AIAA Meeting Papers on Disc, 1996, pp. 282-286
A9634741, AIAA Paper 96-3603

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AIAA/AAS Astrodynamics Conference, San Diego, CA, July 29-31, 1996, Collection of Technical Papers (A96-34712 09-12), Reston, VA, American Institute of Aeronautics and Astronautics, 1996, p. 282-286

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AIAA-96-3603-CP

MISSION DESIGN FOR THE FIRE AND PSI MISSIONS

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Abstract

The PSI (Primordial Structures Investigation) and FIRE (Far Infrared Explorer) Missions were proposed as NASA Midex missions to study the cosmic microwave background radiation, following in the footstep of the highly successful COBE Mission. We examine the design of these low-cost missions and discuss the various trades considered. Sun-Earth L2 libration point orbits have become extremely popular for many NASA astrophysics missions due to the constant cold observation environment and low energy required for access. Both PSI and FIRE have selected a small amplitude Lissajous orbit about L2 for their missions.

1. Introduction

The Cosmic Microwave Background (CMB) contains an imprint of the structure of the early universe at age 300,000 years. The COBE (Cosmic Background Explorer) Differential Microwave Radiometer first measured the anisotropy of this structure at an angular resolution of 7° nearly 30 years after its discovery. A critical test of current cosmological theories requires much higher angular resolution to answer questions such as: How old is the universe? Will the universe expand forever, or eventually collapse? What was the origin of structure in the early universe? How did the Universe evolve to its current complexity from near-homogeneity?

To answer these questions, the PSI (Primordial Structures Investigation) Mission proposed a mission using InP HEMT detectors and the sorption cooler to measure CMB amplitude fluctuations to a resolution of 15 arcminutes. The FIRE (Far Infra-Red Explorer) proposed a mission using the Caltech bolometer detector system with a sub-Kelvin cooler to image the CMB with a resolution of 8 arcminutes.

2. Mission Requirements

In order to achieve the high resolution, the sensitivity of the instruments for both missions require a constant and cold space environment for the observations. They must be shielded from the radiation of both the Sun and the Earth. This is particularly important for FIRE which has an onboard cryostat whose lifetime depends strongly on the spacecraft temperature. As the Midex missions are cost-limited (about $70 Million per mission excluding launch cost) and launch-limited (launch capacity as provided by the new MedLite Launch System), a simple spacecraft bus design and mission operations concept are required. A minimum of two full sky surveys are required to achieve the science goals, leading to a 2-year mission duration requirement.

3. Mission Design

Earlier studies for the SIRTF Mission (Ref. 1) indicated that a low Earth orbit (LEO) is highly undesirable for missions requiring a cold environment. In LEO, the spacecarft is in and out of Earth occultation every orbit; this affects both the power and the jitter requirements of the spacecraft. Furthermore, Sun-Earth avoidance maneuvers are required every orbit. This makes for very intensive, inefficient, and complex mission operations where most of the on-orbit time is spent avoiding a line of sight directed toward the Sun or the Earth. By contrast, a Lissajous orbit about the Earth-Sun L2 provides an ideal orbit for astrophysics missions (Ref. 2). L2 is about 1.5 million km from the Earth along the Sun-Earth line as depicted in Fig. 1 (not to scale) below. Lissajous orbits (oval in Fig. 1) are a variant of the halo orbits used successfully by the ISEE3, SOHO, and ACE missions.

In this orbit, both the Earth and the Sun are always on one side of the orbit. Thus half of the celestial

