Multidiscipline Design as Applied to Space

Charles F. Lillie*, Michael J. Wehner and Tom Fitzgerald

TRW Space & Electronics Group
One Space Park, Redondo Beach, CA 90278

Abstract

The objective of this paper is to look at the spacecraft design process and see how that process balances desired spacecraft features within an imposed set of operational and cost constraints. The constraints often show up as competing multidiscipline interactions, which in their resolution lead to practical spacecraft designs. This paper gives examples of how the design process was implemented in a feasibility design study for NASA's proposed Next Generation Space Telescope (NGST), and describes how the project organization was used to effectively deal with multidiscipline design. Orbit selection, spacecraft propulsion, station keeping, and overall mechanical and thermal subsystem designs are discussed as examples of multidisciplinary design optimization. The final section is an across-the-board discussion of multidiscipline design optimization, what its benefits are, what are the negative points and what can be done to improve the process.

Introduction

This paper deals with work performed by the TRW-led study team under National Aeronautics and Space Administration Cooperative Agreement No. NCC5-137, awarded May 24, 1996 by the Goddard Space Flight Center, for research entitled: Next Generation Space Telescope Feasibility Assessments. The report was presented to the NGST Integration Team at GSFC on August 20, 1996.

The study was to involve industry, universities and/or non-profit organizations in the early planning for the NGST in a search for the best ideas for accomplishing the mission. The NGST project office felt that it would be necessary to go beyond simple parameter trades to non-linear thinking in order to break the current cost-aperture paradigm to achieve the $500M goal for NGST development, with a total life-cycle cost of $900M in 1996 dollars..

This paper describes some of the major features of our approach to developing the NGST spacecraft, launching it, and operating it for 10 years. The paper includes the mission requirements which we derived from the Dressler Committee’s “HST and Beyond” report, and examples of the trades and analyses which we performed to develop a mission concept and baseline configuration for the NGST, a development plan for enabling technologies, a cost estimate and a recommended management approach.

Figure 1 shows the organization of our study team and each team's responsibilities. Our organization paralleled that of the ongoing government study to facilitate the integration of our results with those from the other teams.

During the study the Integrated Product Teams (IPT's) responsible for the Optical Telescope Assembly and for the Science Module worked closely together to define an integrated payload

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*Senior Member AIAA

with an optimum partitioning of functions between the two assemblies. The Spacecraft systems team was responsible for the classical subsystems as well as thermal shields, vibration control, and the fine pointing system. The Operations team was responsible for the end-to-end data flow, including the ground system architecture and partitioning flight and ground system functions. The System Engineering team had responsibility for design integration as well as requirements definition, mission analysis, and interface definition

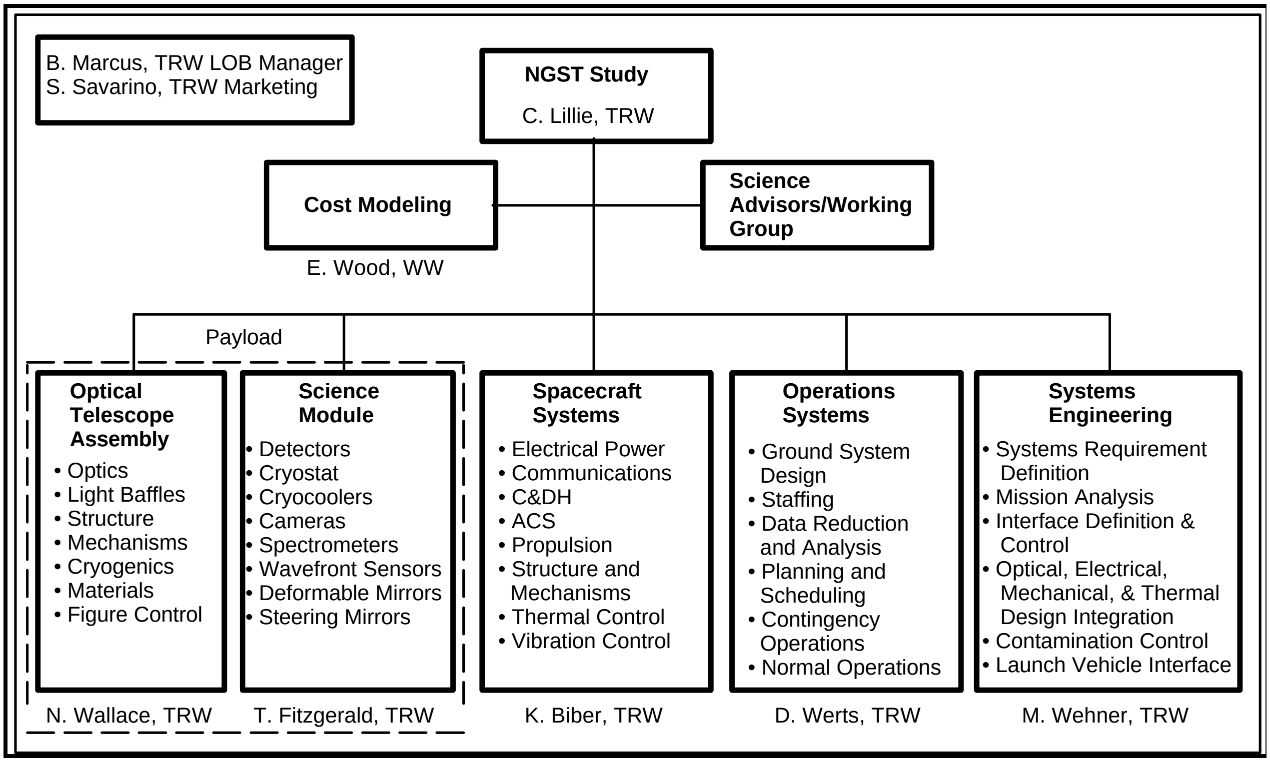


Figure 1. Study Organization

The responsibilities of the study team member organizations are summarized in Figure 2. TRW personnel led the NGST study IPTs and took the lead in the system engineering, design integration, science module, spacecraft bus and operations activities. HDOS led the optical design activities and supported the NGST study in requirements development, materials selection, performance modeling, active optical systems, and mirror assembly concepts. Swales personnel supported our NGST study in thermal design, contamination control, structure/ mechanism design, science module design, and operations.

Swales worked closely with personnel from Goddard Space Flight Center with experience in optics, structures, electromechanical devices, thermal control, cryogenics, contamination control, instrument design, and operations, who were members of our IPTs and supported our study activities. Additional support to the IPTs was provided by scientists and engineers from the Langley Research Center with expertise in analysis and control of flexible structures, active structures, active materials, isolation systems, and spacecraft analysis and modeling. The study was also supported by scientists from several universities who worked with their industrial counterparts to define the system requirements, develop conceptual designs for the instruments, assess system performance and review the outputs of our study.

Most interaction between team members was accomplished by weekly telecons and individual phone, fax, and e-mail communications. We also established an Internet homepage for the study team to facilitate the flow of information, including direct file transfers. This approach worked reasonably well, once the tools were in place and face-to-face introductions of team members had been accomplished.

	Team	TRW	HDOS	GSFC/SWALES	LaRC/Science Team
1	System Engineering	Lead, system requirements, trades, analysis, design integration	Requirements Development, optical performance modeling	Thermal design, Contamination control	Science team models system performance for typical targets
2	OTA, Including structures/ mechanisms	Lead, deployable structure, mechanisms	Optical design, material selection, modeling, assembly concepts	Support for structure and mechanisms design	LaRC supports active structures design, technology roadmap development
3	Science Module	Lead, system design, payload accommodation,	Wavefront sensor, fine guidance sensor, active optics	Instrument design	Science team supports instrument design
4	Spacecraft Bus	Lead, classical bus design, vibration control, fine pointing		Identify Enabling technologies, alternative designs, attitude control	LaRC supports vibration control, spacecraft analysis and modeling
5	Operations	Lead, ground system. design, mission operations. planning		Operations plans and scenarios, communications link trades and analyses	Science team supports mission scenario prep., MO&DA planning
6	Science/ MO&DA	Coordinate science advisor, working group activities		Science Support	Science team reviews study results,

Figure 2. Team Responsibility Matrix

Opening a website at TRW to external team members required the development of new network security procedures, which were successfully implemented midway through the study. Once established, this website was very useful for the disseminating data to the team and archiving the results of the study, as well as providing pointers to other relevant information on the Internet.

