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FINDING ACCEPTABLE JAMES WEBB SPACE TELESCOPE MISSION ORBITS FROM A FIXED ARIANE FLIGHT PROFILE

Mark Beckman[†], Leigh Janes[‡]

The James Webb Space Telescope (JWST) will be launched into orbit about the Sun/Earth L2 libration point. Trajectory design was recently completed which included expected separation states from the Ariane launch vehicle, constraints such as eclipses, maximum orbit size, maximum Sun-Vehicle-Earth/Moon angles, and launch opportunities. The results of this trajectory design give a set of possible trajectories for JWST with bounded stray light zones and provide a complete launch window. This data is also used to design the initial trajectory correction maneuver such that a maneuver towards the Sun is not required.

INTRODUCTION

The James Webb Space Telescope (JWST) is a large, infrared-optimized space telescope scheduled for launch in August, 2011. JWST will observe primarily the infrared light from faint and very distant objects. But all objects, including telescopes, also emit infrared light. To avoid swamping the very faint astronomical signals with radiation from the telescope, the telescope and its instruments must be very cold. Therefore, JWST has a large shield that blocks the light from the Sun, Earth, and Moon, which otherwise would heat up the telescope, and interfere with the observations (see Figure 1). To have this work, JWST must be in an orbit where all three of these objects are in about the same direction. The most convenient point is the second Lagrange point (L2) of the Sun-Earth system, a semi-stable point in the gravitational potential around the Sun and Earth[1].

The JWST project is managed by the National Aeronautics and Space Administration (NASA) out of the Goddard Space Flight Center (GSFC). International partners include the Canadian Space Agency (CSA) and the European Space Agency (ESA). The prime contractor is Northrop Grumman Space Technology (NGST) who is responsible for the design of the observatory. The Ariane 5 launch vehicle (LV) is being provided by ESA as one of their contributions to the program.

[†] Lead JWST Flight Dynamics Engineer, Goddard Space Flight Center, Code 595, Greenbelt, MD 20771
[‡] Flight Dynamics Engineer, Goddard Space Flight Center, Code 595, Greenbelt, MD 20771

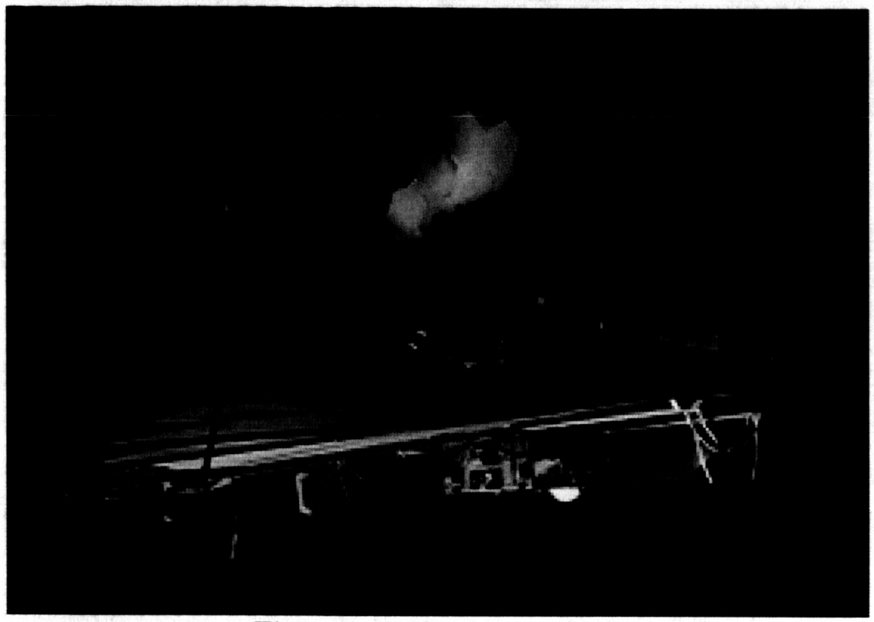

Figure 1: JWST Observatory

LAUNCH VEHICLE

JWST will be launched by an Ariane 5-ECA LV provided by ESA. The Ariane 5-ECA is a three-stage LV that can lift a significant amount of mass to high energy orbits. For a JWST high energy launch to a C3 of -0.7 km^2/s^2, the Ariane 5 ECA can lift 6800 kg. Ariane requires fixed flight profiles that are developed months prior to launch. This flight profile fixes the powered ascent and earth-fixed trajectory for any launch using that profile. Multiple profiles may be developed though, each with a different earth-fixed trajectory.

The Ariane 5 ESC launch vehicle consists of the following three stages. The EPC is the main cryogenic stage with a Vulcain 2 engine. The EAP are the solid propellant stages attached to the main rocket. The ESCA is the upper cryogenic stage which ignites four seconds after EPC shutdown. There is no coast phase. The ESCA stage cut-off command occurs when the guidance algorithm detects the final required orbit.

Ariane launches from a launch site in Kourou, French Guiana. The latitude of Kourou is a low 5.1° in latitude resulting in outgoing trajectories very near the equatorial plane.