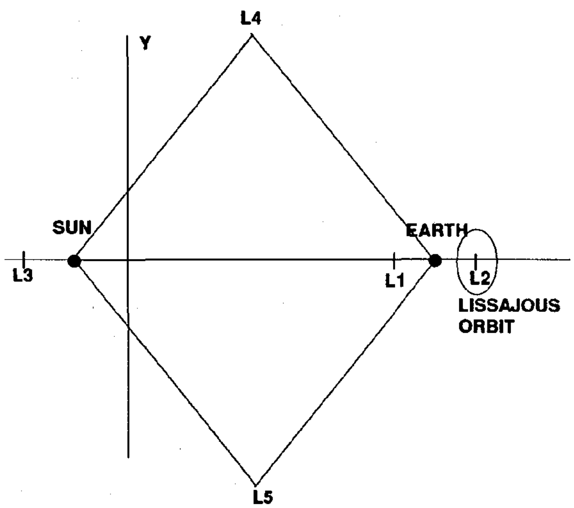

Figure 1. Sun-Earth System in Rotating Coordinate System

sphere is always available for continuous observations at all times. A simple fixed-solar-panel-sunshade easily isolates the instruments from the radiation of the Sun and Earth. The spacecraft rotates slowly (10 rpm for PSI, 0.15 rpm for FIRE) about an axis pointed at the Sun. The optical axis of the instrument is perpendicular to this axis. The rotation sweeps out a great circle of the sky and maps the entire sky in 6 months. The mission duration is around 2 years which provides a maximum of four all-sky surveys. The Earth-pointed fixed S-band antenna at the rear of the spacecraft can always maintain a link whenever a DSN station is available. Other than a daily maneuver to Sun-point the spacecraft to maintain this geometry, there are no avoidance maneuvers or target pointing maneuvers. Compared with a LEO mission, this approach is more efficient and requires simpler mission operations.

4. Trajectory Design

In order to satisfy the 5° antenna pointing requirement, a relatively small amplitude (Az=Ay=120,000 km) 'circular' Lissajous orbit was selected. Small amplitude orbits about L2 generally require more ΔV forLissajous Orbits Insertion (LOI) than large amplitude orbits. This is because these orbits all have a period of about 180 days. Thus a spacecraft in a small orbit is moving much slower. A spacecraft transferring to such an orbit must slow down more for LOI. In our case, the LOI ΔV is about 180 m/s. Since our total propellant budget is only 200 m/s due to launch and cost limitations, following the approach for the Relict-2 mission design suggested by David Dunham (Ref. 3), a lunar swingby was introduced and reduced the LOI ΔV to 15 m/s. While this added some complexity to the operations, it also reduced the risk from the large 19 m/s (3σ) MedLite launch dispersion (see next section).

The Mission Sequence is as follows (see Fig. 2 at the end of the paper): the spacecraft is launched by the MedLite Launch Vehicle System with a STAR37 upper stage into a 10-12 day highly elliptical phasing orbit with apogee just beyond the moon. Two to four revolutions in the phasing orbits were considered to optimize mission performance. During this time, the perigees provide several opportunities to correct the launch error. Near the last apogee, a gravity assist from a lunar swingby transfers the spacecraft to the Lissajous orbit insertion point. Early in the 90 day transfer phase, the science mission can begin, once the 5° sun-spacecraft-earth angle for the communications geometry is achieved. A final 15 m/s maneuver inserts the spacecraft into the desired Lissajous orbit about L2.

While on orbit, station keeping maneuvers are required every 6 to 8 weeks totaling 4 m/s per year. Z-axis control is not required. Unlike orbits around L1 where the interference of the Sun requires the maintenance of a Sun avoidance zone for the communications link, this is not a problem at L2 since the Sun is on the other side of the Earth. By inserting into the Lissajous orbit in its circular phase, occultation is also not a problem for the two year duration of the mission. Table 1 below lists the ΔV's with a total required of 148 m/s. The 200 m/s ΔV constraint provides a 30% margin to give the mission designer flexibility to work with the launch period and for recovery from anomalies.

Table 1. ΔV Budget

Maneuver	ΔV (m/s)
Trajectory Correction	120
Lunar Swingby	0
Traj. Balance (small ΔVs along nominal path)	5
LOI	15
Station Keeping (2 years)	8
TOTAL	148

5. Launch Analysis

The lunar swingby transfer to L2 provides greater performance and less risk than a direct launch. For the direct launch, a large single maneuver within 24 hours from launch must be performed to correct the launch error. Uncorrected, the error propagates in proportion to 1/velocity from the vis viva equation. Unlike planetary launches where the spacecraft quickly escapes the Earth's gravity at a high velocity, launches to libration point orbits typically have a C3 of about -0.6. Thus, the spacecraft slows down very quickly. At 24 hours after launch, the 19 m/s launch error will require 158 m/s to correct with an additional 16 m/s trim maneuver 7 days after launch. This brings the direct launch error correction ΔV total to 174 m/s.

The lunar swingby strategy reduces the the C3 from -0.6 to -1.8. For the MedLite Launch Vehicle System, this increased the launch mass from 585 kg to 604 kg. In addition, as mentioned before, it reduced the LOI maneuver from 180 m/s to 15 m/s. It also reduced the launch error correction maneuver from 174 m/s to 120 m/s while providing many opportunities for error correction.

