IEEE P335E
A Collaborative Optimization Approach to Design and Deployment of a Space Based Infrared System Constellation

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in the proceedings of:

2000 IEEE Aerospace Conference

18–25 March 2000 Big Sky, Montana

A Collaborative Optimization Approach to Design and Deployment of a Space Based Infrared System Constellation[1]

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Abstract—Collaborative optimization, as a design architecture, has been used successfully in solving large-scale multidisciplinary optimization problems related to aircraft and space vehicle designs. The study presented in this paper attempts to demonstrate yet another application for this architecture, i.e., to satellite constellation designs. As an example, it is implemented for the design and deployment problem of a space based infrared system placed at low earth orbit.

Preliminary results on a simplified problem are presented as a proof-of-concept. The constellation configuration is fixed to be a 28/4/2 Walker delta pattern (four planes with seven satellites per plane and relative phasing of two). The mission orbit, spacecraft design, and deployment strategy are varied to determine the optimal system (i.e., one with the minimum cost to deployment).

Problem reformulation required by the collaborative optimization architecture and some implementation issues are discussed. The three analysis tools used in this study are also described in this paper. The constellation design module finds orbit parameters and constellation configurations that satisfy the coverage requirements. The spacecraft model performs the payload and bus design. Finally, the launch manifest module finds the best strategy, in terms of total launch cost, to deploy the constellation system.

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1. INTRODUCTION

Current Trends

In recent years, there has been an explosion of interest in all sectors of the aerospace community for utilizing Low Earth Orbit (LEO) satellite architectures. The growing demand for global wireless communication systems has motivated several companies to develop LEO constellations to work with ground-based systems to support voice, data and fax transmissions.

The military sector has also shown interest in utilizing LEO satellite architectures for various missions, such as surveillance and space-based defense systems. Certain applications may require sensitivity and resolution that is only feasible at lower altitudes.

Another major trend in the aerospace community results from a paradigm shift that emphasizes economic viability rather than maximization of system performance. No longer is *the best system at any cost* an acceptable design philosophy. Rather, *faster, better, cheaper* is the motto of the day, pushing for minimum cost spacecraft that meets performance and schedule constraints.

Current Methods

The design of satellite constellation architecture is a complex iterative process, involving many parameters and variables, both discrete and continuous. The constraints involved and the multidisciplinary nature of the problem also add to its complexity. For certain missions, the best constellation (i.e. the constellation with the minimum life-cycle cost) may be the one that requires the fewest number of satellites to provide the specified total coverage. For others, launch vehicle capabilities may favor a constellation configuration with more (perhaps smaller) satellites at lower altitudes. When considerations such as fold of coverage (required coverage overlap) or cross-linking between satellites (connectivity requirements for passing information) are taken into account, the solution may look different yet [1]. Optimizing the overall system is therefore not a straightforward task.

The problem is characterized by three contributing analyses, represented in the Design Structure Matrix (DSM, [2]) format in Figure 1. The constellation design module involves trade studies to find the best orbital parameters and constellation configuration to provide the coverage required. The spacecraft modeling tool estimates mass, power, and costs for the payload that meets the resolution and sensitivity requirements and for the spacecraft bus to support that payload. Finally, the launch manifest module finds the best strategy to deploy the entire constellation (including on-orbit spares, if applicable).

The common satellite constellation design practice [3] begins by establishing several alternate concepts that can meet the mission objectives and requirements. Here, possible constellation configurations, payload characteristics and launch vehicle options are identified. These concepts are further explored and refined within an iterative process employing several trade analyses, such as one that relates payload instrument capability with mission orbit. At lower altitudes, smaller sensors (and therefore smaller and cheaper spacecraft) can be used to meet the resolution and sensitivity specifications, but more spacecraft are needed to satisfy the overall coverage requirement.

