A COMBINED SPACECRAFT AND LAUNCH VEHICLE SYSTEMS APPROACH TO MISSION DESIGN FOR THE IRIS MISSION

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Abstract: *The Interface Region Imaging Spectrograph (IRIS) mission was launched in June 2013 on a Pegasus XL launch vehicle. To perform its solar observations, the IRIS spacecraft required periods of continuous eclipse-free viewing of the Sun. A typical spacecraft mission will have a well-defined orbit solution early in the planning phases. IRIS, however, was driven by science requirements, and did not require a specific mission orbit. Therefore, the selection of the mission orbit was made using a unique approach. The design team performed an integrated systems analysis approach to achieve a final mission orbit design that balanced the launch vehicle trajectory and performance with optimizing the IRIS science goals, to meet and exceed the requirements. This paper details the results of the integrated spacecraft and launch vehicle mission design trade studies that led to the final orbit targets and the Pegasus launch vehicle guidance scheme utilized for the IRIS mission. The actual achieved results are also presented.*

Keywords: *IRIS, mission design, Sun-synchronous, optimization, guidance.*

1. Introduction

The Interface Region Imaging Spectrograph (IRIS) mission successfully launched aboard an Orbital Sciences Corporation Pegasus XL launch vehicle on June 27, 2013 from Vandenberg Air Force Base (VAFB). IRIS is a National Aeronautics and Space Administration (NASA) Small Explorer (SMEX) mission to observe how solar material moves, gathers energy, and heats up as it passes through a little-understood region in the Sun's lower atmosphere. Observing how material and energy cross through this region is a crucial part of understanding the dynamics of the Sun. It can help explain what causes the ejection of solar material that travels toward Earth, causing space weather that can disrupt human technology. The concept of operations is for IRIS to point continuously at the Sun while conducting coordinated solar observations with other satellites and ground-based observatories. To conduct science observations, the IRIS spacecraft requires periods of continuous eclipse-free viewing of the Sun. Specifically, the mission orbit must provide IRIS with a minimum of 7 months of eclipse-free time (EFT) the first year and maximize the probability of achieving 7 months per year of eclipse-free viewing averaged over the first two years, starting one month after launch [1].

A typical spacecraft mission will have a well-defined orbit solution early in the planning phases, based on the science objectives. Therefore, the launch vehicle trajectory design will focus either on delivering the spacecraft to that mission orbit, or to an intermediate transfer orbit, followed by spacecraft maneuvers to reach the final mission orbit. For example, a launch into a geosynchronous transfer orbit (GTO) is often optimized to minimize the spacecraft delta-V needed to achieve a geosynchronous mission orbit. For the IRIS mission, the EFT science requirements drove the mission design, but allowed for a large trade space of orbit options. In addition, the IRIS spacecraft had no propulsion system – the launch vehicle injection orbit would be the mission orbit. Consequently a unique, integrated systems analysis approach was used throughout the mission integration process to achieve a final mission orbit design that balanced the launch vehicle trajectory and performance with maximizing the IRIS EFT.

The first step in the mission integration process was the launch vehicle selection. NASA's Launch Services Program (LSP) selected the vehicle following a competitive bidding process for all qualified vendors on the NASA Launch Services (NLS) contract. As part of the Task Order process, each proposal included a mission solution demonstrating how the mission requirements would be met. For IRIS, the core requirements were based on EFT. The calculation of this parameter involved propagating the various injection orbits and following a specific set of criteria that defines a valid EFT. The flexibility in the final orbit solution thus required some level of custom development to perform the EFT calculations as part of the proposal. Therefore, to better address this mission-unique requirement and to ensure consistency from each proposal, LSP developed a software tool (IRIS_EFT) that would allow users to quickly perform accurate EFT performance assessments based on an initial injection orbit. This tool was then later used in the design studies that followed.

After the Pegasus launch vehicle was selected, an orbit trade study was conducted between LSP, the IRIS Project, and Orbital to determine an optimal mission design for IRIS as a combined spacecraft-launch vehicle system. The EFT science requirements described above were the primary driver for the mission orbit. In general, higher altitude, Sun-synchronous orbits improve EFT periods. However, the spacecraft also had radiation and orbital debris requirements that limited the available altitude range. Since the launch vehicle injection orbit achieved on launch day would be the IRIS mission orbit, the Pegasus capabilities were included in the analyses to select the target orbit. This included the vehicle performance, orbital debris compliance, and dispersion characteristics. Additionally, the Pegasus vehicle offers multiple guidance strategies. Therefore, a second trade study was performed to select the best guidance method that maximized the probability of meeting mission science requirements.

This paper details the results of the integrated spacecraft and launch vehicle mission design trade studies, which led to the final orbit targets and Pegasus launch vehicle guidance scheme for the IRIS mission. Following this integrated design approach, IRIS was successfully launched into a nominal orbit (within 1-sigma). The resulting EFT predictions presented here show science observations are expected to exceed the 7-month requirement not only for the baseline two-year mission, but will continue to exceed the 7-month EFT requirement an additional four years.

2. IRIS Mission Requirements

The IRIS mission had several design requirements driving the target orbit trade space. As a solar observing mission, the primary requirement was for IRIS to have continuous eclipse-free viewing periods of the Sun. Specifically, the requirement was stated as follows:

> "The trajectory design and launch vehicle performance shall provide IRIS with a minimum of 7 months of eclipse-free viewing the first year and maximize the probability of achieving 7 months per year of eclipse-free viewing averaged over the first two years starting 1 month after launch. Vehicle performance and injection accuracy will determine the extent to which eclipse free viewing periods can be maximized for a given launch vehicle. The ascending node shall be selected near 6 am so that eclipse-free viewing is available at the June solstice. This allows coordination of science with the La Palma ground observatory. The IRIS SC has no propulsion system to correct for injection errors; therefore, the launch vehicle injection state is the only means of achieving the proper orbital conditions to meet the IRIS science requirements. The launch service provider shall provide the capability to achieve the science orbit requirements with 3 sigma level dispersions on the injection accuracy." [1]

The calculations for eclipse-free periods were also explicitly defined by the spacecraft.

- For eclipse calculations in orbital simulations, the Earth's shadow was determined by increasing the equatorial Earth radius by 250 kilometers and modeling the shadow as a cylinder; 250 km is the Earth's atmospheric altitude where the Sun's ultra-violet light of scientific interest becomes extinct
- All orbit propagations to determine eclipse free periods were performed using the 95.0 percentile solar flux (F10.7) and geomagnetic index (Ap) data from the NASA Marshall Space Flight Center (MSFC) monthly solar data report
- Orbit simulations would be run for 760 days from the launch date, simulating IRIS's nominal 2-year science mission plus a 30 day checkout period
- To compute EFT periods, any single eclipse-free period shall be at least 1 day in duration. The sum of these durations in months was computed by dividing the total number of eclipse-free days by 30.436875 (the average number of days per month per average Gregorian calendar year). This figure was then divided by 2 to get the overall average eclipse-free duration in months per year.

As IRIS has no propulsion system, the straightforward solution to provide reasonable EFT science periods would be to inject into a Sun-synchronous 6 am mean local time of the ascending node (MLT-AN) orbit at a high altitude. This would reduce drag impacts so that the Sun-synchronous orbit would be easier to maintain, which in turn would reduce shadow periods. However, there were three factors that would limit the altitude of the orbit.

First, IRIS imposed a requirement to limit the maximum three-sigma dispersed semimajor axis to 7078 km to meet the spacecraft radiation design requirements. Second, all NASA missions are subject to orbital debris requirements [2]. Since IRIS had no propulsion system to perform an end-of-life de-orbit maneuver, the spacecraft would be required to reenter through natural decay within 25 years after mission completion, thus limiting the altitude. Finally, the performance capability of the launch vehicle selected would factor into the achievable altitude. To select the target mission orbit, each of these design factors would need to be considered through the mission design process. The first step was to select the launch vehicle.