In September 2004, a Technical Interchange Meeting (TIM) was held between GSFC, ESA and NGST[2]. A preliminary flight profile was developed at this meeting that was the basis for this analysis. The flight profile was developed by constraining only the apogee altitude (equivalent to C3) of the target orbit. No constraints were placed on inclination, perigee altitude, or lighting conditions. Ariane constraints, such as range safety and ground station coverage, were obviously still included. The resulting spacecraft separation state is fixed in an Earth Centered Fixed (ECF) frame regardless of launch time. However, the resultant trajectory is very much a function of launch time.

Included as part of the preliminary flight profile was a covariance state at separation[3]. This covariance was used to perform Monte Carlo analyses for the first Mid-Course Correction

(MCC) maneuver[4]. The ΔV cost of the first MCC maneuver is used to further define the launch window.

LAUNCH WINDOW METHODOLOGY

During Phase A planning, the JWST project had few constraints on the mission orbit for JWST. This analysis was performed to initially define the solution space for mission orbits achievable from a fixed Ariane flight profile. The solution space was then pared by adding various new mission constraints.

The preliminary flight profile was designed to meet an apogee altitude target of 1.3e6 km. For this analysis, the ECF separation state of the flight profile was used as the initial condition. The analysis however varied the velocity magnitude of the separation state in order to achieve orbit about L2. So for each solution, a unique set of initial conditions exist which differ from the flight profile only in the initial energy of the orbit.

The methodology for this analysis consisted of generating fully dynamic solutions that remain in orbit about L2 for 10 years. Since the sensitivity of the orbit after several years to the initial energy becomes less than that capable of double precision computers, small correction maneuvers were added about every 18 months.

Solutions were found for every day of the year in order to see the dramatic seasonal effect of a fixed equatorial inclination separation state. Different solutions were also found by varying the launch time of day. Starting at local noon, the algorithm searched forward and backward to the edge of the daily launch window. When small correction maneuvers were no longer sufficient to achieve 10 full years in orbit about L2, the search ended.

The software used for this analysis is a commercial off the shelf (COTS) software package called Satellite Tool Kit™ (STK™). The STK/Astrogator™ module within STK™ allows trajectory design and targeting. The STK/Connect™ module allowed the analysis to be scripted and automated through an executive program in Matlab™.

The coordinate frame used for this analysis is the Rotating Libration Point (RLP) frame. The primary axis is defined as the line from the Sun to the Earth. The RLP +*Z* axis is along the north ecliptic pole. Figure 2 graphically shows this coordinate frame.

For each launch day and time, the script varied the velocity magnitude at separation in order to extend the time the trajectory spent about L2. A control box was constructed about L2, one side of which was perpendicular to the Sun-Earth line and through the Earth (see Figure 3). Because of the instability of the region around L2, the trajectory would eventually leave the box once it entered. The side of the box that the trajectory left through determines the change in energy required to extend the stay in the box. Exiting the top, bottom and outside edges of the box indicated that the energy level was too high. Exiting the inside edge of the box, through the Earth, indicated that the energy level was too low. The script iterated until the change in velocity magnitude was zero to double precision. At this point, a small correction maneuver had to be added to the trajectory. Future targeting would vary the impulsive ΔV magnitude in the velocity direction while still trying to increase time in the box.

The solution sets consisted of solutions for all 365 days of the year. Solutions were obtained every 30 minutes during the daily launch window which extended from about 0930 GMT to about 1700 GMT. Each solution consisted of a 10-year trajectory with a unique initial energy. In all, over 4700 solutions were found giving over 47,000 years of trajectory data.

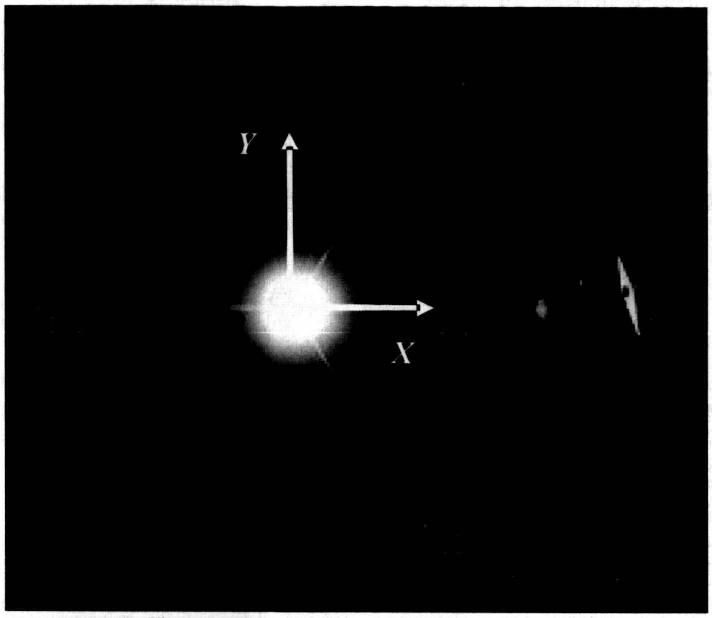

Figure 2: RLP Coordinate Frame

CONSTRAINTS

The only existing mission orbit constraint was that no earth or moon eclipses occur during the transfer or mission orbit[5]. Earth and moon eclipses can only occur near the Sun-Earth line. Any large halo orbit about L2 could not possibly generate any eclipses during the mission orbit. The elongation angle (angle off the Sun-Earth line) represents a good predictor of whether eclipses are likely.

