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Structures and Materials Technology Needs for Communications and Remote Sensing Spacecraft

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List of Acronyms

AO	Atomic Oxygen
AMSC	American Mobile Satellite Corporation
BeAl	Beryllium Aluminum
BOL	Beginning of Life
C&DH	Command & Data Handling
CCU	Charge Control Unit
CDR	Critical Design Review
CME	Coefficient of Moisture Expansion
COSR	Cerium-Doped Optical Surface Reflector
CTE	Coefficient of Thermal Expansion
DOD	Depth of Discharge
E	Modulus of Elasticity
ELV	Expendable Launch Vehicle
EMI	Electromagnetic Interference
EMC	Electromagnetic Compatibility
EOL	End of Life
EOS	Earth Observing System
EPS	Electrical Power System
FCC	Federal Communications Commission
GaAs	Gallium Arsenide
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation, and Control
GPS	Global Positioning System
GSFC	Goddard Space Flight Center
GSO	Geostationary Orbit
HGA	High Gain Antenna
I&T	Integration & Test
K	Thermal Conductivity
Kx, Ky	In-Plane Thermal Conductivity
Kz	Out-of-Plane Thermal Conductivity
Kb	Bending Mode Gain Factor
LEO	Low Earth Orbit
LLV	Lockheed Launch Vehicle
LMSC	Lockheed Missiles & Space Company
LV	Launch Vehicle
MELV	Medium Expendable Launch Vehicle

List of Acronyms (Cont.)

MEO	Medium Earth Orbit
MIL-STD	Military Standard
MLI	Multi-Layer Insulation
MMD	Mean Mission Duration
MTPE	Mission to Planet Earth
NASA	National Aeronautics and Space Administration
NEA	Non-Explosive Actuator
NRE	Non-Recurring Engineering
OSC	Orbital Sciences Corporation
Pan	Panchromatic
PDR	Preliminary Design Review
PDU	Power Distribution Unit
PSTN	Public Standard Telephone Network
PWA	Printed Wiring Assembly
RAM	Random Access Memory
RE, REC	Recurring
SMEX	Small Explorer Program
S/C	Spacecraft
SGLS	Space-Ground Link System
SOA	State-of-the-Art
SOP	State-of-the-Practice
SSTI	Small Spacecraft Technology Initiative
STDN	Spaceflight Tracking and Data Network
TT&C	Telemetry, Tracking, & Command
UHM	Ultra-High Modulus
UV	Ultra-Violet

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Warren Hoskins, Structural Dynamics

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1.0 INTRODUCTION

Fueled by the expansion of geosynchronous-orbit commercial communications constellations into lower earth orbits, the accelerating growth of the spaceborne commercial remote sensing industry, and the current NASA trend toward smaller, high-performance small spacecraft, the small satellite industry is in the process of explosive growth. This growth presents new challenges and opportunities for the spacecraft industry, and to the research and technology development efforts which contribute to it.

The objective of this study is to conduct trade studies from the perspective of a small spacecraft designer/developer to determine and quantify the structures and structural materials technology development needs for future commercial and NASA small spacecraft. The study is focused on small satellite commercial communications and remote sensing missions to be launched in the period from 1999 to 2005.

This report describes the results of a brief, six man-month study performed by Lockheed Missiles & Space Company, Inc., (LMSC) for the NASA Langley Research Center under Task 30 of Contract NAS1-19241. The format includes charts which were presented at the Final Review for this task along with accompanying text.

1.1 BACKGROUND

During the planning effort for this study, two mission classes were selected from among six candidates as the focus missions for the study. The first, Low Earth Orbit (LEO) Commercial Communications, was selected because of the large constellations of small satellites being developed to serve this rapidly expanding $14 Billion market. Examples include the 66-satellite Iridium® system, the 48-satellite Globalstar system, and the planned 840-satellite Teledesic system. The second, LEO Remote Sensing, was selected because (1) it is a major national thrust as part of the Mission to Planet Earth, (2) there are a moderate number of NASA remote sensing small satellites planned with the potential for many more, and (3) there are at least four small satellite systems in development for commercial remote sensing ventures (e.g., Space Imaging™/℠, Eyeglass, EarthWatch). These two focus mission areas account for about 300 to 1,000 spacecraft to be launched during the next ten years.