3. Launch Vehicle Selection

3.1. Selection Process

NASA LSP selected the IRIS launch vehicle following a competitive procurement process for all qualified launch service contractors (LSCs) on the NLS contract. For each LSP mission, a Launch Service Task Order (LSTO) is initiated and each LSC may bid their vehicle and launch services by submitting a mission-specific proposal. NASA then selects the vehicle for that mission based on the proposal that provides the best value to the government.

As part of the LSTO process, each proposal includes a mission solution that demonstrates how the mission requirements will be met. Typically a fixed, Earth-relative set of orbital elements provide the necessary orbit targeting to meet spacecraft mission objectives. The IRIS mission however, required only EFT compliance, which resulted in a large trade space of acceptable orbits.

The calculation of EFT, as discussed in section 2, involves propagating the various injection orbits and computing eclipse-free durations following a mission-specific definition of a valid EFT. To assist each LSC in providing a valid launch vehicle orbit solution, LSP developed a simulation tool, IRIS_EFT, which quickly evaluates the eclipse-free periods based on an initial injection orbit (refer to the Appendix for a more detailed discussion on the tool). This tool was made available to each LSC so their focus could be on developing a mission solution without the need to develop custom software to perform the EFT calculations. The availability of IRIS_EFT helped ensure consistency and accuracy from each proposal and also afforded each LSC the opportunity to use the same tool that was being used by the evaluation team to assess the validity of the proposed mission solution(s).

3.2. IRIS Launch Vehicle

At the conclusion of the LSTO process for the IRIS mission, the launch service was awarded to Orbital Sciences Corporation for the Pegasus XL launch vehicle. In their mission solution, Orbital used IRIS_EFT to demonstrate their vehicle could meet the science requirements for the 2-year mission by targeting a 620-km circular Sun-synchronous orbit at a 6 am MLT-AN. Following the vehicle selection, additional studies were performed to further optimize the mission design and launch vehicle profile.

4. IRIS Nominal Mission Design

The characteristics and capabilities of the Pegasus launch vehicle to optimize the EFT potential for the mission were critical to selecting the target orbit since the spacecraft had no propulsion system. An orbit optimization study was conducted as a joint effort by the Flight Design Working Group (FDWG), comprised of members from NASA LSP, the IRIS Project, and

Orbital. All EFT evaluations were performed using the IRIS_EFT tool. Some of the hard-coded features of the tool for the proposal phase were unlocked so that current data could be evaluated through the mission design phase, such as new solar flux predictions and mass updates.

4.1. Altitude Selection

In order to assess alternatives to the baseline 620 km circular Sun-synchronous orbit, additional design factors were needed as discriminators to quantify the benefits. Shadow durations were inversely proportional to the orbit altitude, leading to higher orbit altitudes as the desired solution. However, the target orbit would be limited by orbit lifetime, as it pertained to meeting orbital debris requirements, as well as the spacecraft radiation requirements. Altitude selection also impacts EFT for any extended science beyond the 2-year baseline mission. Therefore, the FDWG decided to perform EFT calculations for a total of 6 years on orbit.

One final design factor was that an elliptical orbit was preferred for a launch on the Pegasus launch vehicle. Pegasus is a three-stage solid motor vehicle air-launched from an L-1011 aircraft. One characteristic of a launch vehicle with a solid motor final stage is that the dispersions on the final state can be large due to the tail-off impulse variations. Typical 3-sigma dispersions for a Pegasus are +/-10 km for the insertion apse and +/-80 km for the non-insertion apse [6]. Since a low injection perigee would be subject to higher drag, the orbital decay rate would increase, degrading the Sun-synchronous orbit much quicker. Biasing the target orbit with a higher non-insertion apse would protect against this condition in the event of a low-performing vehicle.

The first step in the altitude selection was to determine the available trade space. The IRIS disposal method for orbital debris compliance was to reenter within 25 years after mission completion, but not exceeding a total of 30 years on orbit. The baseline mission is 2 years of science following a 30-day checkout period, meaning that IRIS would need to reenter from its mission orbit within 27.1 years. Figure 1 shows the range of IRIS apogee and perigee heights that resulted in an orbit lifetime of 27.1 years using the DAS Orbital Debris Software [7]. A range of launch dates was analyzed to account for the variations in the solar cycle that could occur.

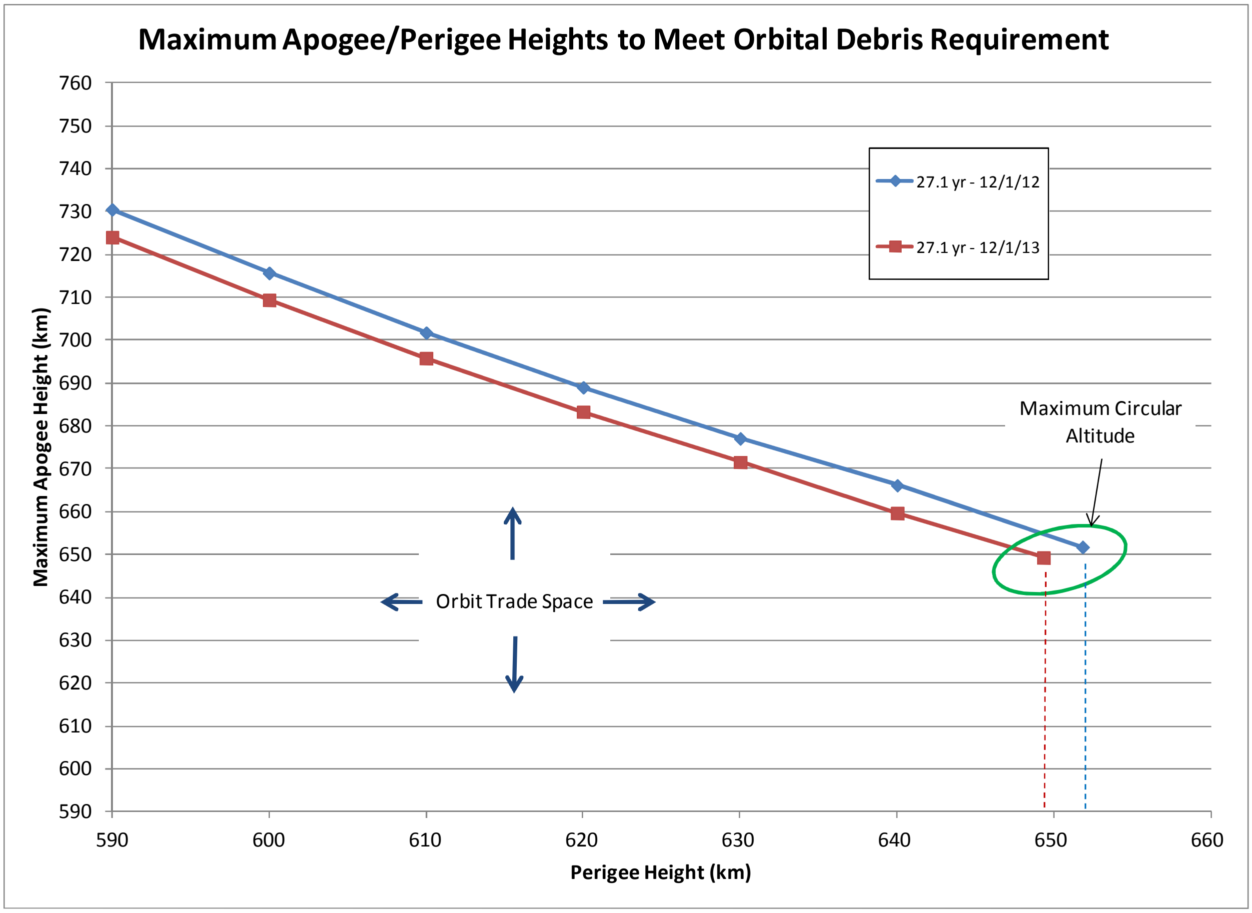

Figure 1. Maximum Apogee/Perigee Heights for Orbital Debris Compliance

The orbit trade space that satisfied the reentry disposal requirement was the region below the red line (more restrictive launch date) which contains a wide range of perigee and apogee pairs. Lower perigees would allow for significant differences in the target apses (ex: 600 x 709 km), but the low perigee targets would be exposed to higher drag. Conversely, higher nominal perigee altitudes would experience less drag but would only have small allowable differences between the insertion and non-insertion apse (ex: 640 x 660 km). The highest circular orbit that re-entered within 27.1 years was 649 km.