Trade studies involving payload and spacecraft bus designs are also conducted. The bus must provide power, command and data handling, communication, thermal and attitude controls, pointing and mapping to ensure a successful mission over the design lifetime. Thus the payload requirements mainly drive the sizing of the spacecraft. The objective here is to identify possible combinations of structure materials, attitude control approach, solar array types, etc., to satisfy the payload requirements (including reliability) at the lowest cost. Payload fairing volume constraints and maximum g-loading to be expected during launch are also taken into account.

Having specified through these trades a constellation configuration and a set of orbit parameters, and having computed the spacecraft unit mass, different combinations of launch vehicles with inherently varied capabilities, availability and costs are explored. The winning strategy (that which yields the minimum total launch cost) is evaluated to identify changes to the orbit or constellation characteristics that can possibly lead to savings in the resulting launch cost, a major component of the overall system cost. For example, perhaps by lowering the altitude slightly, a cheaper vehicle with lower capability can be used instead. Information such as this is fed back to the beginning of the process. The iteration is continued until convergence is found, yielding an optimal constellation configuration, orbit parameters, spacecraft design and deployment strategy for this particular concept.

Motivation

Given the increasing interest in LEO constellations and the shift towards design-to-cost, the work presented in this paper attempts to investigate implementing a Multidisciplinary Design Optimization (MDO) technique to improve the current process described above. Specifically, Collaborative Optimization (CO, [4]) seems promising and can potentially aid system analysts to explore the design space in a systematic manner, without neglecting cost considerations.

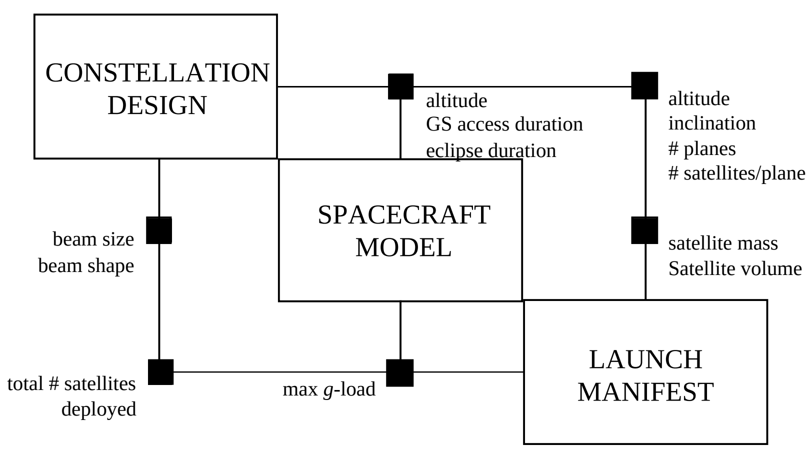

Figure 1. DSM representing the coupling between the satellite design modules.