Other candidate commercial mission classes that were considered are Medium Earth Orbit (MEO) and Geosynchronous Earth Orbit (GEO) Communications. These were not selected due to a lack of small spacecraft in these orbits and a sustained trend toward larger spacecraft that is forecasted to continue. Other candidate focus science mission areas that were considered but not selected are Space Science and Astrophysics, due to the specialized nature of the satellite designs and the relatively few missions planned, (e.g. the NASA/GSFC SMEX program).

A few key assumptions were made regarding the focus missions. Since this study addresses missions to be launched between 1999 and 2005, it is assumed that the subject technologies would need to be developed and space-qualified in the period from 1995 to 2001. This assumption provides sufficient time for the advanced technologies to be included in early system definition and design studies and traded in or out of the system design prior to the Preliminary Design Review (PDR). PDR is traditionally the latest milestone at which advanced State-of-the-Art (SOA) technologies can be cost-effectivley integrated into the satellite design. Based on this rationale, this study emphasizes technology development and near-term research for about the next five years.

Another assumption is that "small" satellites are defined as being less than 1,800 lb (818 kg) in weight, based on the results of the focus mission characterization study described in Section 2.1. The study results indicate that all four of the major LEO commercial remote sensing satellites and most of the "Big" and "Mega LEO" communications satellites weigh between 880 and 1,800 lb (400 to 818 kg). There are also some 5,000-lb, medium-class satellite systems (e.g. Odyssey, Inmarsat P) in MEO that are often included in the "Big LEO" communications category. These results are viewed as the product of extensive trades conducted by the spacecraft manufacturers on how to best meet their commercial market objectives, given the relative weight, volume, and cost of the current stable of launch vehicles.

While there are a number of lighter small satellite systems planned (known as "Little LEO's"), they have a much lower performance capability associated with the smaller "store and forward" communications market. The adjectives "Little", "Big" and "Mega" are a reference to the LEO constellation cost and capability, rather than the spacecraft size and weight[4]. As discussed in Section 2.1, Little LEO's are incapable of supporting continuous real-time world coverage for telephony (Big LEO's) or high-bandwidth multi-media communications (Mega LEO's).

Undoubtedly, the development of advanced small spacecraft technologies will have payoffs throughout the industry. Incorporation of these technologies will enable higher performance in large constellations of small LEO satellites, dramatically increased performance in small constellations of large MEO and GEO satellites, or both.

1.2 APPROACH

The approach to performing this task is focused on value-added from the perspective of LMSC as a designer and developer of small spacecraft. Though LMSC is currently involved in supporting several commercial communications and remote sensing missions (including Iridium®, Space Imaging™/℠, and to some extent Teledesic), there are many other small spacecraft missions being supported by the other corporations in the industry. To address this, information on the design and function of other small satellite systems was obtained from public sources (e.g., press releases, conference papers, and especially FCC license applications), providing a broader basis for the study of general technology needs and design trends.

The overall study effort was divided into two subtasks: (A) Small Satellite Systems Analysis and (B) Structures and Structural Materials Technology Development Trades. Section 2 describes the three main activities in Subtask A:

> *2.1 Focus Mission Characterization* Characterize the two focus mission areas. This includes the number of satellites, their orbits, weights, design life, performance, functions, design drivers, design trends, etc.
>
> *2.2 Technology Development Needs Survey* Conduct interviews with LMSC personnel to ascertain their structures and structural materials technology development needs. Consolidate the information obtained from the interviews, including a discussion of the small satellite design perspective and associated design integration issues.
>
> *2.3 System Analysis Approach for Technology Trades* Develop a systems analysis approach, including a small satellite State-of-the-Practice (SOP) technology baseline, for use in conducting the technology development trades and quantifying the system benefits of incorporating a subset of the technologies identified in item (2.2) above.