The sunshield blocks direct sunlight onto the delicate optics of the primary and secondary telescopes. However, it is not designed to prevent indirect light from the Earth and Moon. This stray light could potentially cause damage to the telescope. Concurrent analysis from the science team indicated that minimizing the angular separation between the Sun and the Earth or Moon would limit the exposure to stray light. It was proposed that a maximum excursion from L2 in the Y direction be restricted to less than 800,000 km. A similar requirement was proposed on the Z direction with a limit of 500,000 km. This second requirement is actually redundant since no solution met the first requirement but failed the second one.

The second proposed mission constraint was that the first MCC maneuver not be performed with a component in the Sun direction. All the thrusters on JWST are on the opposite side of the sunshield from the telescope. With the telescope always on the anti-sun side of the sunshield, direct sunlight would illuminate the optics during a sunward burn if protection wasn't added. Initially, it was thought that the stowed configuration of JWST would provide this protection through the first MCC burn at L+12 hrs. This attitude constraint on MCC1 did not directly affect the solution set of mission orbits generated. However, added with the spacecraft ΔV budget and the LV dispersions, this constraint did eliminate launch window solutions.

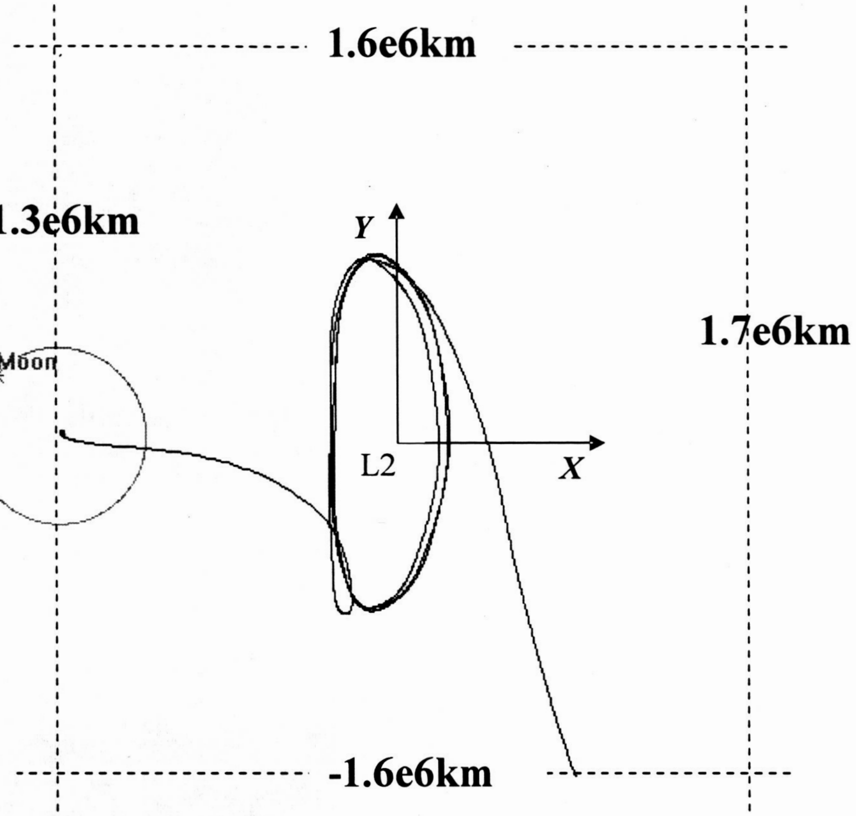

Figure 3: Control Box About L2

RESULTING LAUNCH WINDOW

Depending on launch day and time, different missions' orbit solutions were found even though the separation state is fixed in the ECF frame. The JWST project has no requirement on the type of L2 orbit achieved (i.e. halo, lissajous, or torus). The resultant mission orbit solutions covered a wide spectrum from typical halo orbits to large kidney-shaped degenerative orbits as shown in Figure 4. The orbit projections in Figure 4 represent just two specific cases. The projections on the left represent a typical near-halo orbit within the excursion limits proposed by the stray light analysis. The projections on the right represent a typical large kidney-shaped orbit that is achieved with a very late launch time. All of these orbits have similar stability properties and would require similar stationkeeping strategies to maintain.

Within a single launch day, the resulting solutions varied quite a bit because of the different outgoing asymptote of the transfer trajectory. Figure 5 shows the *YZ* projection of nine different launch times on the same day. Very early launch times give a torus-like orbit about L2. By midday, the orbit becomes nearly a halo. Late launches give a lissajous-type orbit.