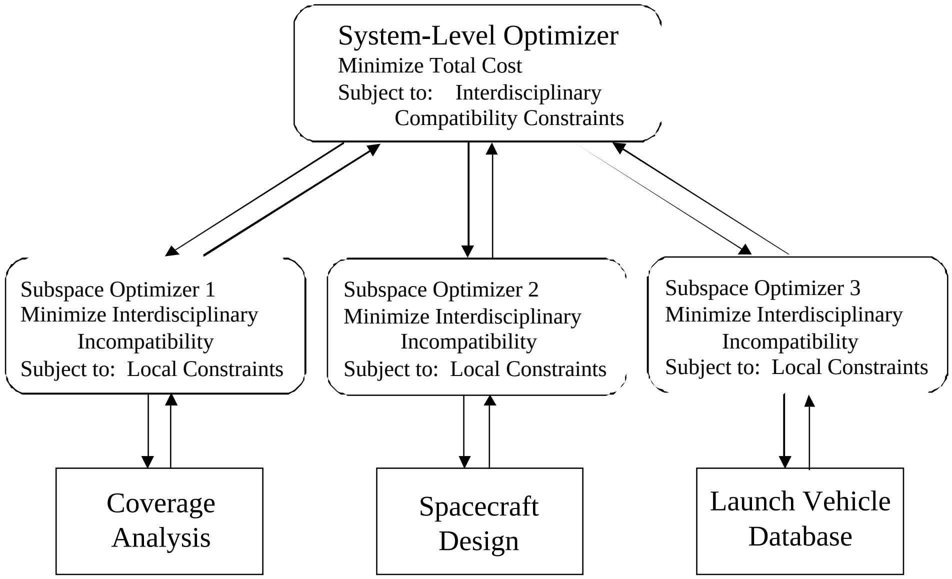

Figure 2. Collaborative Optimization implementation to the satellite constellation design and deployment.

CO is a design architecture that has been successfully applied to large-scale multidisciplinary optimization problems related to aircraft and space vehicle design. The constellation design problem shares similar characteristics as these previously established applications. That is, the design and deployment of a satellite constellation is multidisciplinary in nature with complex interactions among the individual disciplines. Furthermore, each of the three modules typically involves some type of optimization process. CO, unlike other approaches, allows these to be retained by employing an optimizer at the system-level to coordinate the overall process and circumvent the possibilities of having conflicting sub-level (disciplinary) objectives (Figure 2).

Preliminary results on the collaborative approach to design and deploy a space based infrared system placed at LEO are presented. Reformulation of each of the disciplinary problems is necessary and is described along with some discussions of implementation issues encountered. Certain simplification of the problem is allowed for this proof-of-concept stage.

2. Application Example

The U.S. Air Force Space Based Infrared Systems (SBIRS) is to provide global and theater missile warning, national and theater missile defense, technical intelligence and battle-space characterization [5]. The overall architecture consists of GEO and LEO satellite constellations, as well as sensors mounted on satellites placed in Highly Elliptical Orbits (HEO). The application selected for the proof-of-concept in this work is similar to the LEO components, also commonly referred to as SBIRS Low (Figure 3).

SBIRS Low complements the SBIRS High (SBIRS GEO and HEO satellites) in providing high-confidence missile launch identification. However, the major task of SBIRS Low is to acquire accurate post-boost state vector data of the target. This precision midcourse tracking capability is required for effective Ballistic Missile Defense (BMD). Off-line post-processing of infrared data obtained by the SBIRS Low system will also be used to enhance target detection and characterization.

The current SBIRS Low design plans for 20-30 satellites of approximately 700 kg each. Each platform carries a high resolution, wide field-of-view acquisition sensor, which operates in Short Wavelength Infrared (SWIR) band, and a slew-and-stare multi-spectral infrared tracking sensor. The acquisition sensor scans from horizon to horizon (plus a few degrees above the horizon) to search, detect and track missiles during their boost phase. It then hands over to the tracking sensor, which continues to monitor the missiles' trajectories through post-boost and reentry. The design life of the spacecraft is 10 years.

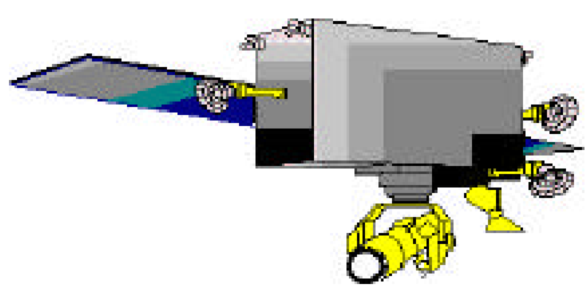

Figure 3. Current design of SBIRS Low.