The IRIS FDWG collectively decided the target orbit would be 620 x 670 km, based on engineering judgment considering all the aforementioned factors. This target orbit increased the semimajor axis sufficiently to reduce drag impacts from the 620 km circular baseline. The predicted reentry time was 21.6 years, which allowed for some potential increase in the decay time should future solar flux predictions result in less decay. The 50 km difference between the insertion and non-insertion apse was believed to be the proper balance in targeting based on typical 3-sigma insertion errors. Orbital also performed an initial examination that showed the maximum dispersed semimajor axis was below 7078 km, meeting the IRIS radiation requirements. This analysis also showed the target orbit was within the performance capability of the launch vehicle, while maintaining the appropriate reserves for contingencies.

4.2. Inclination and MLT-AN Biasing

Once the target orbit altitude was determined, a parametric study was performed to determine the planar targets, specifically the inclination and orientation (defined as MLT-AN). For a 620 x 670 km orbit, the corresponding Sun-synchronous inclination is computed as 97.965 degrees. The pairing of these parameters would maintain the MLT-AN at injection in an ideal environment. However, orbital decay due to drag lowers the semimajor axis and disrupts the altitude/inclination pairing, causing MLT-AN drift to occur. A concept often utilized with Sun-synchronous orbits to counter this is inclination biasing [8], which is a planned offset from the Sun-synchronous inclination to control the drift rate and maximize the time between MLT-AN stationkeeping maneuvers. Inclination biasing could better pair the average semimajor axis and inclination that would be experienced over the mission lifetime as it decays, thus better preserving a near Sun-synchronous state.

In addition to inclination biasing, MLT-AN biasing from the initial 6 am target was also examined. Figure 2 shows the results of the parametric scan for a +/-1.0 degree inclination variation and +/-30 minutes in MLT-AN variation and the resulting impacts to EFT times for years 1 and 2.

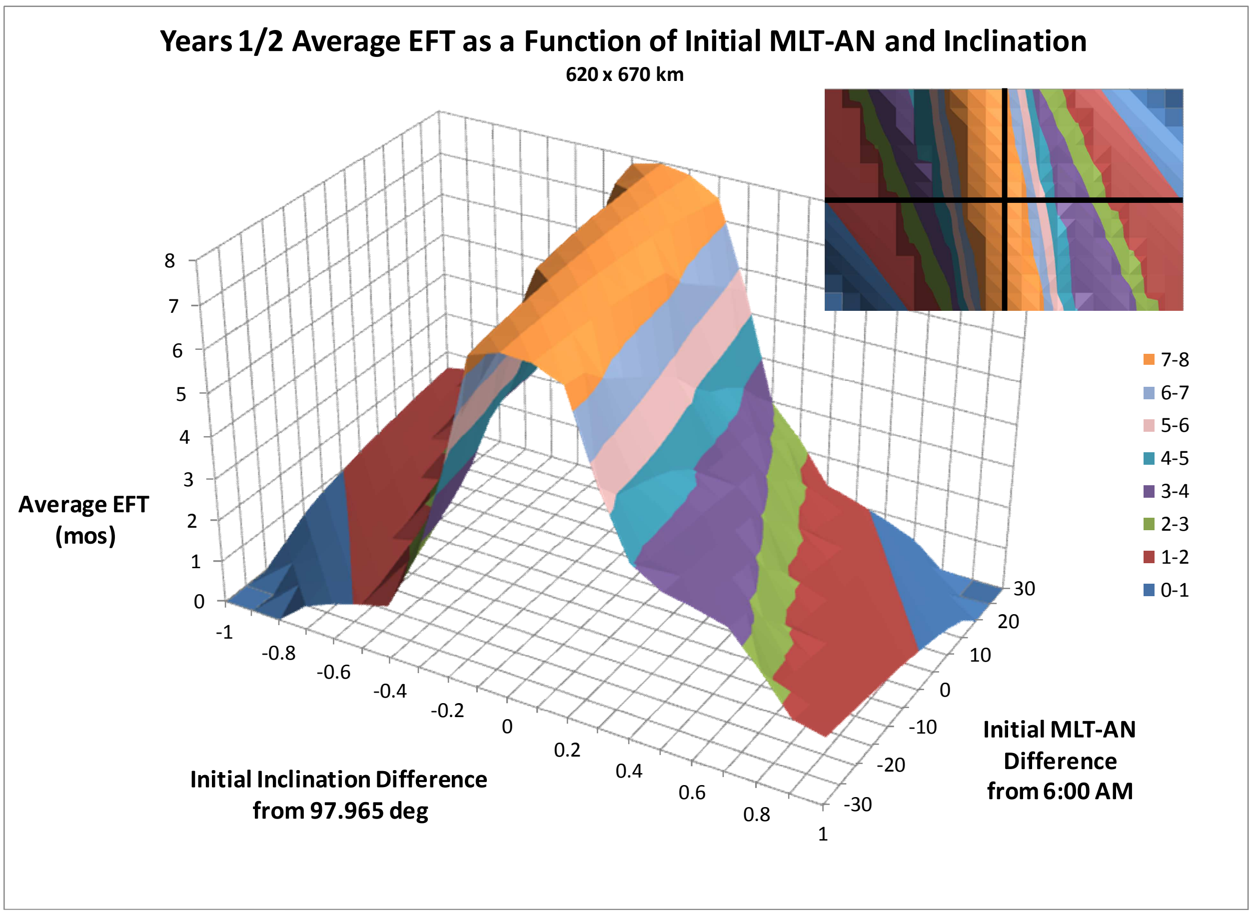

Figure 2. Effect on Biasing on Year 1 and 2 EFTs

This surface plot shows the resulting Year 1 and 2 EFT average as a function of initial inclination and MLT-AN. To aide in the visualization, the upper right portion of the plot contains a two-dimensional top-down view of the data. The plot shows the resulting EFTs are highly dependent on the initial inclination selection, but fairly insensitive to the large range of MLT-ANs that were examined.

Figure 3 shows this same data with a smaller range of inclination differences (+/-0.2 deg). As the plot shows, the orange region is shifted to the left of center, indicating that a negative inclination bias does improve the EFT predictions.