Space environmental materials technologies development needs, including environmental effects and life certification, are also addressed briefly in Section 2.2.3. Descriptions of performance requirements, structural and mechanical design details, and cost and weight breakdowns for a SOP small spacecraft bus designed by LMSC are provided in Section 2.3.1.

Section 3 describes the two main activities in Subtask B, Structures and Structural Materials Technology Development Trades :

> *3.1 Enhancing Technology Development Trades* Conduct abbreviated trade studies to determine the system benefits and incremental cost savings per satellite based on the substitution of State-of-the-Art (SOA) technologies into the design to meet *equivalent* system performance requirements. Rank the SOA technologies based on system benefits.
>
> *3.2 Other Technology Development Trades* Identify benefits of SOA technologies which enable dramatic *increases* in performance or system capability. Rank the SOA technologies based on system benefits.

The enhancing technology development trades are conducted using the technology development needs, systems analysis approaches, SOP performance models, and metrics resulting from Subtask A. For these trades, the reference SOP technology design approach meets the system performance requirements, which are held constant in the quantification of the payoffs. Conversely, the other technology trades represent special cases where the reference SOP requirements and performance models do not apply. These technology trades are conducted from the perspective of dramatically increasing performance or solving highly specialized, unique problems, such as disturbance isolation.

Within the resources available for the study, trades were conducted using a selected subset of 11 out of the 25 technology development needs identified in Section 2.2. Emphasis was placed on technology development needs for spacecraft buses, solar arrays, thermal control, and launch vehicle integration for both commercial communications and remote sensing missions. With regard to payloads, greater emphasis was placed on commercial communications antennas (e.g. large phased arrays) than for remote sensing payloads (e.g., optical benches, dimensional stability, contamination, and other environmental effects). During the conduct of the trades, the goal was to be as quantitative as practical in the discussion of the advanced technology benefits.

In conclusion, Section 4.0 summarizes the focus mission characterization, technology development needs, trades, technology rankings, and recommendations for further study.

2.0 SMALL SATELLITE SYSTEMS ANALYSIS

The three main Subtask A "Small Satellite Systems Analysis" activities performed during the first phase of this study are discussed in this section. Section 2.1 describes the LEO commercial communications and remote sensing focus missions under study. Section 2.2 summarizes the results of an LMSC-wide survey of small satellite structures and structural materials technology development needs for these missions. Finally, Section 2.3 describes the development of a systems analysis approach, including a small satellite State of the Practice (SOP) technology baseline and system metrics, for quantifying the system benefits of the technologies outlined in Section 2.2. This approach is subsequently applied in Subtask B (Section 3.0) to conduct technology trades and rank the State-of-the-Art (SOA) technologies based on system benefits.

2.1 FOCUS MISSION CHARACTERIZATION

Section 2.1.1 describes the market trends, planned satellite systems, orbit characteristics, and design drivers for the LEO commercial communications focus mission. Section 2.1.2 provides the same description for the LEO remote sensing focus mission.

2.1.1 Commercial Communications In the past, most space-based telephone and direct broadcast commercial communications services have been supplied by large satellite platforms in equatorial GEO orbits known as Geostationary orbits (GSO). Recently, the rapidly increasing global demand for telecommunications has spawned the planning and development of constellations of small LEO satellites for mobile telephony and high bandwidth (video) communications. After a discussion of the characteristics of the LEO commercial communications focus mission, a brief discussion of some of the trends in the GEO arena is provided.

The following charts describe the exploding LEO communications market. The systems serving the LEO communications satellite market form three groups, dubbed Little LEO's, Big LEO's, and Mega LEO's. The Little LEO's target a small, non-continuous global coverage, "store and forward" market for paging, messaging and data collection. This market will be served by very small satellites, such as OSC's Orbcomm system. The Big LEO's provide continuous worldwide coverage and need to capture only a small share of the cellular telephone market to be profitable. Mega LEO's provide high-bandwidth multimedia service for a telephone company-size market, and will compete with terrestrial fiber-optic networks, particularly in developing countries with rugged terrain.

The Big and Mega LEO constellations are composed of large numbers of high-performance small satellites that address the lion's share of the market and are the focus of this study.