The size of the orbits in the solution set covered a wide range too. The maximum excursion along the RLP *Y*-axis for each solution is shown in Figure 6 for all 4773 solutions found. Orbit box size, in the RLP *Y* direction, is primarily a function of launch time of day. The larger Y excursions were seen in the early and late launch times of day. The lower *Y* excursions, around 800,000 km, were generally since around a 1200 GMT launch.

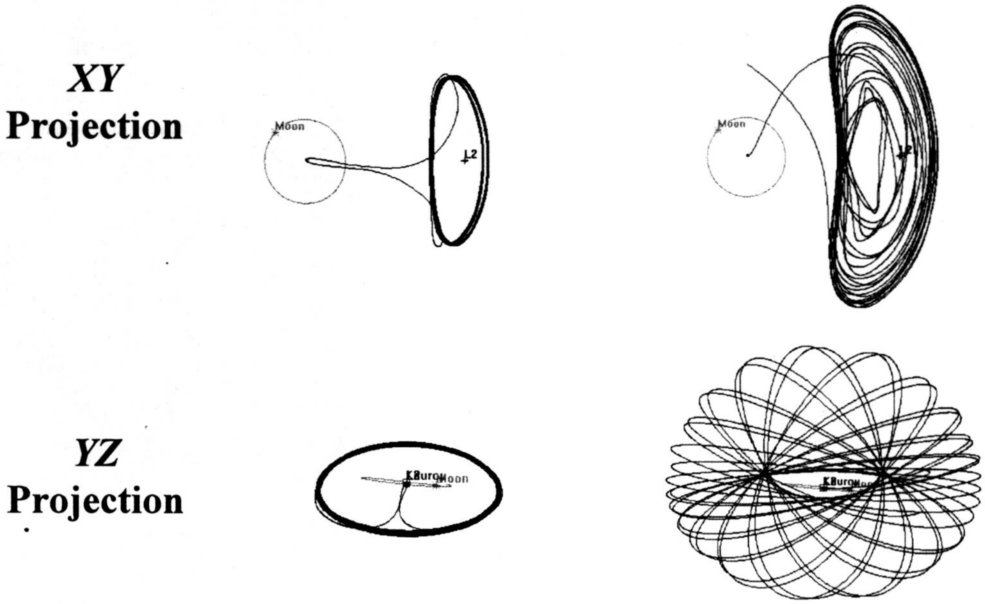

Figure 4: Mission Orbit Examples

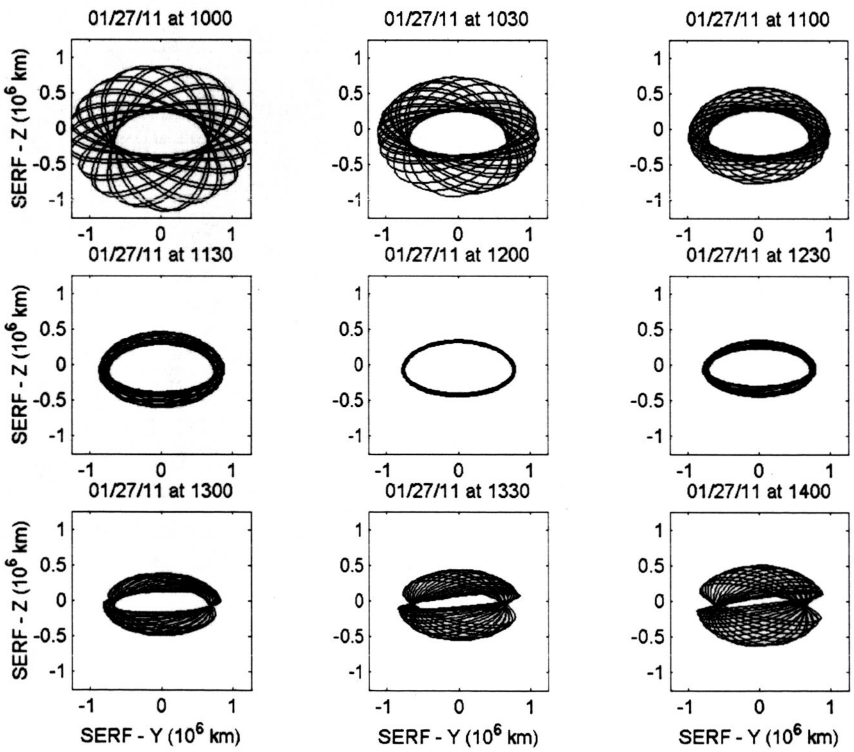

Figure 5: Orbit Types From Varying Launch Times Over a Single Day[§]

[§] Figure provided by ai solutions, Inc.