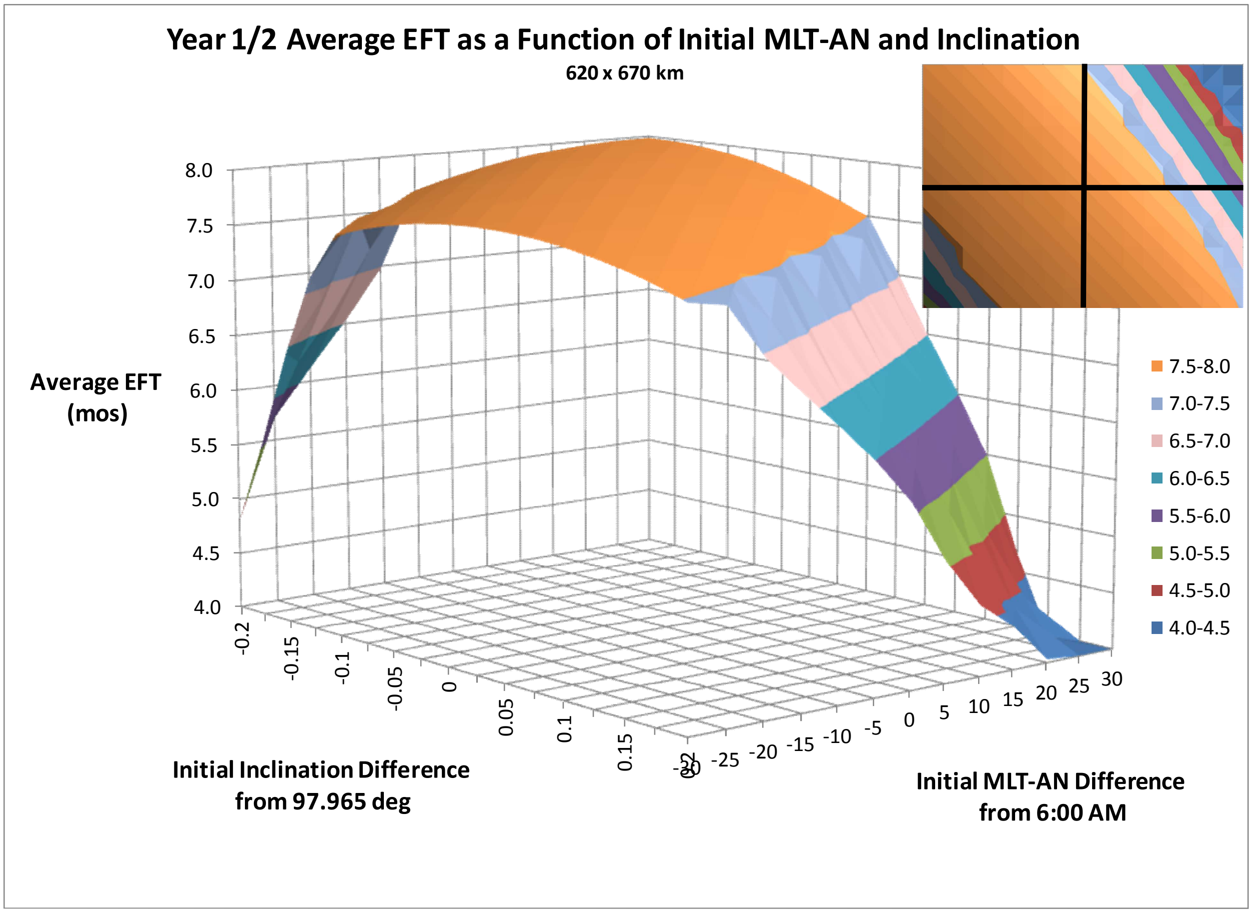

Figure 3. Effect on Biasing on Year 1 and 2 EFTs, Enlarged

Since the secondary goal was to achieve extended mission science, these parametric studies were also performed assuming a mission lasting up to 6 years. Figure 4 shows the results of these scans. Note that the Z-axis scale for EFT is expressed in total EFT months over years 1-6.

The results shown in Figure 4 further demonstrate there are clear advantages to inclination biasing for a mission extension up to 6 years. This trend was expected since drag has more of an impact on the semimajor axis. The FDWG used these results to set the target inclination to 97.89 degrees, a bias of -0.075.

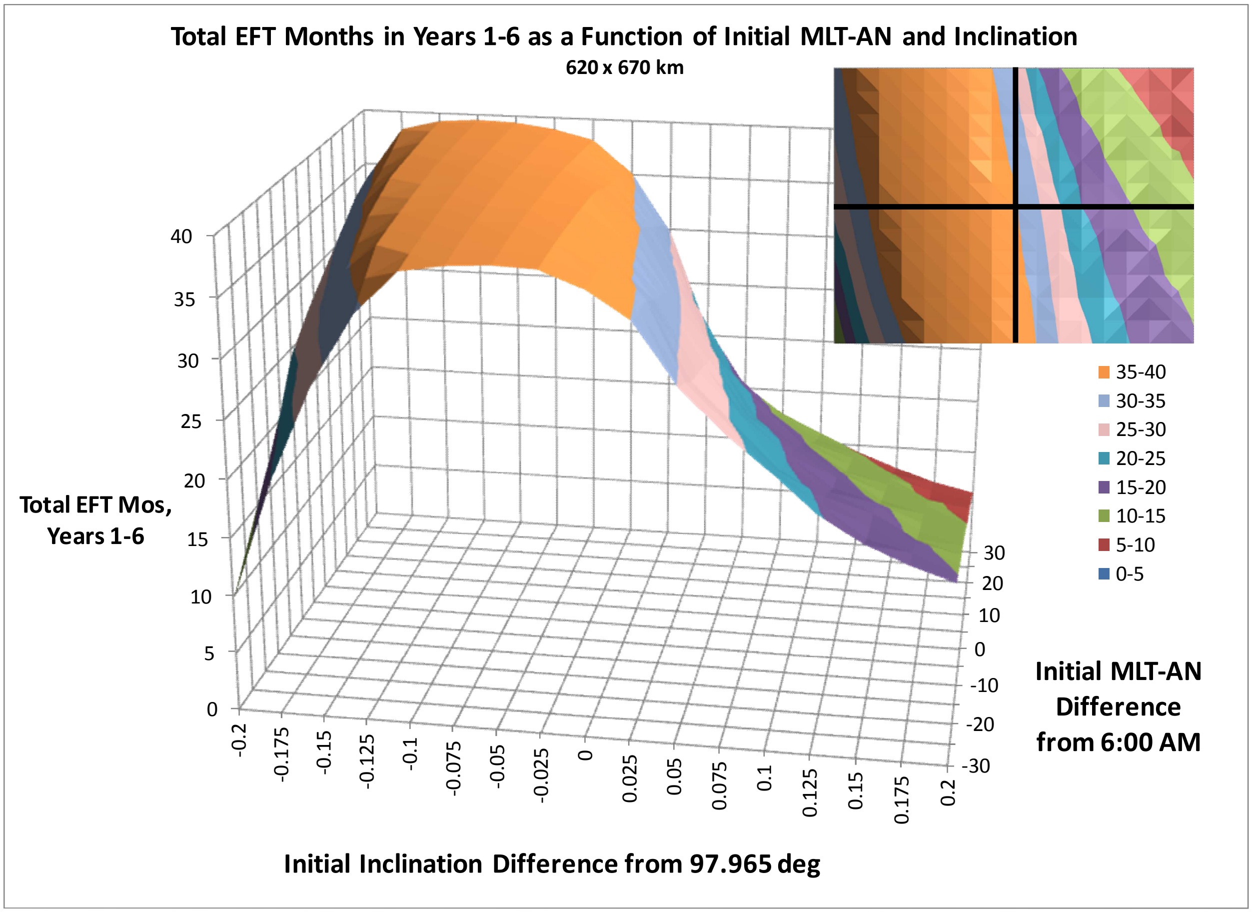

Figure 4. Effect on Biasing over Years 1 through 6

Figure 4 also shows only minor advantages (near zero slope) for a MLT-AN bias, with slight gains if the MLT-AN target was set with a +5 minute bias. Consequently, the launch window duration was set at 5 minutes, which balanced the IRIS project desire to minimize the MLT-AN variation at injection against the minimum window duration for a Pegasus launch. Therefore, the FDWG set the final target MLT-AN to 06:00 – 06:05 am, corresponding to a 5 minute window.

The final set of nominal orbit targets resulting from these trade studies was as follows:

Table 1. Nominal IRIS Orbit Targets

Orbit Target Parameter	Nominal Target Value
Perigee and Apogee	620 x 670 km
Inclination	97.89 degrees
MLT-AN	06:00 - 06:05 am

The detailed data for Figures 3 and 4 are listed in Tables 2 and 3, respectively.