LEO Communications
Market Trends

- 3 - 10 $Billion/Year Communications Market Projected by 2001
- Growth Rates Exceeding 40%/Year in Mobile Communications have Spawned Creation of New LEO Systems
- May Supplant Terrestrial Fiber Optic Communications in Foreign Countries
  - Half of the world's population lives more than 2 hours from a telephone
- Current Domestic Cellular Market Covers 90% of Population bus less than 50% of territory
  - In 1993, $1.4 Billion for Subscribers Travelling Outside of Service Area
- Planned Services
  - Telephone, Fax, Messaging, Data Collection, Positioning, Radio Broadcast
  - Video (Teledesic)
- Ten LEO License Requests Filed with FCC
  - First Round of Big LEO Licenses Granted in early 1995
  - Current Number of Applicants Expected to Exceed Spectrum Availability

Source: References 3,4,5,6

LEO Communications Services

- "Little LEO's"
  - Noncontinuous Worldwide Coverage
  - Paging, Messages, Alarms, Data Monitoring, E-Mail Text, Fax, Positioning
  - No Intersatellite Links "Store & Forward"
  - Domestic Delay Times up to 30 Minutes
- "Big LEO's"
  - Continuous Worldwide Coverage
  - Some Intersatellite Links, Gateways, PSTN Connections
  - Local Cellular Company Size
  - Long Distance Delay Times Similar to Terrestrial Services
- "Mega LEO's"
  - Continuous Worldwide Coverage (Terrestrial Dial-Tone Availability)
  - Intersatellite Links, Gateways, PSTN Connections
  - High Bandwidth Voice, Data, and Video; Bandwidth on Demand
  - Typical Phone Company Size and Features
  - Delay Times Less Than Fiber

Source: Reference 4

Planned Commercial Communications Systems

Proposed System	Service Date (yr)	System Cost ($B)	No. of S/C	Orbit (km)	Orbit Type	Weight Class	Channel Capacity (000's)	Data Rate Voice/Data (kbps)
BIG LEO								
Aries	1997+	0.3	46	1,020	Polar	Small	0.05	4.8/2.4
Iridium®	1998	3.3	66	780	Polar	Small	7	4.8/2.4
Globalstar	1998	2	48	1,414	Inclined	Small	5	4.8/2.4
Ellipso		1	16	500 x 7800	Elliptical	Small	0.6	4.8/2.4
Odyssey	1999	2	12	10,370	Inclined	Medium	3	4.8/1.2
Inmarsat P	2000	2.6	12	~10,000		Medium		Voice & Data
AMSC			12	~10,000		Medium		Voice & Data
MEGA LEO								
Teledesic	2001	9	840	700	Polar Sun Synch	Small	2,000	16/2000 (1.2 Gbps Video)
GEO								
MSAT	1995	0.5	2	35,786	GSO	Large		Voice & Data
Spaceway	2000	3.2	9	35,786	GSO	Large		Data & Video

Source: References 1,4,6

Data obtained from public sources about the planned "Big LEO" and "Mega LEO" commercial communications systems are shown, along with a few competitors in GEO. With the exception of the three ~5,500 lb, medium-class satellite systems in 10,000 km (MEO) orbits, all of the Big and Mega LEO systems are small satellites weighing less than 1,800 lb. Note the wide variety of system design solutions (e.g., orbits, inclinations, and constellation size) to meet similar service objectives for the Big LEO's.

The 840-satellite "Mega LEO" Teledesic system is noteworthy in terms of its size, high bandwidth capability, and expressed interested in advanced structures and materials technology. The competition for Teledesic in the next generation multimedia communications market is the Spaceway system in GEO, which is based more on present design technology.

While the advanced technologies addressed in this study may be available in time for incorporation in the Teledesic system, they are already too late to be used in the first generation of many of the Big LEO systems (e.g., Iridium® has already completed its Critical Design Review). However, advanced technologies could be incorporated in the second generation (Block 2) replacements for these systems, some of which are already in the study phase. These would be designed in the period 1995 to 2001 for launch in 2000 to 2005.