NGST System Design Process

We used our proven system engineering process on this project in defining mission requirements, deriving system requirements and developing system concepts. Several of these processes are iterative. The design features are balanced against the cost, risk and complexity of the concepts to produce a baseline concept. As the concept evolves the system requirements are finalized. The final product is a baseline NGST design and the associated technology development necessary to implement the design.

Design Reference Mission

We developed a Design Reference Mission based on the Dressler Report and our Science Team's expertise in astronomy. The NGST system was optimized to provide high quality information for investigating the early universe formation (using a large aperture and IR imaging). NGST would also continue the Hubble telescope role of determining the Hubble constant via Cepheid variables and other techniques. NGST would have very significant capabilities in ‘ordinary’ astronomy involving stellar evolution, galactic structure, planetary astronomy, etc.

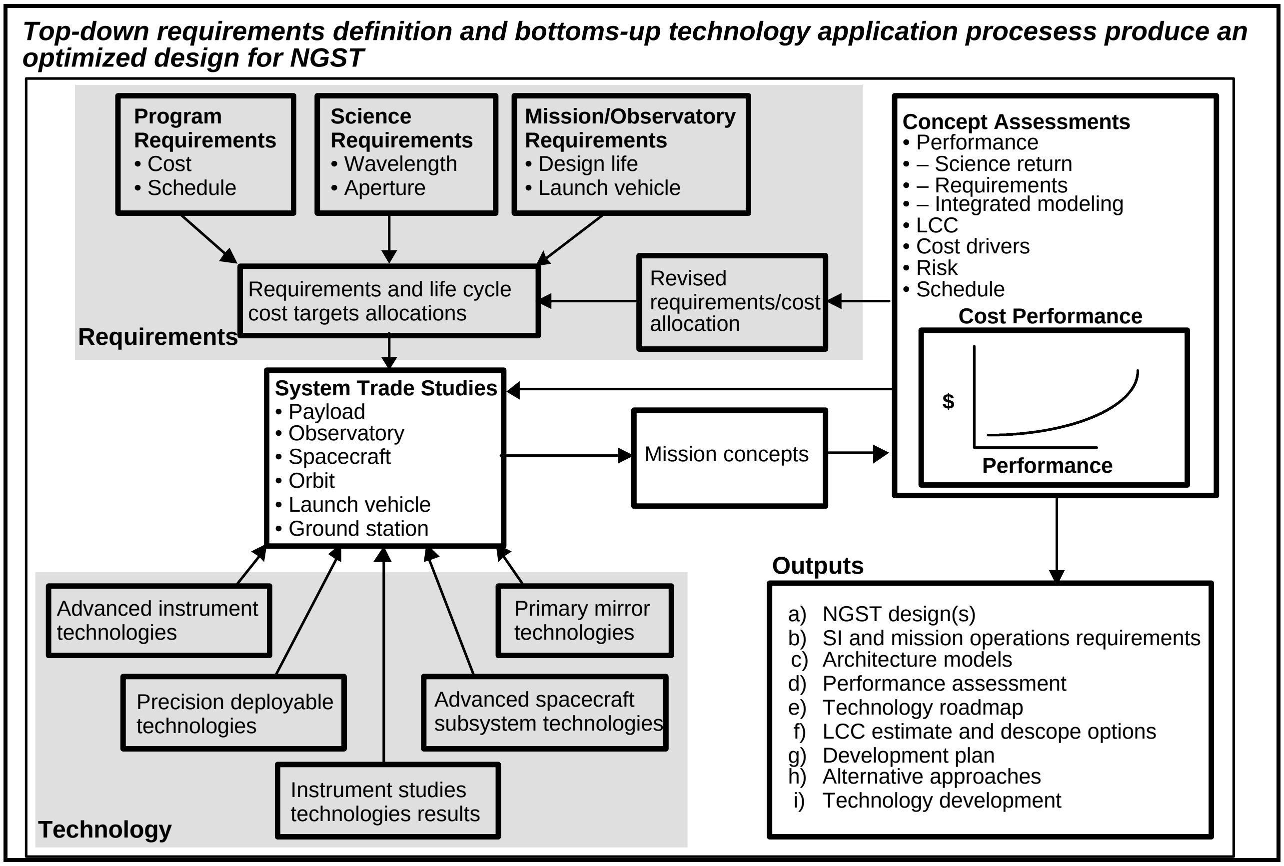


Figure 3. System Design Process

- Early Universe Investigation (Z ~4 to 10)
  - ~50% of NGST observing time
  - 100 to 200 survey fields at high galactic latitudes
  - Integration times ~10e3 to 10e5 sec
- Foreground Galaxies (Z ~0.5 to 3)
  - ~20% of NGST observing time
  - Observation of Cepheids, supernovae, etc. (Hubble Constant)
  - Integration times ~10e3 to 10e4 sec
- Local Galaxy (including Local Group)
  - ~10% of NGST observing time
  - Stellar evolution, brown dwarfs, etc.
  - Integration times ~10e3 to 10e4 sec
- Solar System Objects
  - ~10% of NGST observing time
  - Planets, comets, asteroids, Kuiper Belt objects
  - Integration times ~10 to 100 sec
- Targets of Opportunity
  - ~12 to 24 hour response time

Figure 4. DRM

Mission Requirements

Based on the DRM and our Science Teams guidance, we developed a set of Mission Requirements for NGST. These requirements are essentially concept independent, demanding only that NGST be a large aperture, imaging and spectroscopic IR optimized space telescope. Note that there are four graduations of importance in the requirements: 1) required, 2) highly desired, 3) desired and 4) goal. These are guidance to the concept designers as to the importance of these requirements. We placed some emphasis on targets of opportunity. Our design incorporates features dedicated to this. We believe that such flexibility is essential to provide data on comets and transient targets, such as supernovae.

The Dressler report and the DRM are directly responsible for the 'quality' requirement on this page. Early universe objects are highly red-shifted, which reduces the need for visible light observations. Therefore, we designed the NGST for diffraction limited performance at 1 µm. Note also the required bands correspond to the Dressler reports recommendations, but it was considered advantageous for NGST to exceed this band range if possible and cost effective.

Slit spectrometers are required. It was also desired that an imaging spectrometer be added, if feasible. We did not want NGST to be limited in stare time by design features. Therefore, a very long (~28 hours) requirement for stare time was included.

The agility requirement of 30° in 15 minutes is expected to not be stressing from a design viewpoint, and to provide a reasonably small loss in total observing time. Given that the majority of observations are long exposures (~2 hours based on the DRM), this implies that the telescope is repositioned ~10 times per day, resulting in down-time of 2.5 hours out of 24, which is roughly 10% down-time.

Field of view of the imager has been a parameter much discussed. Larger is of course better, but has significant cost implications in requiring large numbers of pixels and stresses the optics design. The value chosen is the same as the current Hubble Wide Field camera (if the square was filled).