Table 2. Effect on Biasing on Year 1 and 2 EFTs

								6:00 AM						
		-30	-25	-20	-15	-10	-5	0	5	10	15	20	25	30
	-0.2	4.790	5.702	6.122	6.527	6.972	7.673	7.760	7.834	7.892	7.935	7.965	7.982	7.985
	-0.175	5.575	6.274	6.703	7.230	7.679	7.766	7.839	7.896	7.940	7.971	7.988	7.990	7.981
	-0.15	6.429	6.897	7.578	7.679	7.767	7.838	7.895	7.941	7.971	7.989	7.992	7.984	7.962
	-0.125	7.156	7.574	7.676	7.762	7.835	7.891	7.935	7.965	7.984	7.989	7.981	7.960	7.924
	-0.1	7.563	7.667	7.754	7.826	7.883	7.926	7.956	7.975	7.981	7.973	7.953	7.919	7.873
	-0.075	7.651	7.739	7.810	7.869	7.913	7.945	7.963	7.969	7.961	7.942	7.909	7.862	7.802
	-0.05	7.720	7.790	7.851	7.896	7.928	7.945	7.952	7.945	7.925	7.894	7.849	7.789	7.716
	-0.025	7.769	7.828	7.873	7.906	7.925	7.931	7.925	7.905	7.873	7.830	7.771	7.700	7.614
97.965	0	7.800	7.847	7.878	7.898	7.906	7.900	7.883	7.851	7.808	7.750	7.679	7.593	7.491
	0.025	7.816	7.848	7.869	7.876	7.872	7.853	7.823	7.780	7.722	7.652	7.567	7.467	7.016
	0.05	7.814	7.836	7.844	7.839	7.822	7.792	7.749	7.692	7.621	7.538	7.438	6.861	6.499
	0.075	7.798	7.806	7.803	7.786	7.756	7.713	7.658	7.588	7.504	7.107	6.717	6.373	6.021
	0.1	7.766	7.762	7.746	7.717	7.676	7.619	7.549	7.465	6.930	6.578	6.245	5.887	5.489
	0.125	7.718	7.703	7.674	7.633	7.578	7.506	7.165	6.773	6.444	6.105	5.741	5.308	4.669
	0.15	7.657	7.627	7.587	7.530	7.462	6.965	6.622	6.301	5.956	5.569	5.067	4.451	3.991
	0.175	7.576	7.535	7.479	7.194	6.795	6.474	6.150	5.791	5.361	4.652	4.416	4.105	3.183
	0.2	7.481	7.426	6.969	6.634	6.320	5.986	5.601	5.047	4.618	4.372	4.028	3.454	2.990

Table 3. Effect on Biasing over Years 1 through 6

								6:00 AM						
		-30	-25	-20	-15	-10	-5	0	5	10	15	20	25	30
	-0.2	9.58	12.32	15.03	17.65	20.63	24.28	26.01	27.49	28.84	30.11	31.44	32.74	33.43
	-0.175	20.14	23.14	25.37	27.66	29.71	30.99	32.23	33.55	34.95	35.71	36.48	37.38	38.37
	-0.15	28.30	30.42	33.04	34.58	35.73	37.10	37.82	38.50	38.74	38.94	39.09	39.20	39.27
	-0.125	34.25	36.16	37.81	38.19	38.51	38.78	39.02	39.20	39.35	39.46	39.52	39.54	39.52
	-0.1	37.92	38.29	38.61	38.88	39.10	39.29	39.43	39.53	39.59	39.60	39.58	39.51	39.41
	-0.075	38.52	38.79	39.02	39.19	39.33	39.42	39.47	39.49	39.46	39.40	39.29	39.13	38.94
	-0.05	38.75	38.92	39.06	39.15	39.20	39.21	39.18	39.10	38.99	38.84	38.64	38.39	38.10
	-0.025	38.62	38.70	38.74	38.75	38.71	38.63	38.51	38.23	37.61	37.13	36.64	36.13	35.12
97.965	0	37.26	37.08	36.88	36.64	36.37	36.06	35.70	35.13	34.07	33.21	32.32	31.38	29.78
	0.025	34.95	34.69	34.37	33.36	32.53	31.72	30.83	29.85	28.68	27.00	24.34	22.70	20.35
	0.05	29.87	29.12	27.92	27.41	26.82	26.15	25.34	23.33	21.60	20.30	18.74	16.88	15.22
	0.075	26.37	25.68	24.46	24.00	22.73	21.31	19.95	19.17	18.31	16.62	14.88	13.86	12.68
	0.1	23.49	22.37	20.94	20.21	19.48	18.68	17.66	16.53	15.16	14.09	12.76	11.77	10.98
	0.125	20.34	19.67	18.92	17.97	16.90	16.48	15.48	14.27	12.89	12.21	11.48	10.62	9.34
	0.15	18.15	17.12	16.79	16.39	15.90	14.35	13.24	12.60	11.91	11.14	10.13	8.90	7.98
	0.175	16.64	16.25	15.73	14.39	13.59	12.95	12.30	11.58	10.72	9.30	8.83	8.21	6.37
	0.2	15.47	14.85	13.94	13.27	12.64	11.97	11.20	10.09	9.24	8.74	8.06	6.91	5.98

A comparison of the EFT potentials between the newly optimized nominal elliptical orbit and the baseline 620 km circular orbit is shown in Table 4.

Table 4. EFT Comparison – Baseline vs. New

EFT Results	620 km Circular	620 x 670 km
EFT Year 1, Year 2, (Ave)	7.87, 7.81 (7.84)	8.04, 7.90, (7.97)
EFT Total Months Y1-Y6 (Ave)	45.77 (7.63)	47.16 (7.86)

The new orbit showed EFT improvements in both the baseline 2-year mission and for an extended mission, assuming a nominal trajectory. The improvements shown in Table 4 may seem modest, but greater benefits lie in the EFT improvements resulting from an off-nominal orbit injection. For example, if the insertion apse was 50 km low, the original orbit would have achieved a 570 x 620 km orbit versus a 620 x 620 km orbit for the new targets and would be subject to much higher decay rates and lower EFTs.

5. IRIS Dispersed Orbit Effects

Injecting IRIS into the nominal target orbit resulting from the trade studies would exceed all mission requirements for the baseline mission and for several years of an extended mission. In reality, however, there are always variations in flight that result in a slightly different achieved orbit. Environmental conditions, actual motor performance, and minor misalignments are all examples of factors that impact the day-of-launch flight leading to injection dispersions. Monte Carlo simulations are performed to quantify the variations that may be encountered on launch day as well as confirm the mission will still be successful with up to at least 3-sigma vehicle performance. Additionally, the Pegasus launch vehicle has the capability to utilize different guidance schemes to improve the variations on target parameters of primary interest to meeting mission requirements. The different strategies were evaluated for IRIS to determine which method offered the greatest potential for mission success when factoring in vehicle injection errors.

5.1. Pegasus Guidance Options

The air-launched Pegasus launch vehicle consists of three solid rocket motor stages. An optional liquid propellant fourth stage, the Hydrazine Auxiliary Propulsion System (HAPS), is available for orbit trim and/or mass-to-orbit enhancement, but cost constraints on many Pegasus-class payloads do not allow for this benefit. However, the guidance capability on the standard, three-stage Pegasus launch vehicle is typically sufficiently tunable to allow for mission specific orbit targeting optimization.

The Pegasus guidance system can manage the energy state of the launch vehicle basically in one of three ways: manage in-plane orbit energy only, manage out-of-plane orbit energy only or manage a compromise of both in-plane and out-of-plane orbit energies. These varieties of guidance options result in deterministically different injection state dispersions (i.e. minimal inclination errors, minimal altitude errors, minimal RAAN errors, compromise of all minimal errors, etc.). The objective for a given mission is choosing an energy management strategy that results in the optimal injection state dispersions for the spacecraft.

The first step in determining the optimal guidance solution for the IRIS mission was to perform a 6DOF trajectory Monte Carlo trade study to test multiple orbit targeting strategies. The results of these analyses would provide the corresponding dispersed injection states to use for further analysis. The second step was to perform EFT Monte Carlo assessments for each of the guidance schemes and determine which yielded the optimal science return. The results of the EFT analyses were reviewed by the IRIS FDWG team and an optimal guidance scheme was chosen to support the IRIS mission.

5.2. Trade Study and Selection

There were two guidance strategies of primary interest to IRIS, since the focus of the final orbit was to achieve significant EFT periods by inserting into a near Sun-synchronous orbit. Focusing on out-of-plane energy scrubbing tends to better constrain in-plane (altitude) errors, while in-plane energy scrubbing generally provides better out-of-plane (inclination) errors. The pairing of both altitude (semimajor axis) and inclination to achieve a Sun-synchronous orbit meant there was sensitivity to both error types, so each method was evaluated to determine which strategy would yield better overall EFT results.