LEO Commercial Communications Satellite Design Trends

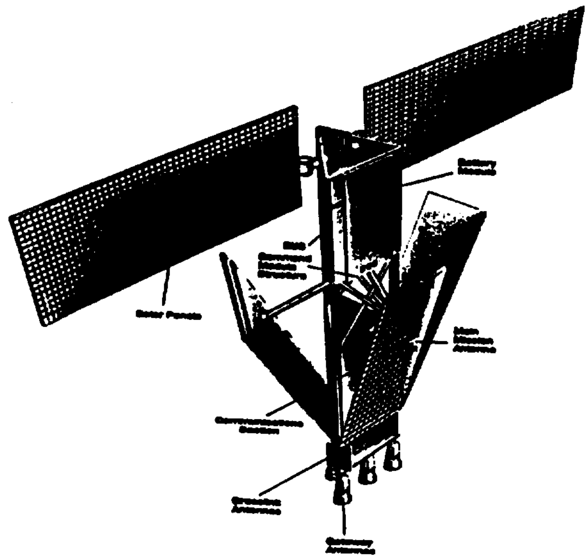
- Flat-Panel Phased Array Antennas
  - Highly-Focused, Electronically-Steered Beams
- Satellite Cross-Links
  - Meet Anticipated Future Demand for Increased Traffic
  - Increased Pointing Knowledge & Stability for Optical Links Needed
- Graphite/Epoxy Structures Becoming More Prevalent
  - Aluminum Still Used for Some Big LEO Systems
- Integrated Thermal Design
  - Minimize Cost and Complexity of Heat Pipes
- Combined Payload and Bus Communications and Processing
- Spacecraft Autonomy and Sparing
- Design for Low Cost, Mass Production and Process Control
- Spacecraft Stacking or Carousel for Multiple Launch on Multiple Launch Vehicles
  - Deployable Structures Technology for Arrays & Antennas
  - Design Must Envelope LV Requirements for Many LV's

Some of the current design trends for Big and Mega LEO commercial communications satellites are shown. The use of flat panel phased arrays with electronically steered beams for LEO missions is noteworthy. Parabolic reflector antennas are not used for LEO L-Band communications payloads because the satellites are moving relative to the user on the ground, and it is preferable to electronically steer the beam rather than gimbal and track a large antenna. Another key trend is the desire to incorporate optical crosslinks for higher bandwidth communications. The associated order of magnitude tighter pointing and stability requirements will require more expensive star-tracker based control systems and potentially damping and isolation technology. Finally, the launch of several satellites in groups on several different types of launch vehicles (including foreign launchers) is a key trend. All of the Big and Mega LEO satellites are launched in groups (arranged by orbit plane) on Medium ELV's or larger launch vehicles. Small ELV's may be considered for the replacement of on-orbit spares, but are not cost-effective for orbiting large constellations with multiple orbit planes.

Examples of typical small satellite designs which will form the global Big and Mega LEO networks follow.

Example Big LEO Commercial Communications Satellites

Iridium®



Source: References 1,6

Launch Vehicles
- Proton (7)
- Delta (5)
- Long March (2)

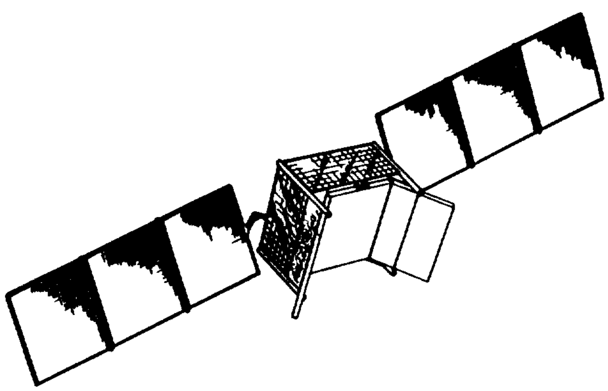
Weight
- 1,515 lbs (wet)

Power System
- 1200W Rigid GaAs Arrays
- 50 A-hr NiH_2 Battery

Attitude Control
- 3-Axis Stabilized
- Momentum Wheel
- 0.4 - 0.75° Pointing (3σ)

Other
- 5 - 8 Year Life
- On-Orbit Sparing
- Gr/Ep Structure
- $700M for 125 Buses

Globalstar



Source: References 1,6

Launch Vehicles
- Ariane (8)
- Delta (8)
- Proton (?)