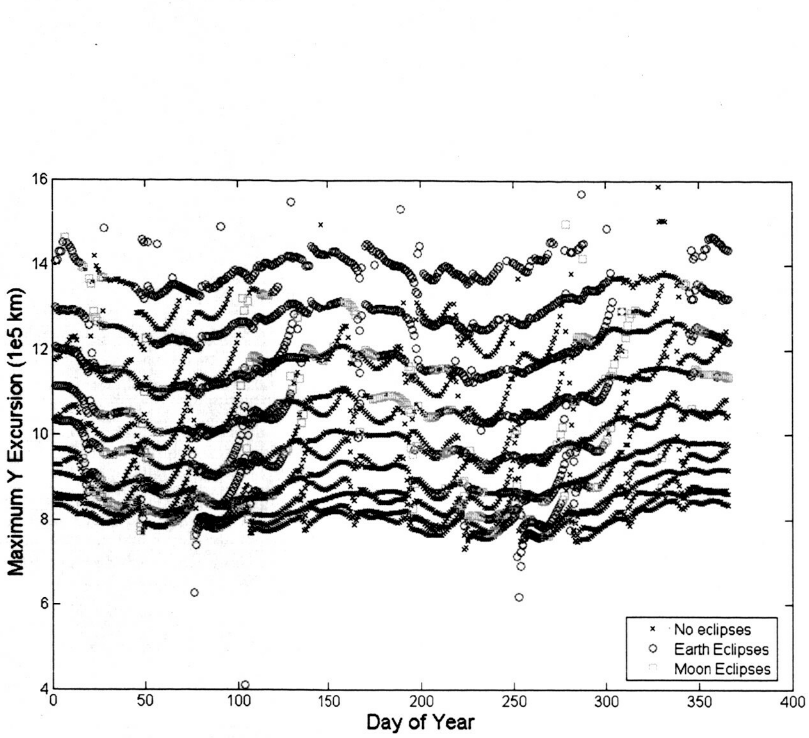

Figure 6: Maximum *Y* Excursions

The orbits get quite large near the beginning and end of the daily launch window, over 1 million km in *Y*. The maximum *Z* excursion is correlated to the maximum *Y* excursion. Figure 7 shows the correlation between the two for the 270 solutions that met all final requirements.

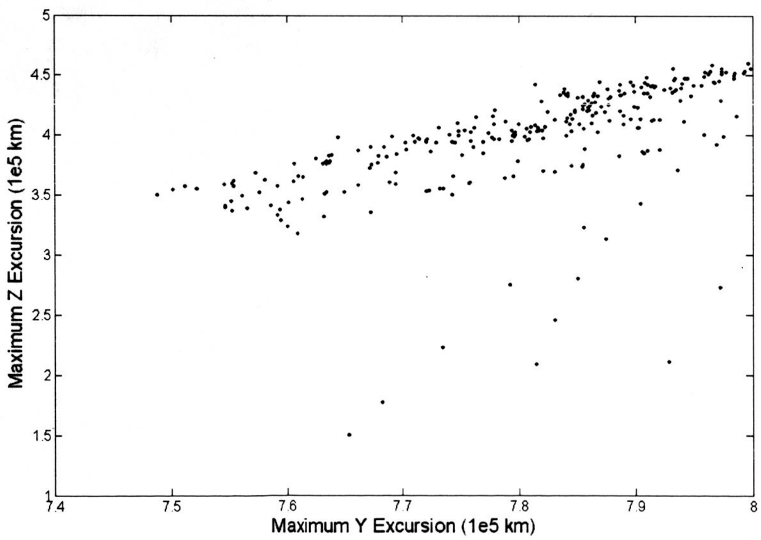

Figure 7: Correlation Between Maximum *Y* and *Z* Excursions

The eclipsed solutions generally occurred only when the minimum elongation angle (angle off the Sun-Earth line measured from Earth) was very small. Figure 8 shows the minimum elongation angles for all solutions that met the final requirements. Note that many of the final acceptable solutions do have small minimum elongation. These solutions generally surround the set of solutions excluded due to eclipse constraints.

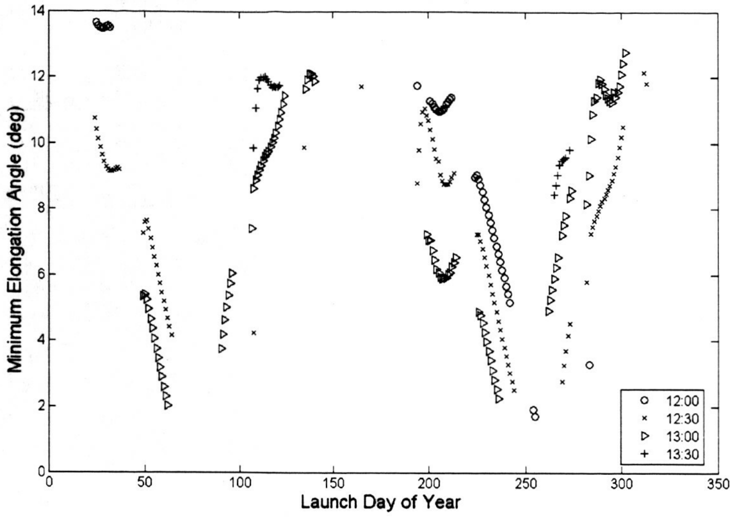

Figure 8: Minimum Elongation Angle

With the orbit box constraints added, the stray light implications are limited. The maximum Sun-Vehicle-Earth (SVE) angle is capped at less than 33 deg. The maximum Sun-Vehicle-Moon (SVM) angle is capped at less than 48 deg. Figures 9 and 10 show the maximum SVE and SVM angles for all final solutions.