Figures 5 and 6 show a comparison of the guidance scheme dispersion results from the 1000 case Monte Carlo 6DOF runs. The out-of-plane scrubbing resulted in an evenly dispersed set of injection states when plotting in-plane versus out-of-plane errors. Conversely, in-plane energy scrubbing minimized inclination errors and showed a large variation in the resulting semimajor axis. Injection errors alone do not necessarily result in a negative impact on EFTs; combinations of in-plane and out-of-plane errors can provide the proper Sun-synchronous pairing of semimajor axis and inclination to produce excellent EFT results.

The Sun-synchronous line (red line) is overlaid on the dispersion plots, which is the ideal pairing of semimajor axis and inclination. Note the red line does not pass through the origin as the IRIS target inclination has been biased (Section 4.2.) The spread of dispersed cases for the in-plane scrubbing method remains closest to the ideal Sun-synchronous line, indicating this strategy has a higher potential for longer EFT periods over the IRIS mission lifetime.

Figure 7 shows the comparison of the two guidance strategies on the resulting EFTs for the first two years of the mission. The statistics are plotted as the percent of cases that exceed various thresholds of EFT duration. The results here show the in-plane strategy that limits inclination error does slightly outperform the results from out-of-plane energy scrubbing, which is consistent with the information in Figures 5 and 6.