Lifetime
- 10-year Mean Mission Duration (MMD) (required)
- 13-year design life (required)

Targets
- High redshift objects (required)
- Local area galaxies, clusters (required)
- Milky Way objects (required)
- Solar system objects
  - Planets (desired), outer solar system objects (highly desired)
  - Near-earth comets/asteroids (goal)
  - Targets of opportunity within Field of Regard

Observations
- Multi-color imaging (required
- Spectroscopy (required)
- Polarimetry (highly desired)
- High speed photometry (desired)
- Astrometry (desired)
- Response times
- Scheduled observations: 1 month (required
- Targets of opportunity: 24 hours (required; 12 hours (goal)

Aperture
- 6 m (required)
- 8 m (highly desired)

Quality
- Optics have diffraction limit (1/14 wave RMS) at 1µm (required)
- Nyquist sampled at lower end of each octave range except for bands < 1 µm

Imaging Spectral Bands
- 1 to 5 µm (required)
- 0.5 to 10 µm (highly desired)
- 0.5 to 20 µm (desired)
- 0.35 to 40 µm (goal)

2-D Spectrometer Bands (Slit Spectrometer)
- l/ l = 1000 selected imaging band (but no greater than 0.5 to 20 µm) (required)
- l / l = 10000 in 0.5 to 20 µm band (highly desired)

3-D Spectrometer Bands (Simultaneous 2-D Spatial Spectroscopy)
- l/ l = 50 in all bands (required)
- l / l = 1000 in 0.5 to 20 µm band (desired)

Stare Time
- No system limitations up to 1E5 sec (required)
- Sufficiently short such that bright targets not over exposed (required)

Agility
- Slew and settle a nominal distance (30°) within 900 sec (required)
- Sufficient to follow planets and outer solar system objects (required)
- Sufficient to follow fast moving comets (e.g., Comet Hyakutake) (highly desired)
  - 0.5 arcsec/sec (highly desired)
  - 2.0 arcsec/sec (goal)

Pointing Stability
- Total short-term jitter and long-team drift during exposure results in 20% larger diffraction image (note that is dependent of diffraction limit selected) (required)

Imaging Field of View
- 2.5 x 2.5 arc minutes (required)
- 4 x 4 arc minutes (highly desired)

Spectroscopic Field of View
- 2D slit 30 arcsec (required)
- 3D array covering 0.5 x 0.5 arc minutes (desired)

Coverage
- 4 steradian coverage of the celestial sphere (required)
- Coverage of any solar system object greater than 1.5 au from the sun, when projected onto the ecliptic plane (required)

Field of Regard
- 1 steradian (required)
- 2 steradian hemisphere centered 180° from the sun (hemisphere zenith pointing anti-sunward) (highly desired)

Figure 5. Mission Requirements

Coverage is defined as the region which can be viewed by NGST over an extended time (like one year). Field of Regard (FOR) is the region which can be viewed by NGST over a short time (like one day). Field of View is the region that can be viewed by NGST instantaneously. With the coverage requirement defined, NGST will be able to view all parts of the celestial sphere and the outer parts of the solar system. The highly desired FOR enables target of opportunity detection over half the celestial sphere at any one time. The required FOR corresponds to a 20° annulus perpendicular to the sun vector. This is commensurate with an NGST design without an elevation gimbal.

Baseline Concept

When stepping from the realm of mission requirements to system requirements, it is necessary to have a baseline system concept. This chart and the one following show the NGST baseline as of the conclusion of the three month study. In this paper we show the key trades and requirements flowdown which led to this baseline.

NGST is in a Lissajous orbit at the Lagrangian L2 point, placed there by an Atlas II AS (specified by the government) which follows an Earth-Moon flyby trajectory. Communication to earth is via X-Band.

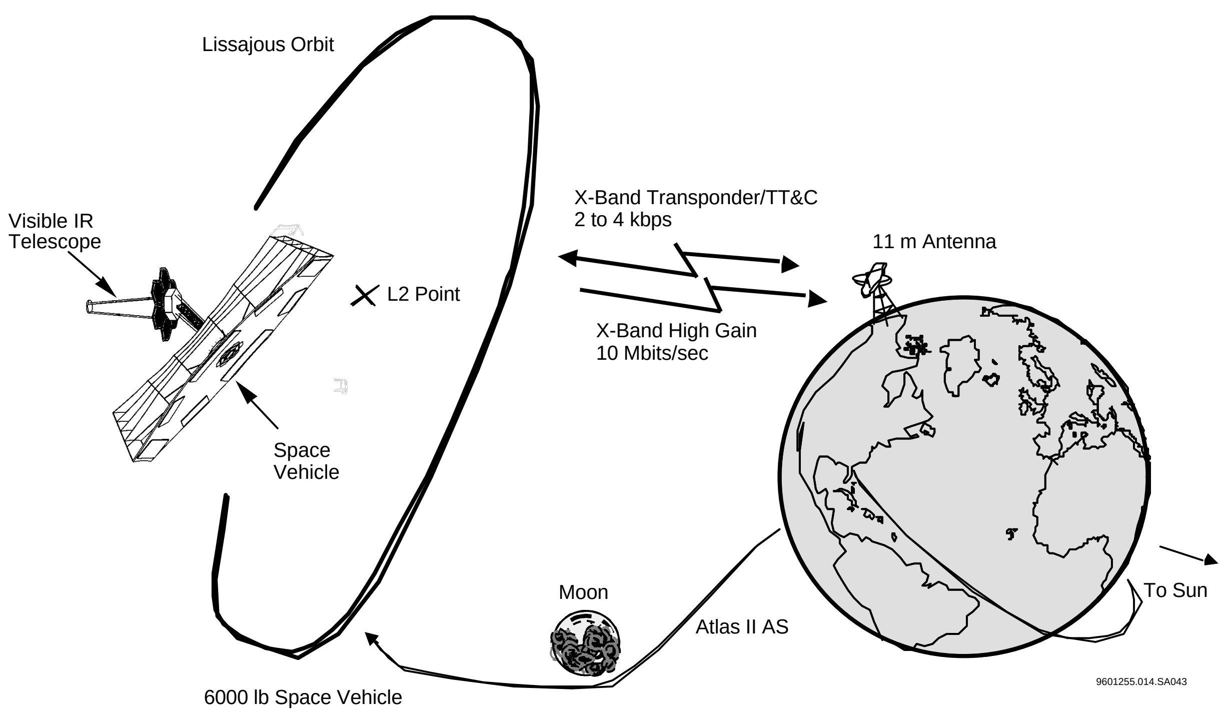


Figure 6. Mission Concept

A small (11 m) X-band antenna on the ground will provide low cost support to the NGST Space Vehicle (SV). A dedicated ground station would schedule and operate the SV.

Figure 7 presents the configuration which we developed for the NGST space vehicle. Note the sun and thermal shields are cut away for clarification. The spacecraft bus is located at the center of the shields, separated from the instrument module and telescope by a boom. The space vehicle is kept oriented such that the shields shade the telescope from the sun and earth. An optional shield sized to shade the telescope from the moon was considered, but rejected (shield size approximately doubled, from ~200 m^2 to ~400 m^2). The thermal load from the moon is negligible; the impact of sunlight reflected off the moon needs further investigation. The shields are supported by struts, attached to the spacecraft. Note the symmetry of the shield. This is to counteract solar pressure. Note also the placement of electrochromic patches, which are used as trim tabs to balance the pressure with the space vehicles center of gravity.

The telescope primary mirror is deployable using TRW's HARD (High Accuracy Reflector Development) technology. The telescope is coarsely pointed with an elevation gimbal. After thermally stabilizing, fine pointing is achieved by 'nodding' the space vehicle and rotating in azimuth about the sun line. A fine pointing mirror provides final pointing and tracking of the targets.

As an illustrative tool and as a guide to our trade space, we present the key trades we performed throughout this study. Note that some of the options are in italics and lined out. These are potential solutions that were rejected. The highlighted options have been baselined.