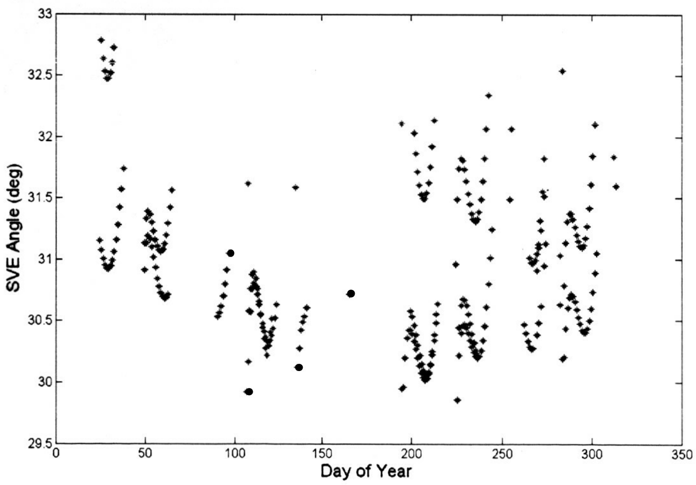

Figure 9: Maximum SVE Angle

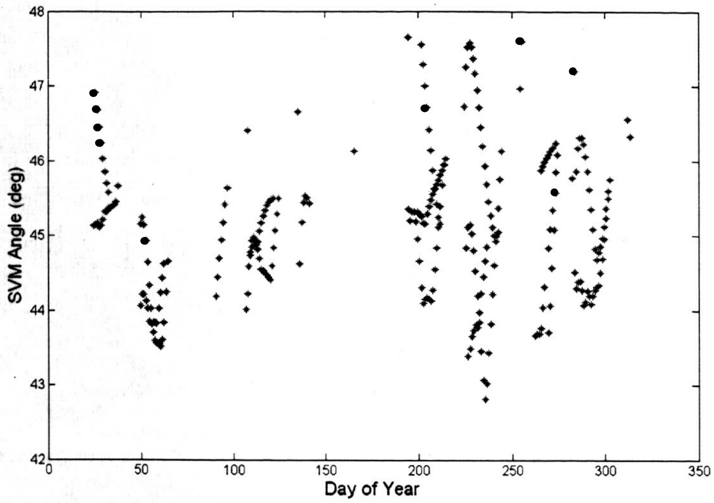

Figure 10: Maximum SVM Angle

Figure 11 shows the susceptibility of the observatory to stray light. The shading indicates level of susceptibility to stray light along various spacecraft body vectors. The rectangle in the center represents the sunshield. The sun vector would be controlled within this box. The earth and moon vector limits are represented by the ovals. Three points, one for Earth and two for Moon, are identified as worst case stray light locations. For earth light, worst case is near the forward extreme of the earth limit box (towards spacecraft body frame +V1). For moon light, there are two cases: one near the forward extreme (towards +V1 axis) and one near the aft extreme (towards –V1 axis). These are the limiting cases within the mission orbit box constraints.