For the proof-of-concept work presented here, the constellation configuration is fixed to use the Walker delta pattern of 4 planes consisting of 7 satellites per plane with relative phasing of 2. The payload design (two sets of sensors per spacecraft) is also fixed. With these discrete parameters kept constant during the optimization process, the problem is still sufficiently complex. The overall objective for each problem is to find the mission orbit (altitude and inclination) and the spacecraft design (sensor size and spacecraft mass) with the minimum cost through deployment. This consists of RDT&E (Research, Design, Testing and Development), production and launch costs. The computations for production cost assume a fairly conservative learning rate effect of 90%.

3. ANALYSIS TOOLS

Constellation Design

Written in Matlab, a coverage analysis tool was created to model and simulate the regional or point coverage provided by a single spacecraft or a constellation of satellites. This program solves and propagates the two-body equations of motion for each spacecraft, which is modeled as a point mass orbiting around a spherical Earth.

Third-body perturbations due to the Moon and the Sun are considered insignificant. The only force acting on the point masses is the Earth's gravity, which is assumed to be constant. The satellites experience no drag and solar radiation pressure is negligible.

The coverage computations are solely based on geometric constraints. There is no capability to set lighting (i.e. to simulate eclipses) or temporal (e.g. to simulate ground operation times) constraints. The satellites' sensors are assumed to have simple conic projections.

The input required for the simulation consists of the epoch information for all satellites. The sensors' half-field-of-view angles and the desired coverage region/points are the necessary inputs for the coverage computations. The program outputs figures of merit, such as percent coverage, maximum coverage gap, mean response time and shortest distance of approach between satellites.

Spacecraft Model

The equations used to size the payload and spacecraft bus are programmed into a MS Excel spreadsheet. Based on altitude, field of view and resolution and sensitivity requirements, the aperture diameter of the sensor is computed. The payload mass and power can then be determined by either scaling from an analogous existing system or by parametric estimating relationships where mass and power are functions of aperture diameter.

For this proof-of-concept work, the analogy approach is applied to size the acquisition sensor. Moderate Resolution Imaging Spectroradiometer (MODIS), a key instrument aboard *Terra* (a.k.a. *EOS AM-1*), is the analogous system used. It has 36 channels ranging in wavelength from 0.4 µm to 14.4 µm. A Pointing-Mirror assembly (PMA) allows for a ±55° scanning pattern across track (sensor nadir angle of 55°), providing approximately 2330 km swath. MODIS, with its 17.78 cm aperture diameter and ~38 cm effective focal length, has a mass of 250 kg and requires an orbital average power of 225 W to perform its mission.

The mass and power calculation for the tracking sensor utilizes the parametric approach. A Mass Estimating Relationship (MER) for optical sensor is borrowed from Lomheim, et al [9] and modified to account for the slew-and-stare configuration. The power calculation is based on first principles, realizing that the major component is the power required by the torque motor to point the sensor assembly during the mission.

For sizing the bus to support the payload, a MER was developed based on data gathered on several existing spacecraft (*EOS*, *SPOT*, *ERS*, etc.). This curve-fit equation relates spacecraft bus dry mass to total payload mass and power. The cost calculations are based on Cost Estimating Relationships (CER) published by Wong [10].

Launch Manifest

An electronic database was created as a MS Excel workbook containing information on several launch vehicles. The payload capability of each vehicle for various altitudes and inclination from different launch sites and for a variety of missions (e.g. circular LEO, sun-synchronous circular orbits, etc.) were obtained from their respective companies' publications (i.e., mission planner's guides). Publicly available cost data was compiled and averaged from various journals and reports [6], [7], [8]. The families of vehicles included are Atlas, Delta, LMLV, Pegasus, Proton and Taurus. For the computations presented here, it was assumed that up to eight of each type of vehicles are available to deploy the constellation within the given schedule.

