade Studies

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# Attitude Control Subsystem

## Requirements and Assumptions

There are three requirements for the Attitude Control Subsystem (ACS) listed in the SRD.

1. The spacecraft shall be nadir-pointed and meet the pointing requirements for the science payload.
2. It must be possible to make a 180-degree turn about any axis in thirty seconds.
3. The solar arrays shall be pointed at the sun within five degrees maximum pointing error.

The mission demands that there be several control modes that are listed below.

Control Mode 1: Launch Mode

During Launch Mode the Spacecraft will have all of the science payloads stored in the respective compartments. This means that there will be no pointing requirements.

Control Mode 2: Cruise

During Cruise Mode the Spacecraft will not have all of the science payloads stored in the respective compartments. For example, the star scanners need to be able to see the surrounding solar landscape. Furthermore, the solar panels will be deployed and need to be angled such that they are perpendicular to the solar rays for as much time as possible. In addition, the high gain antenna will need to be oriented toward earth to maintain communication lines. These requirements naturally lead to a 3-axis-stabilized system.

Control Mode 3: Oriting Mars

During Mars Orbital Mode the Spacecraft will have all of the science payloads exposed. In addition to the cruising requirements the spacecraft will also need to have the previously unused payload devices angled in the appropriate direction. Specifically, there are several nadir-pointing components such as the wide and narrow angle cameras. Also there is the spectrometer that must be pointing toward the surface of Mars.

The proceeding analysis assumes that the thrusters are fired in pair and are at the same thrust level.

## ACS Selection Process and Trades

The decision to go with the 3-axis-stabilized control system was simple to make. By using the Mission Suitability Matrix shown below TODO, it was clear that since many mission-critical objectives could only be met with nadir-pointing science payloads that the decision was between the 3-axis option and the momentum biased configurations. The fact that planetary observance was poor with the momentum biased option meant that although the 3-axis choice was far more expensive; it was the best choice for meeting the mission requirements.

## Environmental Torque Analysis and Disturbance Torques

There are several environmental and disturbance torques that were analyzed for this mission design: solar, magnetic, and gravity-gradient torques.

Solar Torque:

The resulting torque from the solar pressure was calculated on the solar panels near earth (the maximum) and was found to be TODO. The procedure for arriving at this number is as follows. First, the solar torque is the sum of torques due to photon absorption, specular reflection, and diffuse reflection. Together, the sum can be written as

where is the incident solar radiation in W/, c is the speed of light, and .3 is a representative reflectance factor.

Magnetic Torque:

**Acronym List**

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| Acronym | Definition |
| ACS | Attitude Control Subsystem |
| MEL | Mass Equipment List |
| MRO | Mars Reconnaissance Orbiter |
| SRD | System Requirement Document |
| UA-LPL | University of Arizona Lunar and Planetary Laboratory |
| UA-MO | University of Arizona Mars Orbiter |