Key trades

Spectral Band Options: It was decided that a UV capability for NGST would be costly and not in keeping with the Dressler guidelines. Fabricating UV optics is expensive, and coupling that with deployable optics was considered too extreme. Similarly, to achieve

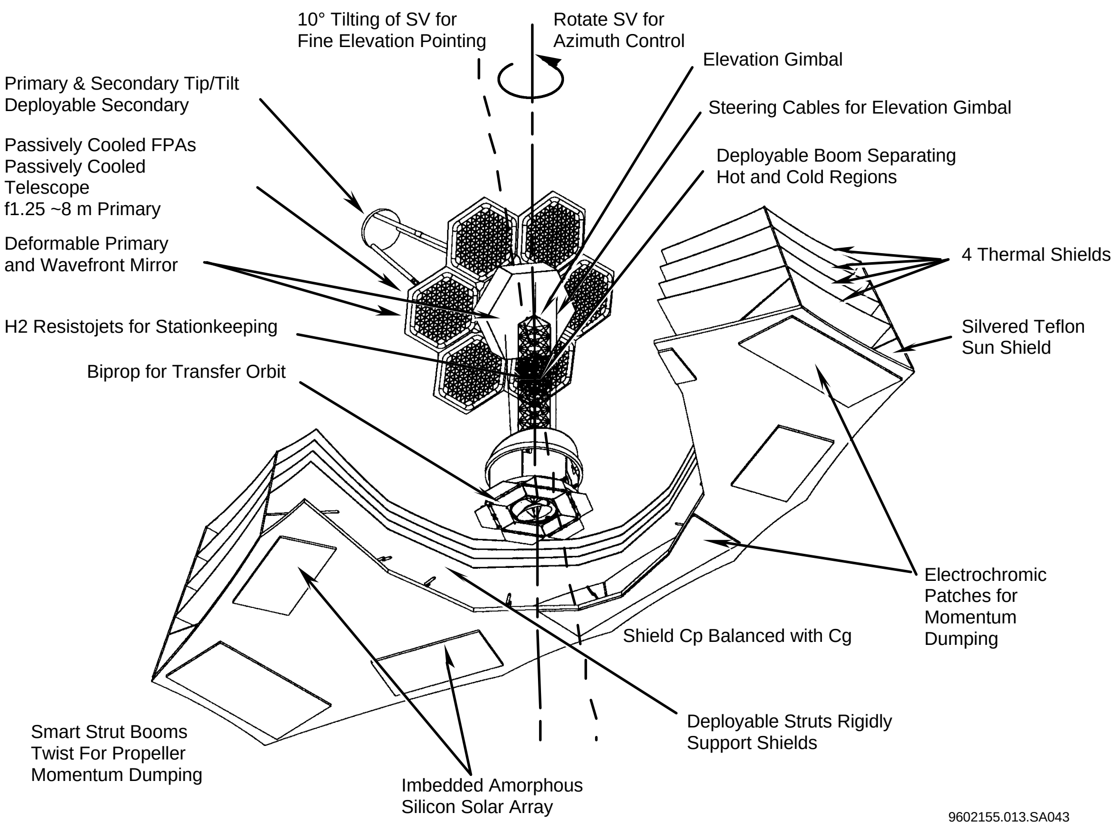


Figure 7. Key Design Features

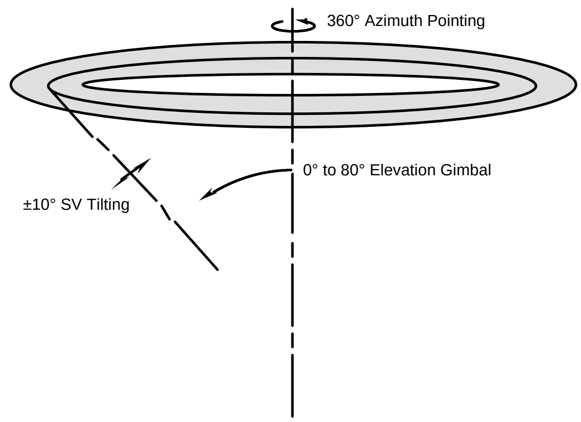


Figure 8. Payload Pointing

40 µm capability, we found that the optics would have to be cooled below reasonable levels (next figure). Later we will show that due to cost reasons, the 20 µm band was also rejected.

Transfer Orbit Options

A number of options are available to deliver the space vehicle to L2. The selected baseline, lunar flyby with phasing loops, offers a large launch window with good throw weight. Direct transfer is advantageous as it has a large launch window, but has the least throw weight of any of the options. Direct lunar flyby has the same throw weight as the selected option, but has a very short launch window. Integral propulsion is attractive as it

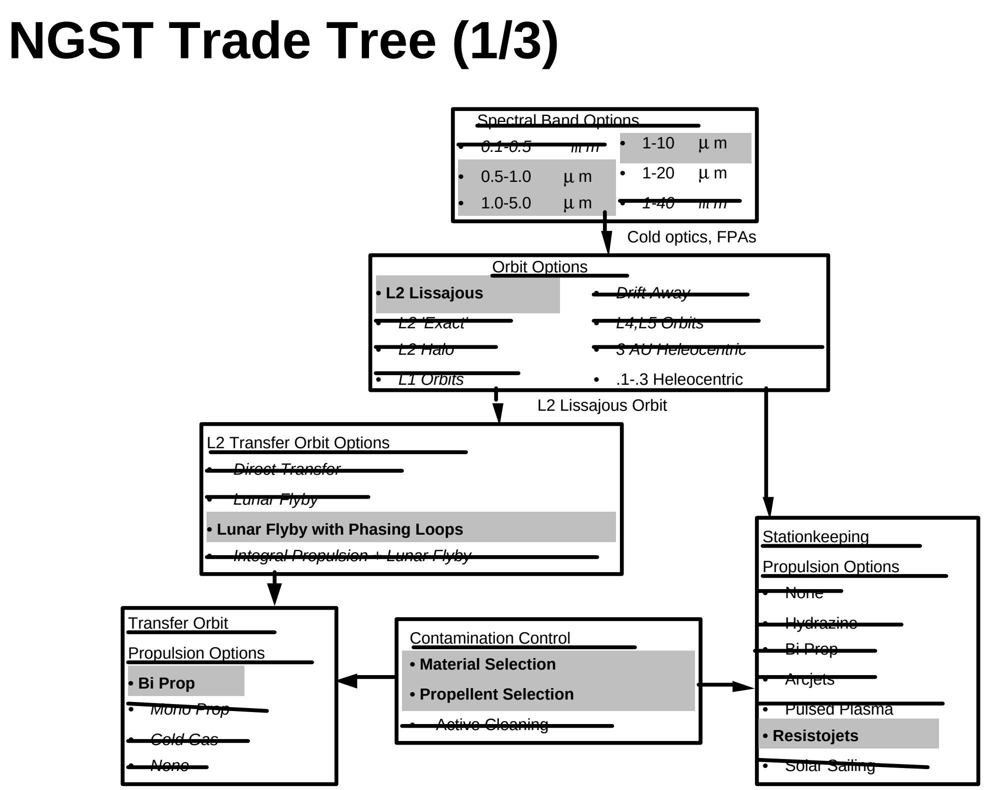


Figure 9. Key Trades

has the best throw weight of the options, but requires a large amount of propellant. Due to launch vehicle size constraints, we do not have the room to accommodate this additional propellant.

Figure 10 illustrates why NGST was not designed to operate at 40 µm. Operating at this point would require very cold mirror temperatures, which are beyond a reasonable design capability.
We allocated to thermal control the objective of passively cooling optics to ~30 K. This preserves the option of including a 20 µm band. The cost of achieving this temperature is very modest, only requiring the inclusion of an additional thermal shield layer. The requirement for cold optics drives the SV configuration.