4. DESIGN METHODOLOGY

Some reformulation of the problem is required in implementing the collaborative optimization architecture (Figure 4). Typically, the constellation design module uses the sensors' coverage capabilities to compute the most efficient constellation configurations and mission orbits that satisfy the overall coverage requirement [11]. In a collaborative environment, these interdisciplinary parameters (i.e., coupling parameters in Figure 1 such as altitude h, inclination θ, and sensor view angle η), become target variables, which in turn form the subspace objective function (J_1). Thus, the task of the subspace optimizer is transformed to minimizing the differences between the target values (h_0, θ_0, η_0) and the values of the local versions of the same variables (h_1, θ_1, η_1). Performance measures, such as maximum gap and mean response time, become local constraints. In this case, the requirement is a continuous one-fold global coverage. The minimum distance, which is useful to ensure no collision and little signal interference between satellites, is also a local constraint.

The built-in Matlab optimizer, which uses Sequential Quadratic Programming (SQP) method, failed to produce consistent solution for the constellation design sub-problem. This is due to the highly nonlinear nature of the problem. Grid search became the method of choice. However, since each coverage analysis is expensive in terms of Computer Processing Unit (CPU) times, a set of heuristics is also applied to help narrow down the design space and to guide to the solution.

The spacecraft model sizes the infrared payload, according to the given mission orbit and requirements. Ordinarily, the optimization here is to determine the combination of structure materials, attitude control approach, solar array types, etc., to support the payload at the lowest cost (or at the minimum spacecraft mass). With CO, these variables remain local. The module will be given targets for altitude (h_0), sensor nadir angle (θ_0), and spacecraft mass (M_0) and costs (RDT&E, R_0, and Theoretical First Unit (TFU), P_0). The local optimizer's task, then, is to meet these target values while satisfying local constraints, such as sensitivity and resolution requirements. Maximum *g*-loading, access duration to ground stations and eclipse duration found in Figure 1 are ignored for this simplified case.

The spacecraft design's optimization process employs a combination of Genetic Algorithm (GA [12]) and MS Excel's Solver. GA's advantage is in finding good solutions for a multi-modal problem, or one with intrinsic non-linearities and discontinuities. The Solver's main task is to find the local minimum given a good initial point found by GA.

The launch manifest problem is to find the strategy with the minimum total launch cost. Typically, given the constellation configuration, the orbit parameters and the spacecraft unit mass, this problem can be solved with Integer Programming (IP) techniques. In CO, the optimization criterion becomes a function of these coupling variables along with total launch cost (L) and the problem is no longer linear. Launch rates are the local variables, and launch vehicle availability and capabilities form the local constraints. The optimizer uses a two-level scheme, the first of which is a one-dimensional search technique to determine values for altitude (h_3), inclination (θ_3) and satellite mass (M_3). IP can then still be used to find the best launch strategy and to meet the constraints, for the given set of variable values.