Weight
- 880 lbs (wet)

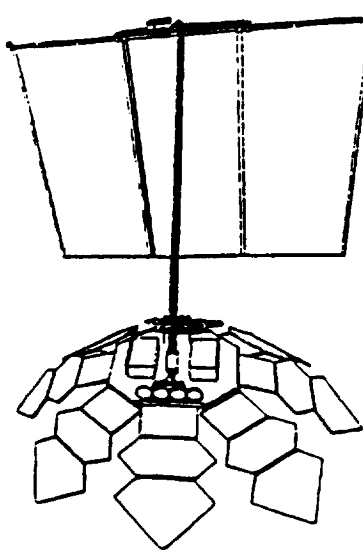
Power System
- 875W Peak
- 95W Nominal (Bus)
- 50W Nominal (P/L)

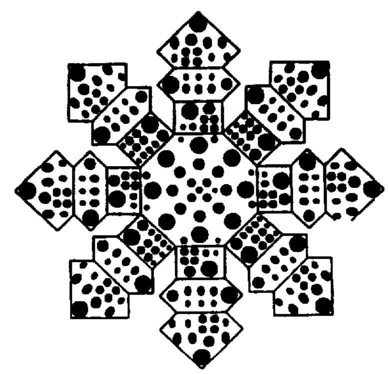
Attitude Control
- 3-Axis Stabilized
- Reaction Wheels
- 1.0° Pointing (3σ)

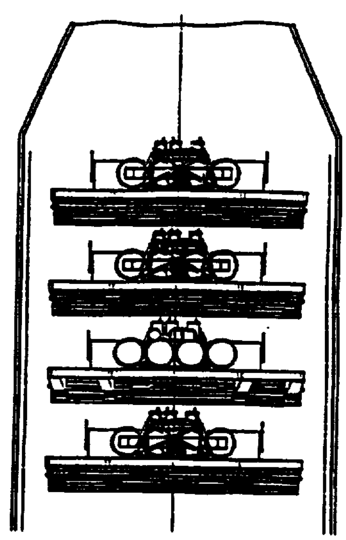
Other
- 7.5 Year Life
- On-Orbit Sparing (2,000 Km)
- Aluminum Structure

Example Mega LEO Commercial Communications Satellite

Teledesic







Source: References 1,6

Launch Vehicles
- Compatible with >20 Existing LV's

Weight
- 1,750 lbs (wet)
- 1643 lbs (dry)

Power System
- 6600W EOL Solar Arrays
- 3000W P/L
- 171W Bus

Attitude Control
- 3-Axis Stabilized
- Reaction Wheels
- 0.2° Pointing (3σ)

Dimensions
- 12m Diameter (Deployed)
- Solar Array 12m x 12m
- 4.2m Diameter (Stowed)
- 1.3m High (Stowed)

Other
- 10+ Year Life
- Gr/Ep Structure

Teledesic Subsystem Budgets

Satellite Subystem	Weight (lb, Dry)	Array Power (W, EOL)	Reliability (%, 10 Years)
Payload	317	3,000	80.049
EPS	526	2,288	98.949
Structure	191		99.999
Mechanisms	114		95.304
Thermal	81		99.999
Cabling	48		99.999
Propulsion	44		99.999
GN&C	26	51	96.700
C&DH	20		98.849
Reserve	275	1,068	-
System Total	*1,643*	*6,407*	*72.2*

Source: References 1,6

The Teledesic satellite conceptual design features a large deployable antenna and solar array attached to a flat octagonal bus structure. The nadir view of the antenna shows the structure composed of flat panel phased array antennas. The view of the satellites in the example launch vehicle shroud shows the concept for compact satellite stowage and stacking for launch.