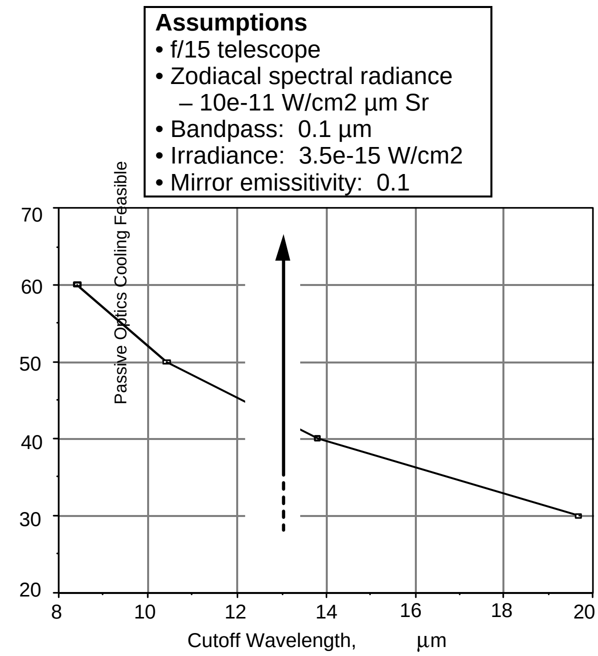


Figure 10. Temperature vs. IR Wavelength

<u>Orbit Options</u>

Orbit Selection Summary

- L2 Lissajous requires no insertion ²V, low stationkeeping ²V
  - Low meteoroid, solar flare flux
  - Negligible thermal from earth and sun
  - Good launch window, throw weight with “Lunar Assist + Phasing Loops” transfer orbit
- L2 halo requires insertion ²V
- L2 exact orbit requires high stationkeeping ²V
- L4/5 have very long communication ranges (1 AU)
- Drift-away orbit limits life; long communication range
- 1 AU heliocentric orbit needs further investigation
- 3 AU heliocentric orbit has lower throw weight; not needed for our bands

Near-Earth (and moon) orbits have a stressing thermal environment. Therefore, only orbits some distance from the earth were considered. The Lissajous L2 orbit was baselined. Attractive features of this orbit are: short range to the earth, low station keeping requirements, and no insertion DV to enter the orbit. L1 orbits have no advantages and the disadvantage of higher solar flux and having the earth shining in the telescopes field of regard. The drift away, L4/L5 and 3 AU heliocentric orbits are at long ranges from the earth and have minimal advantages in the primary band of interest (1-5 µm). Halo and L2 'exact' orbits have high station keeping and transfer DV requirements. The 1 AU Heliocentric orbit located 0.1-0.3 AU from the earth is still under investigation. This orbit may be able to be station kept at a reasonable earth distance.

One of the advantages of the selected Lissajous orbit is that no burns are required to enter, and it requires low DV to maintain. Also, this orbit is very large (300,000 km by 600,000 km axes), and only needs maintenance occasionally. The DV required to meet this the station keeping requirements is 2-4 m/sec/year, or 20-40 m/sec over the mission life.

Station keeping at L2

Considerations

- L2 is an unstable point, so station keeping is required
- Serious contamination concern due to cold optics temperatures

Station keeping Requirements

- Delta V: ~2 m/sec/year
- Station keeping maneuver timeline 3 months

Contamination Concerns

- Contamination is a major concern in cryogenic optical systems
- Acceptable contamination levels have not yet been determined for NGST
- Water, oxygen, argon, nitrogen, etc. can freeze out on cold surfaces
- Contamination control approaches
  - Select low outgassing materials for construction
  - Exercise contamination control pre-launch
  - Protect optical surfaces during launch and during early time on-orbit
  - Perform vacuum bakeout and use molecular absorbers to reduce outgassing rates
  - Minimize vapor and gas flux to cryogenic surfaces
  - Periodic heating of surfaces to remove contamination
- A major potential source of contamination is the propulsion systems
  - Prudent selection of the propulsion systems will reduce contamination issues
- An additional source of contamination is from launch vehicle fairing during ascent

Contamination Concerns

Contamination concerns have driven our selection of the propulsion systems for NGST. As detailed design progresses, contamination concerns will significantly affect material selection and will require designing in vent paths and baffles. Our ~30 K optics will be cold traps for volatile materials to condense on. Of particular concern are the effects from propulsion systems. Some propulsion systems are very dirty. Others are relatively clean, but produce by-products such as water which can condense onto the cold optics. On the following charts we present the propulsion trades and explain how contamination concerns were a driver.

Transfer Propulsion Trades

A number of options were considered as a transfer orbit propulsion system. Such a system, assuming a lunar flyby with phasing loops trajectory, requires ~ 100 m/sec DV (including margin). Multiple burns are required, extending over weeks after launch. Due to contamination concerns, we considered first using a cold gas with no contamination concern, such as hydrogen. However, we found that due to the low ISP and large DV required, it was not possible to package this system in the allowable volume. Electric propulsion was considered and rejected, mainly due to low thrust levels that were not compatible with the mission. Solids were rejected as too dirty and impractical due to restart requirements. Liquid propulsion was selected, specifically a dual mode system. Weight of the system is ~170 lb., using available thrusters. Contamination products are mostly water. This led us to delay deployment of the telescope and sun/thermal shields until after the transfer burns were completed. This would allow time for the propulsion system products to disappear.

Transfer Propulsion Trades

- Propulsion system requirements for lunar assist with phasing loops transfer orbit
  - ~100 m/sec total ΔV
  - Phasing maneuver ΔV at launch + days
  - Mid-course maneuver ΔV at lunar flyby + weeks
  - NO ΔV REQUIRED FOR L2 INSERTION
  - Low contamination system required

System	Advantages	Disadvantages
Cold Gas	• No contamination with right gas • Inexpensive	• Heavy, very large storage tanks needed • Low ISP, thrust
Solid	• Simple • High thrust	• Serious contamination potential • No restart; multiple engines required
Liquid	• High ISP • Restart capability • High thrust	• Contamination control must be considered
Electric Propulsion	• Very high ISP	• Very low thrust • High power requirements

Station keeping Propulsion Options

Contamination was the driving concern in selecting the station keeping propulsion system which led us to reject the liquid system used for transfer orbit. This is unfortunate, as only a few extra kilograms of fuel would suffice to provide station keeping over the mission life. The products (water, etc.) would likely be major contaminants on the cold mirror and other surfaces. Therefore, only non-contaminating fuels were considered further.

Cold gas systems are attractive due to their simplicity. However, the low ISP means that hundreds of kilograms of H2 would be needed over the mission life. There is not enough weight margin or volume to accommodate such a system.

Electric propulsion (resistojets, arcjets, Hall effect thrusters) is attractive, but often entails significant cost and requires high power. However, resistojets are a simple electrical system with great promise. This technology is flight proven, and TRW has past experience with these systems. Resistojets are very small (couple of inches long) and light weight (only ~10-20 kg of H2 needed). They use ~ 250 W of power each, and have a high ISP. As will be seen in the Space Support Module (spacecraft) discussion, resistojets make a lightweight attractive system. Operationally, due to low thrust, they would have to burn for hours. This would probably mean shutting down observations, but as burns are only needed every several months, this is not an issue.

Note the location of the resistojets on the concept description chart. Thrusters should operate though the Cg of the space vehicle. The resistojets are located on the boom at the approximate Cg of the system.
Our early baseline contained a cryostat (for instrument cooling) of solid Hydrogen. Interestingly, the amount of H2 needed in the cryostat for a ten year mission is about the same as needed for station keeping. We expended some effort to try to utilize the cryostat boiloff as fuel for the resistojets. Unfortunately, the H2 in the cryostat is at very low pressure (<<1 psi), and we could find no practical way to pressurize this gas to the 10s of psi required. Lack of synergy with the resistojets contributed to the demise of the cryostat.

Stationkeeping Propulsion Options

Options	Advantages	Disadvantages
Mono or Biprop	• Synergistic with transfer orbit propulsion system	• Serious contamination concerns
Cold Gas	• Very simple system • No contamination concerns - H^2 condenses at 5 K - He condenses at <<1 K - N^2 condenses at 30 K	• Requires 100s of kg of gas • Very large tankage required
Arcjet	• Can use H^2 or hydrazine • Very high ISP	• Complex system • Requires high power (>1 kW)
Hall Effect	• Can use H^2, N^2, Zenon (inert gasses) • Very high ISP	• Complex system • Requires high power (~1 kW)
Resistojet	• Can use H^2, N^2, Zenon • High ISP	• Requires moderate power (~500 W)
Solar Sailing	• Utilizes our sunshade • No contamination	• Tilting increases shade size • Very low thrust. Sufficient?