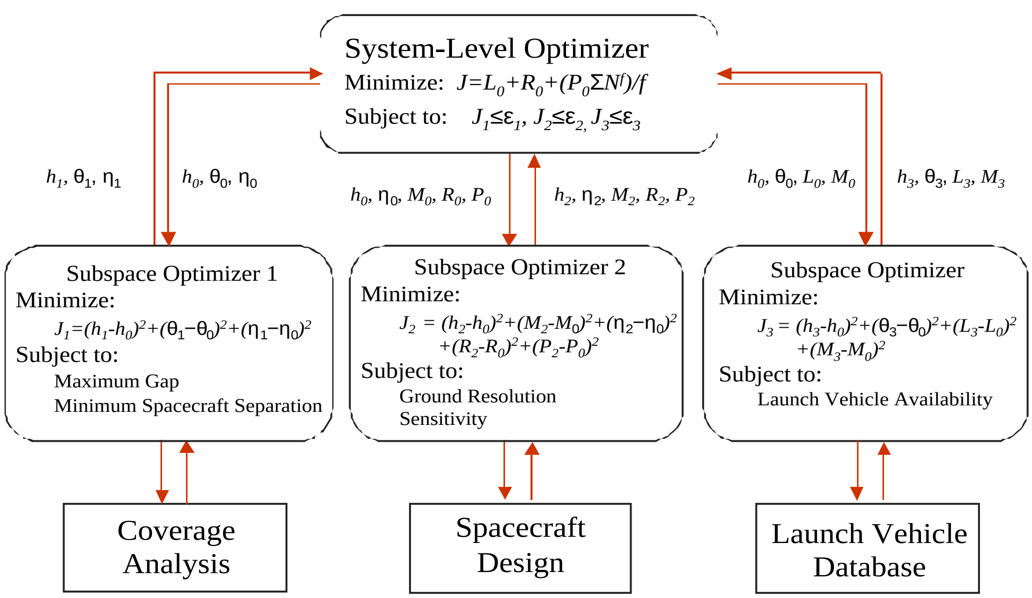

Figure 4. Details of the collaborative architecture.

At the system level, an optimizer coordinates the overall process by selecting target values for all the interdisciplinary variables. The allowable ranges for these variables (side constraints) are summarized in Table 1. The objective function is total cost to deployment and the constraints are that J_1, J_2, and J_3 are less than zero. Exterior penalty function was chosen to handle these compatibility constraints. This allows a constrained optimization problem to be treated as if unconstrained by penalizing the objective function for any violations. Powell's method with quadratic approximation was used to solve the transformed problem. This zero-order method requires no gradient information.

Table 1. Summary of the system-level variables and their range of values.

VARIABLES	Lower Bound	Upper Bound
Orbit Altitude, *h* (km)	1400	1700
Orbit Inclination, θ (deg)	50	90
Spacecraft Unit Mass, *M* (kg)	600	1000
Sensor Field of View, η (deg)	50	55
Total Launch Cost ($M)	100	500
Spacecraft RDT&E ($M)	200	500
Spacecraft TFU ($M)	2	200

5. Results

For the 28/4/2 constellation configuration, the result is summarized in Table 2. The converged design was obtained after 9 iterations at the system level. An iteration is considered complete when the convergence criteria for the unconstrained problem are met (i.e., when changes in the variables or objective function are less than the set tolerances).

Figure 5 through Figure 9 plot the progression of the system-level variables, depicting the negotiations between the disciplines. Altitude is the only variable involved in all three sub-problems. For this particular example, the launch manifest module matches the target altitude, inclination and mass given by the system optimizer well. The constellation design module tends to prefer higher altitude and drove the inclination and sensor view angles up to ensure global continuous coverage.

The cost targets stayed level until iteration 4, where the penalty function multiplier is finally large enough to allow the compatibility constraints greater influence on the direction of the optimization process. Until this point, the quadratic approximation tries to drive the cost variables R_0, L_0, and P_0 to negative values, activating the side constraints. The dip found in Figure 9 in iteration 5 is followed by a similar progression in the mass plot in Figure 8. This behavior is a result of production cost, the major component of cost through deployment as shown in Figure 10, still yielding greater effects on the overall objective function than the compatibility constraints.

In the converged design, the target total cost finally matched the actual cost computed by the individual analysis modules, as shown in Figure 9. A total of 4 Atlas IIIA launches (one for each plane) is required to deploy the constellation. Each one is capable of carrying 8 satellites (one on-orbit spare per plane).

Table 2. Summary of the result.

Number of Planes	4
Number of Satellites/Plane	7
Relative Phasing	2
Orbit Altitude, *h* (km)	1596
Orbit Inclination, θ (deg)	77.8
Spacecraft Unit Mass, *M* (kg)	743
Sensor Field of View, η (deg)	53.41
Total Launch Cost ($M)	360
Spacecraft RDT&E ($M)	392.6
Spacecraft TFU ($M)	151.4
Total Production Cost ($M)	3477.1
Average Unit Production Cost ($M)	108.7
Total Cost to Deployment ($M)	4229.8