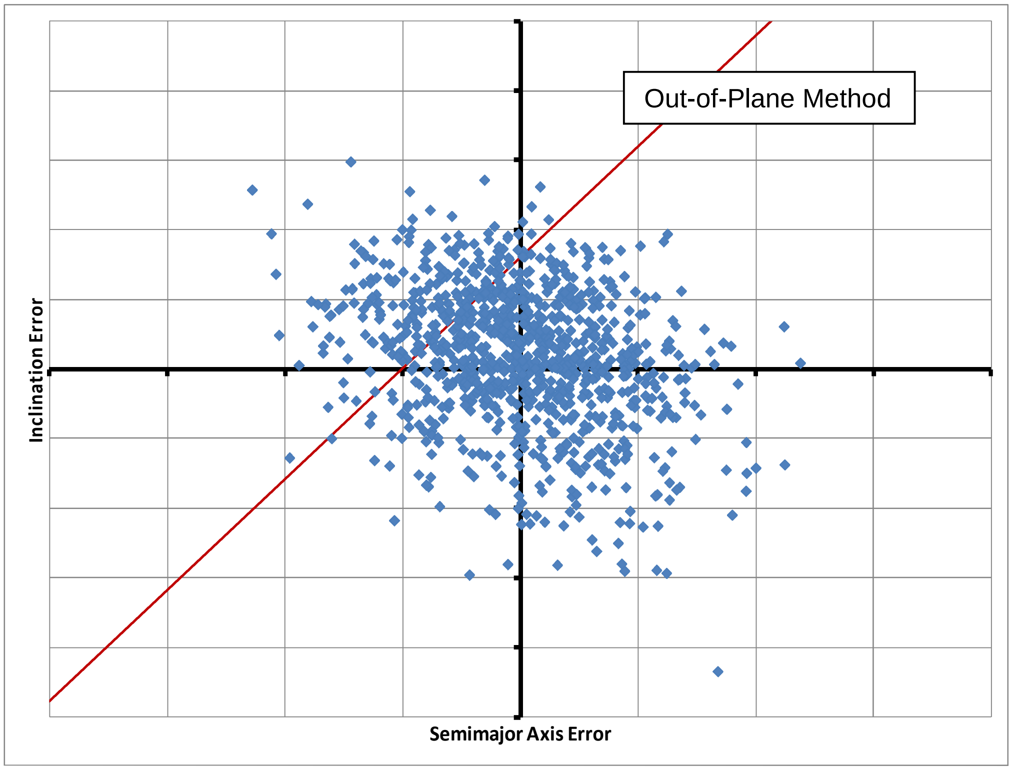

Figure 5. Dispersion Characteristics for Out-of-Plane Energy Scrubbing

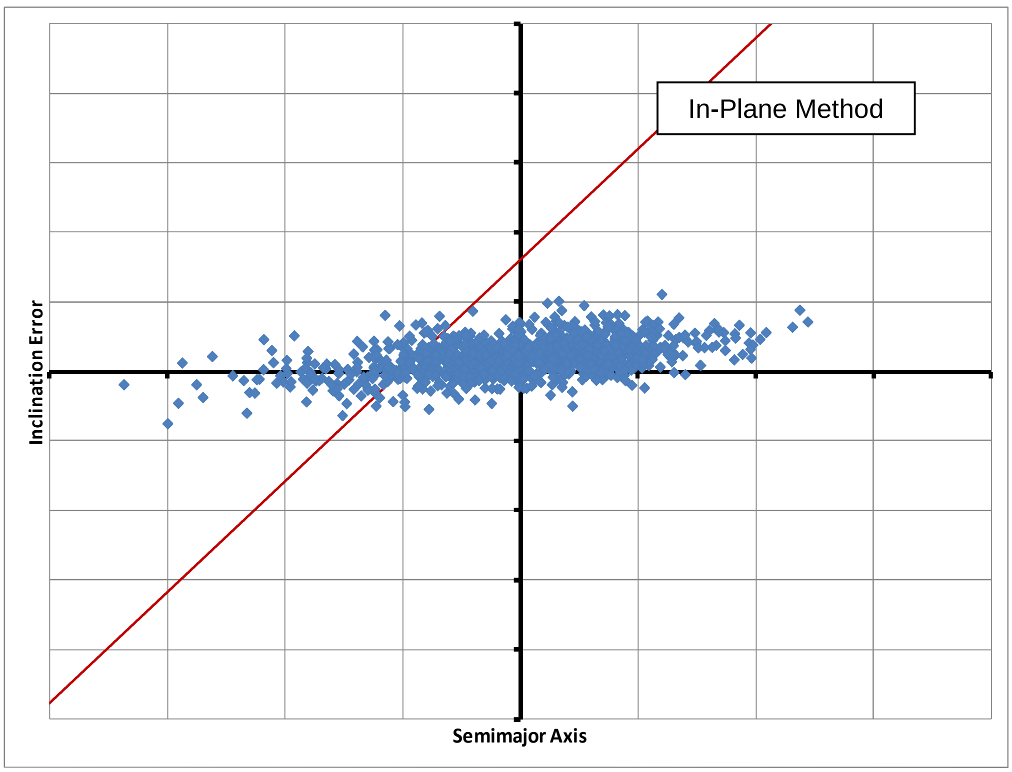

Figure 6. Dispersion Characteristics for In-Plane Energy Scrubbing

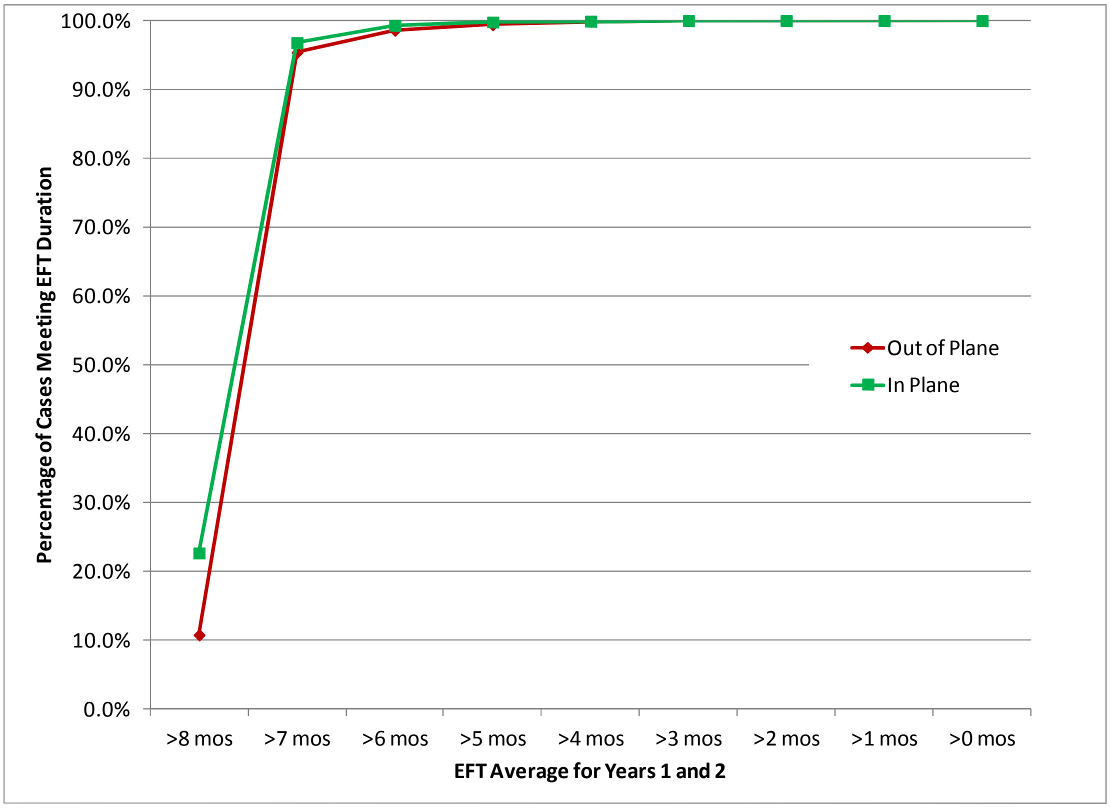

Figure 7. EFT Results for Guidance Strategy Options, Years 1 and 2

For an extended mission, the in-plane strategy clearly increases the likelihood of continued long durations of annual EFTs. Figure 8 plots percentage of cases exceeding 7 months of EFT over the six-year period. By year four, almost 70% of the in-plane cases still exceed the 7-month target, compared to only 57% of the out-of-plane cases. The difference continues to increase through years four and five. Figure 9 similarly plots the percentage of cases exceeding 5 months of EFT as a function of years from launch. The in-plane method continues to provide higher results. Based on these compilation graphs of the Monte Carlo results on EFTs, minimizing inclination provided the optimal IRIS science returns for both the baseline and extended mission and was consequently implemented on the Pegasus vehicle.

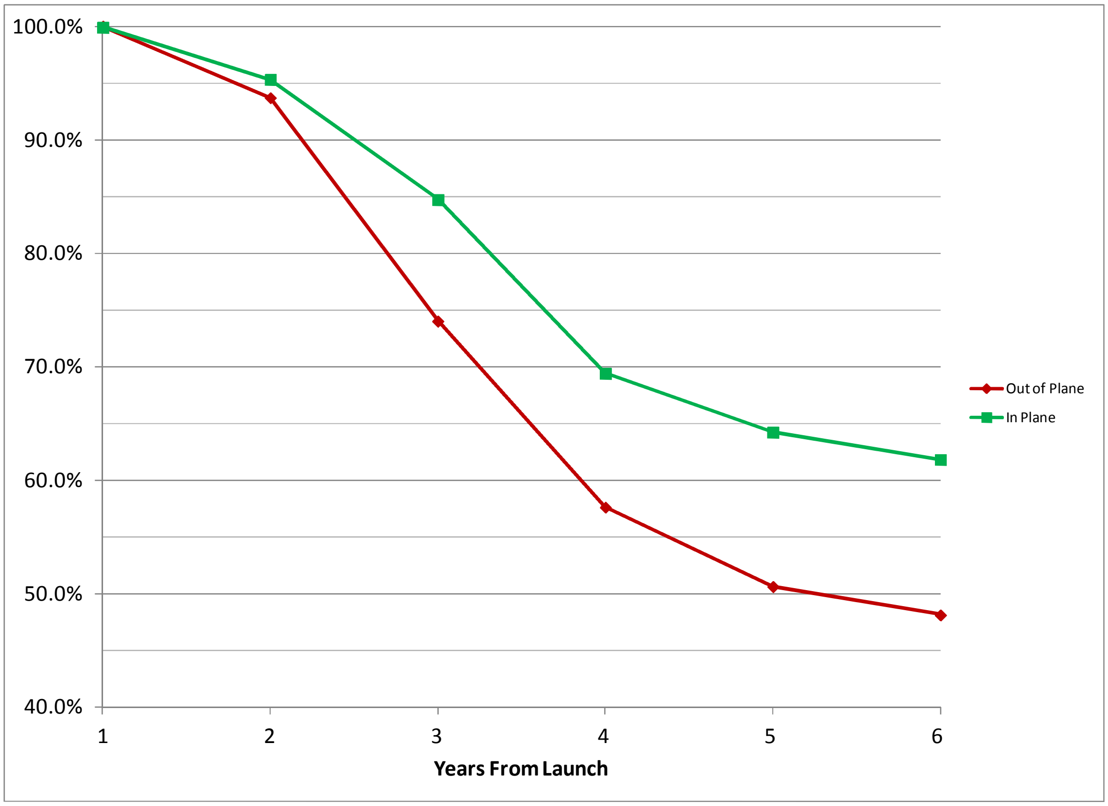

Figure 8. Percentage of Cases that Exceed 7 months of EFT by Year on Orbit

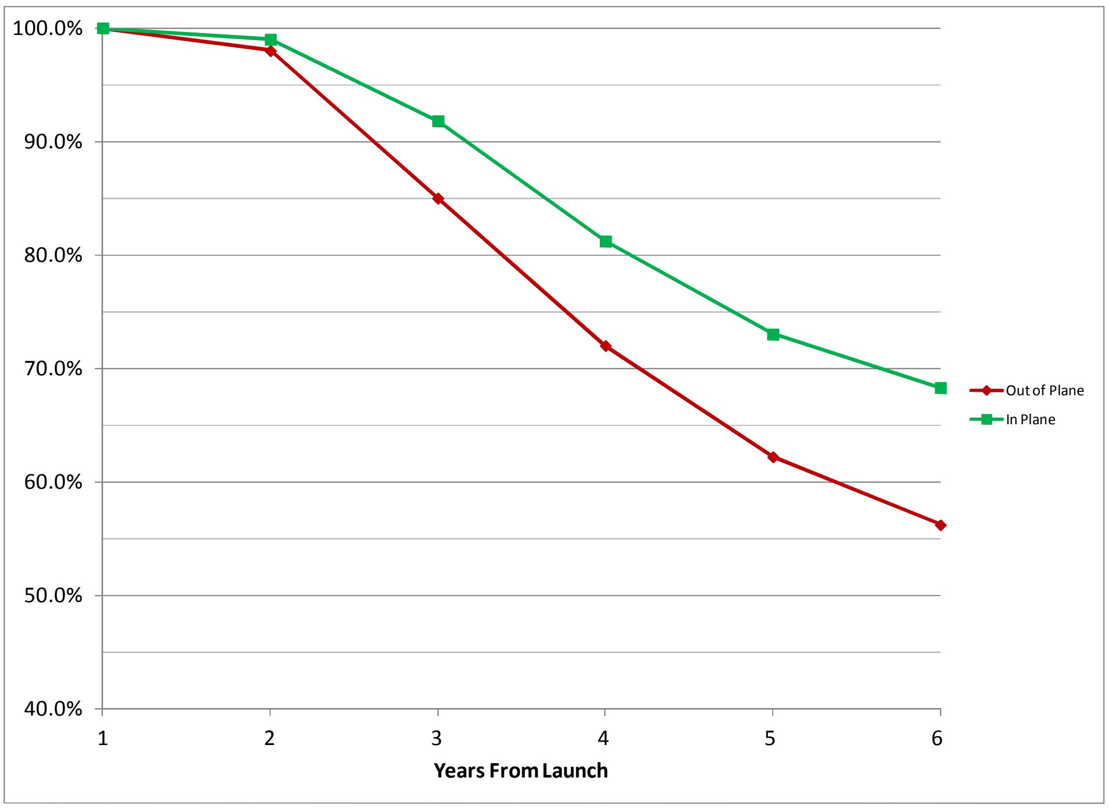

Figure 9. Percentage of Cases that Exceed 5 months of EFT by Year on Orbit

6. Launch Results

IRIS was successfully launched on June 27, 2013 and inserted into a nominal orbit. All target parameters were within 1-sigma of the predicted results. The orbit state at injection is listed in Table 5. The achieved perigee and apogee heights were slightly higher than the target orbit. However, the inclination for IRIS was also slightly higher, which is the ideal combination of the dispersions to produce longer EFT periods.