Launch Vehicle Capabilities

This list of current and anticipated expendable launch vehicles potentially suitable to the NGST mission indicates the relative performance parameters and fairing volume constraints. The foreign vehicles are listed for completeness and comparison, and could be of interest should the program become an international effort. The capabilities of future systems are listed with public performance specifications to protect competition sensitive contractor actual estimates. The trend of all planned future vehicles is increased performance at reduced costs. Fairing dimensions are inside payload usable volume. Approximate (~) performances are not based on specific mission estimates but are extrapolated from GTO capability. All estimates are for optimum inclination for each launch vehicle and launch site. The Atlas IIAR and Delta III vehicles currently under commercial development with contractor funds and are planned for first flight in 1998. Although details are still considered propriety, both contractors have plans to expand these vehicles into a family with increased capability. It is reasonable to expect the commercial market to stimulate substantial performance improvements in the medium and heavy class before NGST is ready for procurement.

Launch Vehicle Capabilities

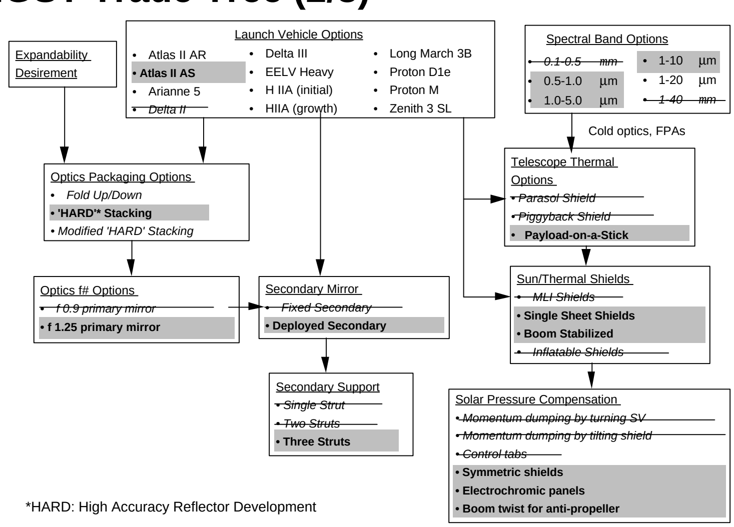
Launch Vehicle	GTO (kg)	C3=0(kg)	C3=-2.3(kg)	Dia (m)	Length(m)	Cylinder Length(m)
Atlas IIAS	3700	2710	2840	3.65	4.2	9.7
Atlas II AR	3900	~2850	~2990	3.65	5.1	10.6
Ariane 5	6800	~4980	~5220	4.6	9.2	15.2
Delta III	3810	2722	~2850	3.75	4.3	8.9
Delta II	1800	~1200	~1310	2.8	-	-
EELV Heavy	>12247	~8970	~9400	4.5	12.2	Unknown
H IIA (initial)	4700	~3450	~3620	5.1	4.9	10
H IIA (growth)	9900	~7250	~7600	Unknown	Unknown	Unknown
Long March 3B	4800	~3500	~3670	3.65	4.7	6.5
Proton D1e	6700	4100	4100	4.1	7.5	7.6
Proton M	7100	~5200	~5450	Unknown	Unknown	Unknown
Zenith 3 SL	5200	3400	3560	3.75	4.9	8.5

Notes:
1) Fairing dimensions are inside payload usable volume.
2) Approximate (~) performances are not established values but are estimated.
3) Performance is for optimum inclination for each launch vehicle site.
4) Atlas is currently doing design trades to develop a 5-meter diameter (outside) fairing for the Atlas IIAR series and the above. For special unique missions they could change the existing fairing ring and stringer design to get to a ~3.8-meter inside payload usable diameter.
5) Atlas IIAR series performance is assumimg a 3-foot stretch of the fairing as indicated.

NGST Trade Trees (2/3)

The following figure provides a roadmap through additional trades used to define our baseline. Here we concentrate on issues related to space vehicle design. Key to concept development is the realization that we have a very limited volume to package a very large structure. The launch vehicle constraints and the thermal considerations drove our configuration.

NGST Trade Tree (2/3)



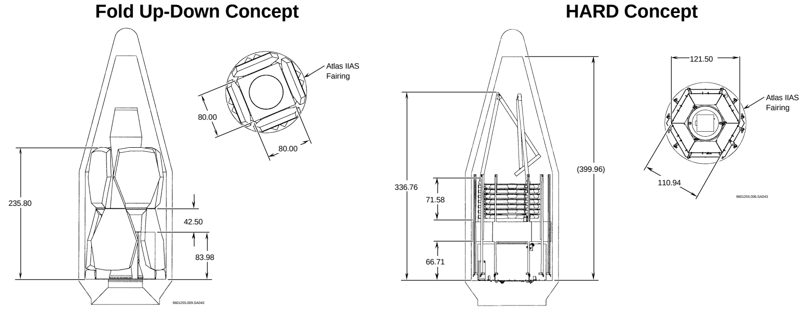
Deployable Mirror Concept

The small fairings available in the Atlas class drove our selection of the deployable mirror. Two general classes were considered, foldable and stackable. The fold up-down is attractive as it is simple. However, it wastes a great deal of fairing volume, limiting the room left for the spacecraft. Packaging studies indicated that we had insufficient volume left for the spacecraft and instruments. The alternate concept, based on the TRW developed HARD deployment

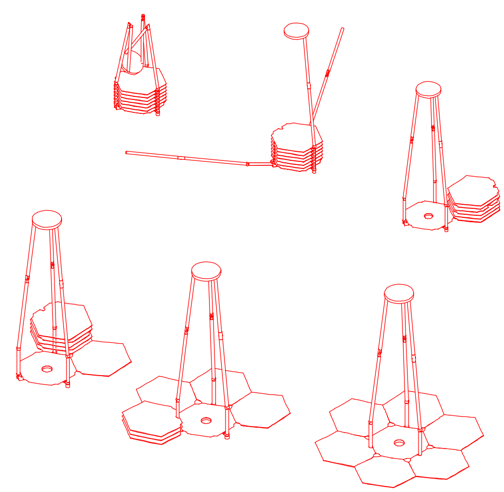
Stacked Versus Fold Up-Down Mirror Configuration Designs

- Space vehicle packaging trade hinges on the mirror configuration
- Fold Up-down is potential simpler mechanically, but has limited volume, also has smaller growth potential
- HARD concepts package more efficiently, increasing useable fairing volume
  - Has significant growth potential

Fold Up-Down Concept **HARD Concept**



Mirror Deployment



concept, is much more compact, and leaves the lower part of the fairing free for spacecraft and instrument packaging. TRW has demonstrated the HARD concept for large deployable RF antennas. Another very attractive feature of this concept is that it is expandable (see next chart). We have baselined the HARD concept.

Support of Secondary Mirror

The support structure for the mirror secondary has evolved significantly throughout our study. The first concepts had three fixed struts holding the secondary. Unfortunately, due to height limitations in the fairing, this required a very fast (f 0.9) primary mirror, which was considered very difficult to build and too sensitive to mechanical disturbances. Once the decision was made to have a slower mirror (f 1.25), we went to a single deployable boom holding the secondary. Analysis showed that the allowable deflections in the secondary location were very small (55 µm perpendicular to the optical axis, 300 µm in axis). Dynamically, when the telescope slewed, we were very concerned that vibration and hysteresis effects would exceed these values. While the secondary has five axis position control, it is desirable to not have to recollimate after every slew. Additionally, even with a very low CTE material, we found that temperature differences had to be kept at ~ 1°F both across the boom diameter and along the boom length. The temperature deltas along the length is considered challenging.

We considered supporting the secondary better by placing the boom in tension and adding guy wires. Deployment of the wires was difficult, and dynamically not much stability was added. Two struts were considered briefly. They were found to offer little additional stability.