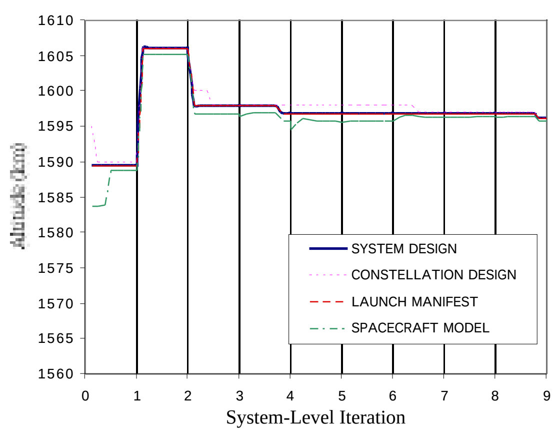

Figure 5. Progression of altitude within the collaborative architecture.

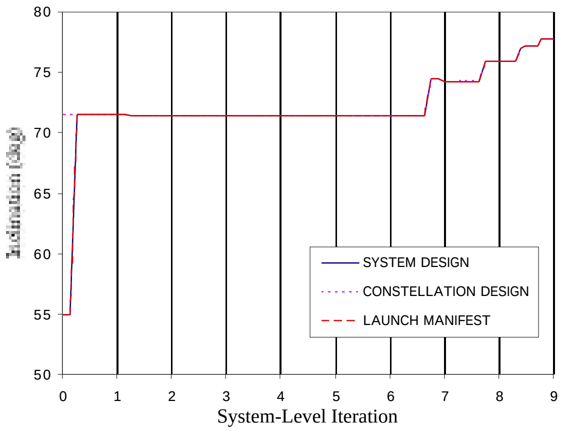

Figure 6. Progression of inclination angle within the collaborative architecture.

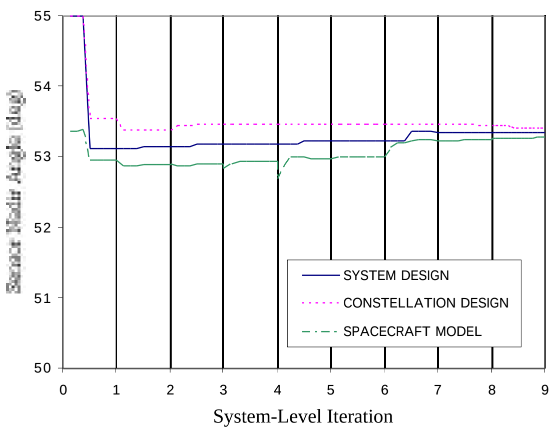

Figure 7. Progression of sensor nadir angle (swath width) within the collaborative architecture.

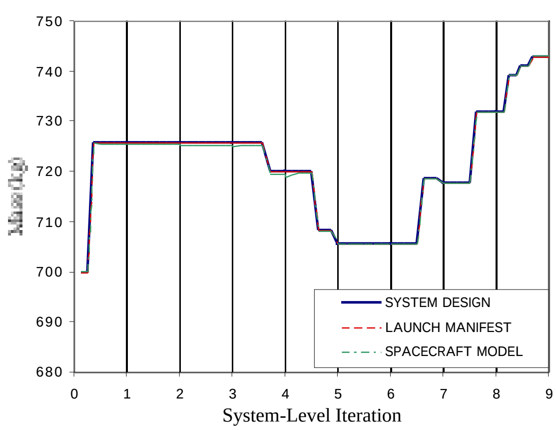

Figure 8. Progression of spacecraft unit mass within the collaborative architecture.