Table 5. Nominal IRIS Orbit Targets and Achieved Orbit

Orbit Target Parameter	Nominal Target	Achieved Orbit
Perigee and Apogee	620 x 670 km	623.02 x 672.89 km
Inclination	97.89 degrees	97.897 degrees
MLT-AN	06:02:30 am	06:02:35 am

The IRIS_EFT tool was run based on the actual achieved orbit. Table 6 presents the results of the EFT durations that were predicted over a 6-year mission. The expectation is that IRIS will achieve greater than 7 months of EFT duration for the baseline 2-year mission and for at least an additional 4 years, providing the potential for extended science.

Table 6. Predicted Annual EFT Durations

Year 1	Year 2	Year 3	Year 4	Year 5	Year 6
7.92	7.85	7.93	7.89	7.77	7.85

7. Summary and Conclusions

NASA's IRIS mission was driven by the requirement to perform solar observations and was not constrained to a particular mission orbit. An orbit optimization study was required to select the target orbit that would produce the mission science, while also satisfying other IRIS design constraints, such as orbital debris and radiation effects. This study was greatly enhanced by incorporating the performance characteristics of the launch vehicle. For LSP-procured missions, a joint LSP-spacecraft-LSC team (the FDWG) is typically formed to address issues that affect the mission design throughout the integration cycle. For IRIS, the FDWG evaluated the spacecraft and launch vehicle as a combined system to determine the mission solution that yielded optimal results. This study included both the nominal target orbit selection and guidance scheme methodology to manage dispersions that maximized the probability of mission success for both the 2-year baseline and 4-year extended mission. Following its successful launch, IRIS is expected to achieve at least 6 years of quality science observation periods, greatly exceeding the baseline mission requirements.

8. References

[1] Lockheed Martin Space Systems Company "IRIS Launch Vehicle Interface Requirements Document", A101-RQ-09-0267 Rev: A, 1 February 2011.

[2] NASA Procedural Requirement "NASA Procedural Requirements for Limiting Orbital Debris (w/ Change 1 - 5/14/09)", NPR 8715.6A, 14 May 2009.

[3] Eagle, C.D. “User’s Manual for IRIS_EFT Program”, Version 1.0, January 2010.

[4] Fehlberg, E., “New One-Step Integration Methods of Higher-Order Accuracy Applied to Some Problems in Celestial Mechanics”, NASA TR R-248, October 1966.

[5] Brent, R. “Algorithms for Minimization without Derivatives”, Dover Publications, 2002.

[6] Orbital Sciences Corporation “Pegasus User’s Guide”, Release 7.0, April 2010.

[7] NASA Orbital Debris Program Office, *NASA Debris Assessment Software*, Version 2.0.2, 16 April 2012.

[8] Dell, G., Hametz, M., Noonan, P., Newman, L., Folta, D., and Bristow, J. “EOS AM-1 and EO-1 Support Using FreeFlyer™ and AutoCon™”, AAS/AIAA Astrodynamics Specialist Conference, Boston, MA, USA, 1998.

APPENDIX A

IRIS_EFT is a scientific simulation which can be used to perform an EFT assessment of candidate IRIS mission orbits [3]. The executable program was created using the Intel Visual Fortran compiler to run on Windows-based computers. IRIS_EFT was developed by C.D. Eagle (a.i. solutions, Inc.) for use by the LSC teams during the mission proposal phase as well as the IRIS FDWG to perform mission design trade studies. IRIS_EFT received a software award by NASA KSC for its innovation in the LSTO process and subsequent optimization studies.

EFT is defined to be those time intervals longer than one day during which the IRIS spacecraft is not in the Earth’s shadow. IRIS_EFT implements a special perturbation technique which numerically integrates the vector system of second-order, nonlinear differential equations of motion of a spacecraft given by

$$\vec{a}\left(\vec{r},\vec{v},t\right)=\ddot{\vec{r}}\left(\vec{r},\dot{\vec{r}},t\right)=\vec{a}_g\left(\vec{r}\right)+\vec{a}_d\left(\vec{r},\vec{v},t\right)+\vec{a}_{sm}\left(\vec{r},t\right) \tag{1}$$

where

$t =$ simulation time

$\vec{r} =$ inertial position vector of the spacecraft

$\vec{v} =$ inertial velocity vector of the spacecraft

$\vec{a}_g =$ acceleration vector due to Earth gravity

$\vec{a}_d =$ acceleration vector due to atmospheric drag

$\vec{a}_{sm} =$ acceleration vector due to the Sun and Moon

This system of three second order vector differential equations is converted to a system of six first-order differential equations using the method of order reduction. The first-order equations of motion are solved using a Runge-Kutta-Fehlberg (RKF78) numerical method, a variable step size algorithm with truncation error control. [4]

The software first read in an injection epoch and orbital element data set from an input file. Multiple injection states could be input and were processed sequentially. The program first propagated the spacecraft for the 30-day checkout period and then propagated for two more years (the baseline mission), while the EFT characteristics were computed. At the end of the propagation, the EFT statistics were output to both the screen and to an output text file.

The shadow calculations used a spherical Earth shape and cylindrical shadow model. The algorithm also assumed the Earth radius was increased by 250 kilometers, per spacecraft requirements. The IRIS_EFT software used Brent's root-finding method to search for shadow boundary crossings during the orbit evolution [5].

During the root-finding calculations, the scalar objective function is given by:

$$f = \psi - \theta = \cos^{-1}\left(-\mathbf{U}_{sun} \bullet \mathbf{U}_{sat}\right) - \theta \tag{2}$$

where $\mathbf{U}_{sun}$ and $\mathbf{U}_{sat}$ are the Earth-centered-inertial (ECI) unit pointing vectors of the Sun and spacecraft, respectively, and θ is the cylinder shadow angle. Shadow entry and exit conditions are a function of the angles between the anti-sun direction, the shadow boundary, and the spacecraft's location.

If we represent the shadow as a cylinder, the shadow angle at the spacecraft's current location is given by $\theta = \sin^{-1}\left(R_e / R_{sat}\right)$ where R_e is the "augmented" radius of the Earth and R_{sat} is the geocentric distance of the IRIS spacecraft. Whenever the IRIS spacecraft crossed the shadow boundary, the scalar value of the objective function changed sign. The orbit propagation software monitored the objective function and invoked Brent's method when this occurred, locating the actual time of the boundary crossing.

To maintain consistency in the EFT calculations from each LSC proposal, several key ground rules were enforced during the EFT computer analysis by hardcoding several parameters in the executable code provided to the LSCs. These ground rules included:

- Mission orbit injection on December 1, 2012
- Thirty day mission check-out before two year EFT assessment began
- Valid eclipse free periods must be at least one day in duration
- Constant spacecraft mass of 142.5 kilograms
- Constant drag coefficient of 2.2
- Constant drag reference area of 1.55 square meters
- World Geodetic Survey 1984 (WGS84) Earth gravity Model of order 8 and degree 4
- Jacchia-Roberts atmospheric density model with 95 percentile solar activity extracted from the MSFC Solar Activity November 2009 bulletin
- Cylindrical shadow with size = Earth equatorial radius + 250 kilometers
- JPL DE405 solar and lunar ephemeris