Three struts is the current baseline. Two of the struts fold out of the way during mirror deployment and then fold back up to catch the secondary. This provides a rigid tripod structure. Thermal considerations remain, which is the primary reason for only moving the elevation gimbal periodically. At a constant gimbal angle, even with the SV tilting 10°, the thermal environment is stable.

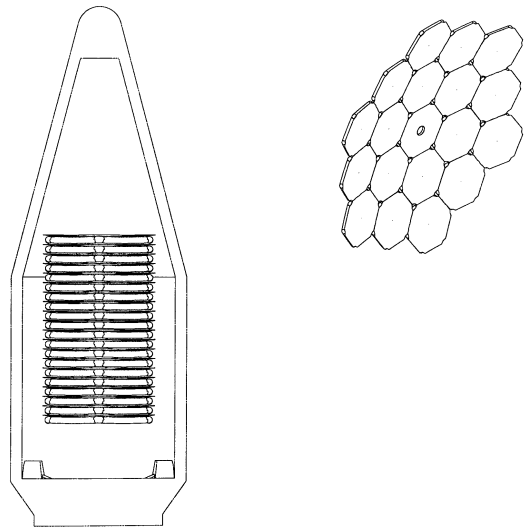
HARD Mirror Concept Expandability

The HARD technology allows easy expansion to much larger surfaces. With hexagonal petals, two rings of petals can be deployed. The entire stack of petals pivots about one corner of the last petal deployed and then drops into place. The remaining petals now pivot about the new petal, continuing the process.

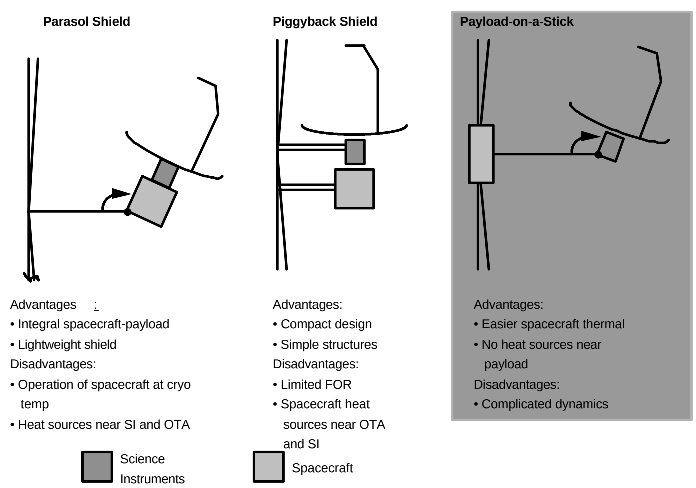
Generic Options for Space Vehicle Design

We examined three generic concepts for the Space Vehicle design. The first two, parasol and piggyback, have the spacecraft behind the sun/thermal shields. Operating a spacecraft in a ~30 K environment is beyond the state of the art. The payload-on-a-stick concept permits the spacecraft to stay warm while the instrument compartment and telescope are behind the shade, staying cold. We examined options for the boom separating the regions. Able has a FASTmast that looks acceptable. The mast is collapsible into a compact package one foot in height, and is stored in within a 47" canister in the spacecraft central cylinder. As it is deployed, the longerons, diagonals and battens snap into place. The boom can be constructed of low CTE material such as T300 graphite, resulting in only ~70 milliwatts conduction from the Spacecraft to the instrument module. Dynamically, the boom is rigid and stable. Even after a slew the boom returns to position very accurately - errors between the star trackers (located on the S/C) and the fine guidance sensors (located on the P/L) are ~arcseconds.

HARD Mirror Concept Expandability



Generic Options for Space Vehicle Design

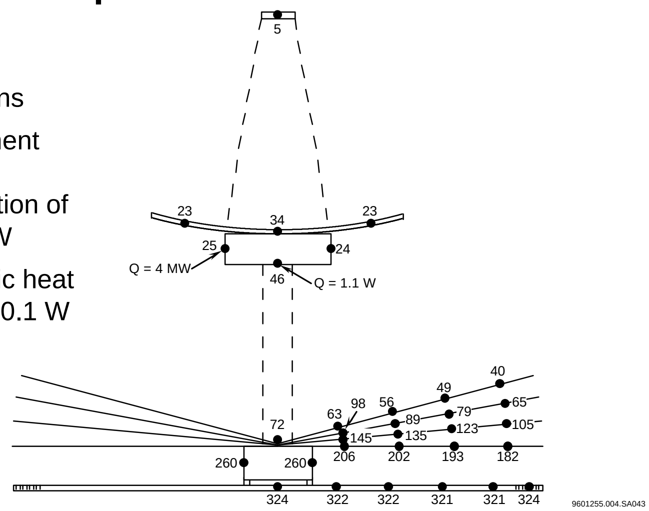


Predicted Temperature Distribution

The following chart presents the NGST temperatures with the telescope located at an elevation of 0°. Note that the mirror temperatures are <30 K, the desired value. The left side of the instrument compartment, where the IR instrument passive radiator is located, is at 25 K, adequate to cool the FPAs to ~ 30 K for near infra-red (NIR) imaging.

Predicted Temperature Distribution

- Assumptions
  - Instrument Module dissipation of 1.004 W
  - Parasitic heat load of 0.1 W



<u>Sun/Thermal Shield Design Options</u>

The baseline is a sun shield of two mils silvered Teflon, followed by four shields of 1 mil mylar with vacuum deposited aluminum on both sides, with an angle of 5° between the shields. This permits the cavity between the shields to radiate to deep space.

Deployment of the shields was a major issue. Early versions had inflatable shields. However, we had serious concerns over the additional weight of the bladders, gas for inflating, and how to rigidize the structure. Outgassing and deployment were other issues of concern. We baselined a strut deployment which would then pull out the sun and thermal shields.

The size of the shields is sufficient to prevent either the sun- or earth-shine from striking the telescope and to accommodate a 10° tilt in the entire SV for pointing.

Early versions of our shields had an asymmetric design (since the telescope gimbals in only one direction) to minimize shield size. Unfortunately, we found that the reaction wheels would saturate in ~11 hours due to unbalanced solar pressure. This led to the present symmetric shield. Eventually residual torque will spin up the wheels anyway, so methods of dumping the momentum were developed.

We considered trim tabs on the edge of the sun shields, but it is difficult to keep the telescope from seeing the hot tabs. An option that looks promising is to use panels covered with electrochromic materials that change reflectivity based on the voltage applied. This changes the resultant momentum by a factor of ~ two. Issues remain on material selection.

Another effect that must be compensated for is spin momentum buildup. Any mismatch in shield symmetry will cause it to act like a propeller. This could be stopped and the wheel momentum dumped by twisting the struts to change the pitch of the propeller.

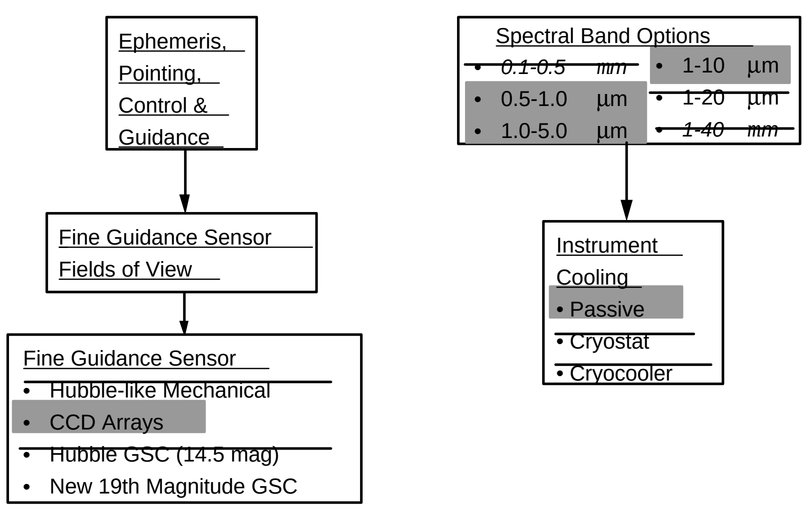
Sun/Thermal Shield Design Options

- Multi-layer insulation was first considered as shields
  - Weight of spacers between layers added considerably to mass
- Analysis showed that single sheets with an angular difference between them was as efficient and saved considerable weight
- Inflatable shields were considered and rejected
  - Added weight for the inflated portions (double thickness) and inflation gas
  - Concern on how to ridigidize the inflated spokes and rims
    - Concern that UV hardening required thermal shields that could withstand direct solar heating
    - Concern that hardening compounds might outgas
  - Concern on rigidity of structure after tilting or twisting
- TRW has demonstrated deployable booms/arrays
  - Booms can be easily applied to this task
  - Wire rigging can pull out the shields