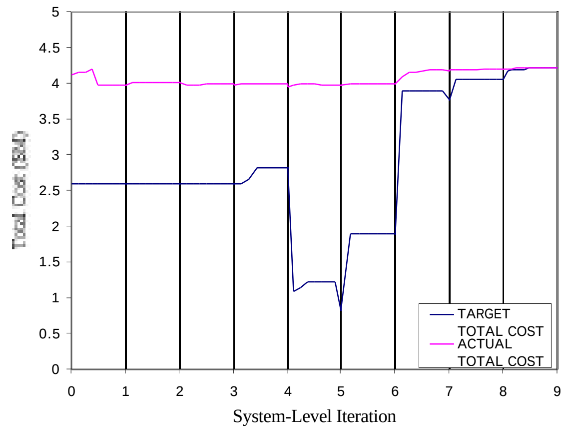

Figure 9. Progression of the total cost through deployment (system objective function) within the collaborative architecture.

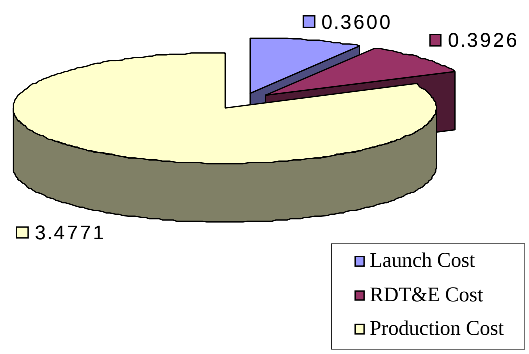

Figure 10. Total cost through deployment breakdown (in US$FY99B).

6. CONCLUSION

Summary

With some reformulation of the original problem, a converged space based infrared system design was achieved using the collaborative optimization architecture. A systematic multidiscipline multivariate optimization, useful for exploring the complex design space, was successfully implemented for the design and deployment problem of satellite constellation. This capability can potentially be very useful in this day when *more knowledge earlier* is emphasized.

The major implementation issues involve choosing appropriate optimization schemes for the sub-problems. The reformulation required by CO architecture causes the disciplinary problems to become nonlinear. Optimization of nonlinear problems is in general difficult.

Finally, the significance of this work is not placed on the numbers themselves. The converged design is based on mission requirements, ground rules and assumptions considered reasonable by the authors. Furthermore, it is limited by the level of fidelity of the individual analysis modules. Instead, the main contribution of the work presented here is that the implementation of collaborative architecture within the constellation design process is feasible.

Future Considerations

Thus far, only continuous variables are allowed at the system level, although discrete variables exist at the subspace level (e.g., launch vehicle rates). The simplified problem presented in this paper kept as constants the system-level discrete parameters, such as constellation configuration (number of planes and number of satellites per plane). To better represent the constellation design problem, however, these parameters must be allowed to vary. This, in turn, will better explore the design space and yield a more justifiable solution, at the cost of higher computational complexity.

Other improvements involve the individual analysis tools themselves. Higher fidelity spacecraft model can be achieved using a bottom-up approach, where mass and cost estimates are obtained by component break-down. More data for the launch vehicles database will also be beneficial.

For systems such as the SBIRS Low, ground station accesses are important considerations and will not only affect the spacecraft sizing, but may also influence the optimal mission orbits and constellation configuration. Eclipse duration, maximum *g*-loading and volume constraints are more factors whose effects are considered negligible in comparison to those of the variables used in this preliminary phase. Future works must include these in the design of the constellation system.

Finally, system analysts using this collaborative optimization methodology will benefit further if some automation is implemented. Programming the heuristics used in the sub-problems, rather than manually executing them, into their respective optimizers will greatly reduce design cycle times. Automation of the interfaces between the system and the sub-system optimizers can further improve the process and make the CO implementation more worthwhile to the designers.

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Her Doctorate research, under the direction of Dr. John Olds, is to apply MDO technique to the design and deployment of satellite constellations. Throughout her career at Georgia Tech, however, she has been involved in several different projects, both individual and team-oriented.

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Dr. Olds is a registered Professional Engineer in Georgia and a senior member of the American Institute of Aeronautics and Astronautics. He is also a member of the American Society of Engineering Educators and the Society of Automotive Engineers.