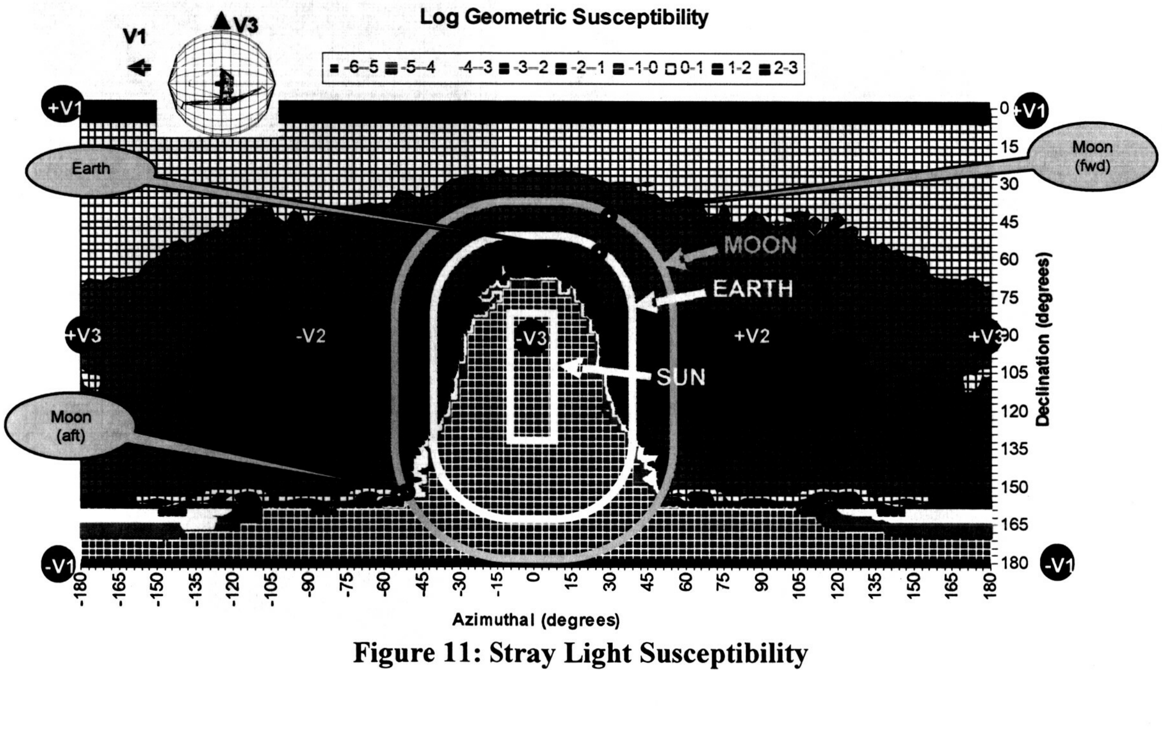

Figure 11: Stray Light Susceptibility

Starting with the entire set of solutions achieved (4774 solutions), mission requirements were then imposed resulting in a reduction in the set of acceptable solutions. The mission requirements imposed consisted of: a) avoiding earth and moon eclipses for the entire transfer and 10-year mission orbit (reducing to 3080 solutions), b) maximum orbit box size in order to minimize the stray light from the Earth and Moon on the telescope optics (reducing to 324 solutions) and c) launch window ΔV constraints including an attitude constraint on the first mid-course correction maneuver that prevents burns towards the Sun (reducing to 270 solutions). The original set consisted of a launch opportunity every day of the year with up to an eight hour launch window. The reduced set consisted of launch opportunities on about 140 days per year with an average daily launch window of about one hour. The available launch window is shown visually in Figure 12. The acceptable range of launch opportunities is the unhatched gray regions. Figure 13 shows the same launch window in a calendar format. No launch opportunities exist for long periods of time (i.e. from mid-November to late-January).

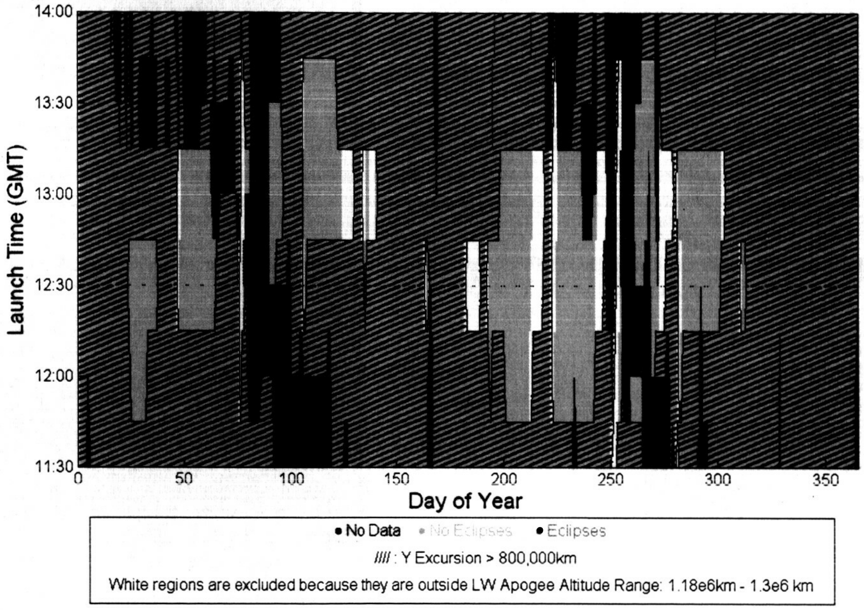

Figure 12: Launch Window

The analysis showed that acceptable orbits are achievable from the optimized ascent profile from Ariane. The impact of the mission constraints is only in launch opportunities. The launch window ΔV constraints, including the restriction in attitude for the first MCC maneuver, also impact the mission ΔV budget. The apogee altitude target given to Ariane will be biased low such that, even in the presence of 3-sigma LV dispersions, the energy achieved by Ariane is less than the desired energy for that particular launch day/time. The spacecraft propulsion system will perform the MCC maneuver to achieve the desired energy level. This maneuver will always be almost entirely in the velocity direction, which at the time of the maneuver, is generally in the anti-sun direction. Figure 14 shows the apogee altitude target for Ariane that is biased low, the LV dispersion range, the final launch window in terms of apogee altitude and the ΔV costs of the launch window and LV biasing.