This approach breaks up the single launch error correction maneuver into several smaller maneuvers each time the spacecraft returns to perigee, the first of which is 11 days after launch. This gives ample time for orbit determination. A total 4.5 revolutions in the phasing orbit is optimal, providing 4 perigees for launch error correction maneuvers. 2.5 revolutions provide the minimum 2 perigees since the launch errors will require two maneuvers to correct for both the launch energy as well as the phasing with the Moon.

In the event that one of the perigee maneuvers failed, analysis showed that many recovery strategies are possible even within the limitations of our 200 m/s propellant budget. Thus, this mission design is extremely robust.

The number of phasing orbits affect the length of the launch period. For a 2.5 rev phasing orbit mission, a 5-10 day contiguous launch period is possible each month, with a 2 minute launch window daily. For a 4.5 rev phasing orbit mission, this can be extended to 15 days (Ref. 4). This results from the phasing with the moon. The more opportunities there are to correct the launch error, the more flexible the launch window. But the 3.5 rev case does not work well as the moon adversely affects the orbits. This has been observed by others (Ref. 5).

6. Analysis Approach

The design of libration point trajectories with a lunar swingby is fairly well understood from the work of Dunham and members of his former team at GSFC and CSC. Nevertheless, it is not a simple exercise and requires painstaking and tedious numerical work along with experience and background. This may not be obvious to members of the astrodynamics community more acquainted with the conic world. Even though the lunar swingby looks like any other planetary flyby, the analysis techniques are really quite different. This is already apparent from the launch analysis as mentioned earlier. The energy regime where these orbits lie is very different from standard planetary flybys where conic approximations apply.

For libration point missions, conics are completely inadequate. Consequently, the orbital elements, or integrals, we take for granted for orbit design are now totally lacking. The designer must rely on experience and the small handful of techniques developed by practitioners in the field to find their needle in the haystack.

However, dynamical systems provide other structures to help with the analysis when the integrals are lacking as in our problem. In particular, for every periodic orbit, we can find its associated stable manifold. These are just the collection of orbits which approach the periodic orbit at the fastest rate possible. Similarly, there also are the associate unstable manifold of orbits which depart the periodic orbit in the fastest rate possible. These objects can be computed using Floquet theory. They are provided by the eigenvectors of the variational matrix with positive (for stable) and negative (for unstable) real parts.

In our analysis, we find the Lissajous orbit first using an algorithm developed by Pernicka and Howell (Ref. 6). Then by examining where the stable manifold of the Lissajous orbit intersected the lunar orbit, an initial transfer trajectory is found. The lunar phasing is then added iteratively to find the final orbit. The SWINGBY Program developed by

GSFC and CSC (Ref. 7) was used to perform the final calculations. More detailed discussion is provided by Ref. 7.

Dynamical systems theory has not enjoyed the wide acceptance in the astrodynamics community as it has in other disciplines. While traditional methods have served us well and produced many spectacular missions, the demands of future missions and the trend in resource limitations will require new approaches. Dynamical systems theory is one of the tools that will help us answer the challenges of the future.

7. Conclusion

Libration point trajectories are ideal for astrophysics missions such as PSI and FIRE. They provide a constant observation environment, a simple observation strategy and mission operations, thereby reducing the mission cost. The PSI and FIRE mission design is forgiving and robust. In comparison to a direct launch trajectory, the lunar swingby trajectory to L2 provides many more opportunities for launch error recovery. Maneuvers are spaced no less than 2 days apart to give ample time for orbit determination and operations planning. Whether the problem is a larger launch period, degraded launch vehicle performance, or a missed critical perigee maneuver, recovery is still possible to achieve the Lissajous orbit and complete the mission. The flexibility of the mission design and the simplicity of the observational strategy make these missions robust and cost effective.

Acknowledgments

The work described in this paper was carried out at the Jet Propulsion Laboratory, California Institute of Technology, and at the School of Aeronautics and Astronautics, Purdue University, under a contract with the National Aeronautics and Space Administration. The mission design was carried out with extensive support of the JPL Project Design Center and its helpful staff. Special thanks to David Dunham of APL for many helpful conversations and for some of the hard-to-find reference material. Thanks to GSFC for providing SWINGBY. Thanks to John Carrico of CSC for a late-night SWINGBY tutorial.

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Figure 2. PSI/FIRE Orbit XY, XZ Projections