Other Key Trades

We point the telescope coarsely by moving the elevation gimbal, and then by tilting the SV and rotating the entire SV about its axis. Reaction wheels will accomplish this. We examined the moments of inertia of the system and found that the required 30° slews can be accomplished in well under the 15 minutes required (30° Az slew in ~8 minutes, 10° El pitch in ~9 minutes). Additionally, we examined the vibration modes of the system and found that the lateral bending modes of the spacecraft/mast/payload are about 1 Hz. The sunshield modes will likely be lower

Other Key Trades (3/3)



in frequency. These are anticipated to damp out quickly, and any residual motion can be accommodated by fine pointing mirror in the optical train of the telescope. Reaction wheels are biased to spin at 10 Hz or higher.

Fine Guidance Sensor Options

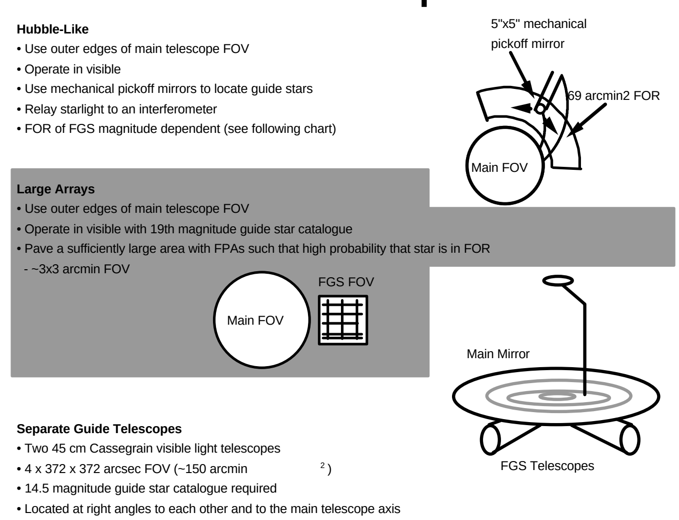
The mission requirements state that pointing must be stable enough to not increase the diffraction blur by <20%. At 1 µm , this corresponds to an AIRY disk diameter of 0.03 arcsec, and with 20% jitter, requires a pointing error of less than 6 milli-arcsec. Since a practical blur centroiding algorithm will provide location to ~1/5 of a pixel, this leads to a fine guidance sensor pixel size of 30 milli-arcsec. Given a FOV of 2x2 arcmin (see next chart), this results in an array size of 4000 x 4000 pixels, an easy value to achieve, with 18th magnitude, adequate signal-to-noise ratio exists to permit centroiding.

Three Fine Guidance Sensor options have been considered. One is like Hubble, which used a large field of regard field of regard but few pixels. Hubble used a mechanical arm to move a very small field of view within the. Given the limited Field of regard that we need at 18th magnitude, and given that we can readily buy enough pixels to cover this field of view, we rejected the Hubble concept. Separate guide

Fine Guidance System Sensor Requirements

- Mission Requirements
  - – Diffraction limited optics at 1 µm (1.2 l/d)
  - – Pointing stability 20% of diffraction blur
- Pointing System Requirement
  - – Diffraction blur: 1.2 x 1e-6/8 = 0.15 µrad = 0.03 arcsec
  - – Allowable jitter/drift: 6 milli-arcsec
- With adequate SNR, can use centroiding to locate a star
  - – ~1/4 to 1/10 of the FGS pixel size
  - – Assuming 1/5 => FGS pixel size is ~30 milli-arcsec
- Given FOV requirement of 2 x 2 arcmin (see previous chart): 4000 x 4000 pixels required
- Adequate SNR exists
  - – Flux from 18th magnitude star: 58,000 photons/sec (8 meter telescope)
  - – Image blurred to cover ~4 pixels: 14,500 photons/sec/pixel
  - – SNR (1 sec): ~sqrt (14,500) = ~120
  - – SNR (0.1 sec): ~sqrt (1450) = ~40
  - – (Note: Quantum efficiency of pixels assumed to be ~1)

Fine Guidance Sensor Options



telescopes were considered and sized. We rejected this concept based on limited volume in the fairing and the potential for misalignments between telescopes. Instead, the Large Array concept uses the existing main telescope field of view.

Multidiscipline Design Optimization.

Our paper describes the process used in the aerospace industry to develop design concepts for space science missions, using our Next Generation Space Telescope Feasibility Assessment Study [1] as an example.

The process begins with articulation of the need for a mission, a definition of its objectives and an estimate of the funding which is available. For NGST, the need and objectives were provided by the report of the "HST and Beyond" committee [2], while the funding level was determined by the savings which NASA could achieve by discontinuing HST maintenance activities after the 2003 servicing mission.

A mission concept and spacecraft design are then developed by a multi-disciplinary team organized by function or spacecraft element into Integrated Product Teams. These teams identify design options which meet the mission objectives, and select the most promising alternatives through a series of trades and analyses. Their selection criteria include system performance as well as cost and risk. If no design solution is found, the requirements are modified and new technologies [3] are introduced until an "optimum design" is achieved.

Design optimization is an iterative process, with more detailed designs and analyses generated during each iteration. For the NGST CAN study we used relatively simple thermal, dynamic and optical models to assess system performance and utilized existing spacecraft designs wherever possible. Much more detailed models and designs were generated during our current mission architecture study, including an integrated model to assess the end-to-end optical performance of our baseline design in the dynamic and thermal environment predicted for NGST.

The use of highly integrated system performance models for design optimization is a new trend in spacecraft design, made practical by recent advances in computer technology. Ideally, integrated models can be used to determine the sensitivity of our design to key parameters and find an optimum configuration. To date, however, high fidelity simulations are best obtained by linking existing stand-alone "industrial-strength" software tools with special purpose "translators". And building a detailed system performance model is a labor-intensive process which can only begin when a detailed design of the spacecraft is available.

Low fidelity integrated models using linked spreadsheets running on PC's are now being used by integrated design teams at many aerospace companies and government laboratories. It is important to have a well developed set of mission requirements and well defined mission concept before going to a design center; however; since a typical design effort a week of effort by 10-12 highly skilled engineers and scientists.

The use of multidicipline design teams is a powerful tool for exploring the design space to find a optimum solution, since experts in all of the relevant areas are readily available. They must be used judiciously, however; to control costs. Powerful analytic tools are available for design optimization, but more work needs to be done to link them together. Before a design centers or integrated models can be used effectively, much effort must be expended to refine the mission requirements and "zero-in" on a feasible mission concept and baseline design.

Summary

The objective of this paper was to look at the spacecraft design process and see how that process balances desired spacecraft features within an imposed set of operational and cost constraints. The constraints often show up as competing multidiscipline interactions, which in their resolution lead to practical spacecraft designs. This paper gives examples of how the design process was implemented in a feasibility design study for NASA's proposed Next Generation Space Telescope (NGST), and describes how the project organization was used to effectively deal with multidiscipline design. Orbit selection, spacecraft propulsion, station keeping, and overall mechanical and thermal subsystem designs were discussed as examples of multidisciplinary design optimization. The final section discusses multidiscipline design optimization, what its benefits are, what are the negative points and what can be done to improve the process.

Acknowledgments

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