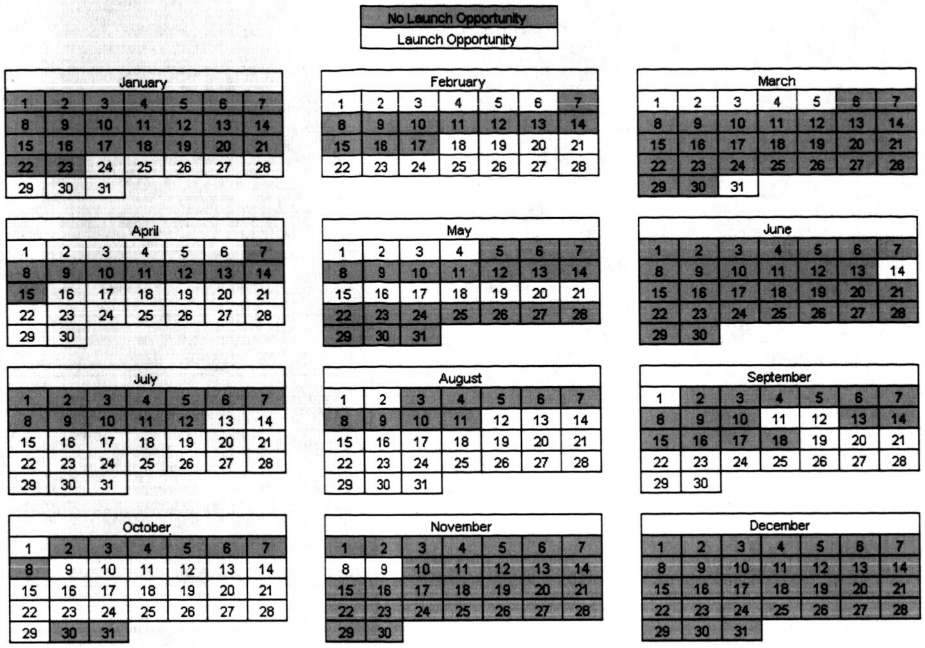

Figure 13: Calendar Launch Window

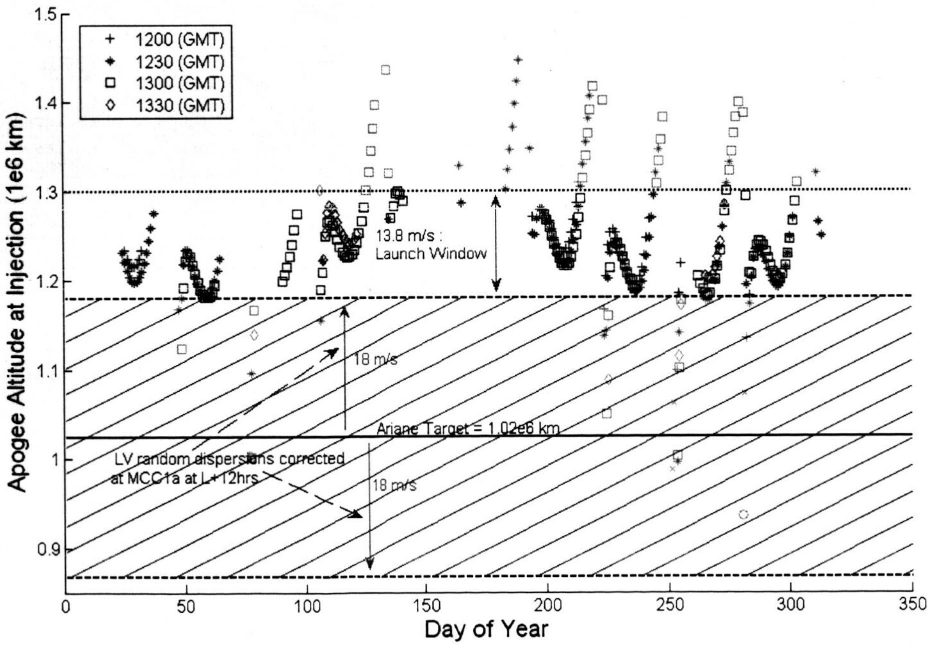

Figure 14: Ariane Apogee Altitude Target

CONCLUSION

Nominal solutions were obtained for JWST mission orbits from a fixed Ariane flight profile for every day of the year and for up to eight hours per day. These orbits varied greatly in character (i.e. halo, lissajous, or torus) and size. Adding the constraint that no earth or moon eclipses occur during the transfer or mission orbit eliminated about 35% of the full solution set. The launch opportunities lost were generally around the equinox periods.

The restriction on mission orbit size due to stray light concerns further reduced the remaining opportunities by 89%. The remaining launch opportunities were restricted to about a month on either side of the equinox periods.

The restriction at attitude for MCC1 required that the apogee altitude target given to Ariane be biased low to account for 3-sigma dispersions of the LV. A fixed range of apogee altitudes, to be targeted with MCC1, was then determined based upon the spacecraft ΔV budget. Further launch opportunities, about 17%, were eliminated because the orbit energy fell outside this launch window range.

The final launch window for JWST, with all constraints incorporated, is 144 days per year with up to 90 min per day. The apogee altitude target is biased low and the MCC1 maneuver will be in the velocity direction (anti-sun direction) and will correct for the random dispersions and energy biasing to achieve the unique energy for that launch day and time.

ACKNOWLEDGMENT

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