The Teledesic system offers many structures and structural materials challenges to the designer. They include:

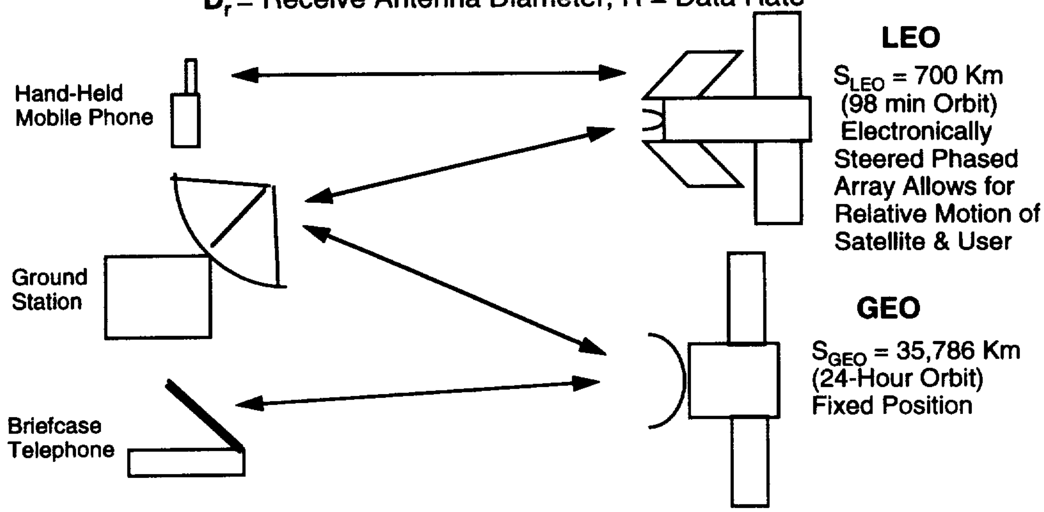
- lightweight, stiffness-critical structures design
- large deployable structures and mechanisms for solar arrays and flat-panel phased array antennas
- integrated thermal/structural design

A key challenge that the Teledesic system shares with the Big LEO systems is the rejection of the waste heat from the communications payload electronics, which can limit the performance and life of the satellite. This is reflected in the selection of a sun-synchronous polar orbit which allows the solar array to serve the additional function of a large sunshade in the design. Thermal issues are also reflected in the reliability numbers above, which are driven by the payload electronics, followed by the mechanisms, and GN&C hardware.

Communications Link Margins

Link Equation $E_b/N_o = f(P_t, D_t^2, S^{-2}, D_r^2, R^{-1})$

E_b/N_o = Ratio of Energy-per-Bit to Noise Density, P =Power
D_t = Transmit Antenna Diameter, S = Distance,
D_r = Receive Antenna Diameter, R = Data Rate



Source: References 4,6

LEO Advantages

- Link Margin and Data Quality (~50 x Closer)
- Time Delay on Par with Terrestrial Fiber Optics
- Smaller Ground Antenna and Power Reqmts for Mobile Phones
  - < 6" Diameter LEO vs ~26" Diameter for GEO (Spaceway)
  - 0.5 - 7 Watts Handset Power for LEO
- Smaller Spacecraft Antenna and Power Reqmts
- Higher Capacity at Lower Marginal Cost

GEO Advantages

- Proven Technology
- Less Costly to Launch and Operate as a System
- Same Satellite Always in View of User

The link equation relates the signal-to-noise quality of the digital signal to the power, antenna size, distance, and bandwidth characteristics of the system. Since noise quality is inversely proportional to the square of the distance between the transmitter and receiver, LEO orbits offer comparative advantages for mobile communications using hand-held telephones because of the lower power and antenna size required by the user on the ground.

LEO Commercial Communications System Design Drivers

- Cost
- Schedule
- Signal-to-Noise
  - Link Margin
  - Ground Terminal Size and Power (Especially Mobile Cellular Phones)
  - Satellite Antenna Size and Power
  - Number of Satellites (Cell Size and Slant View Angle)
- Time Delay
  - Orbit Altitude
- Capacity
  - Bandwidth, Data Rate, Frequency Re-Use, Number of Satellites
  - Computational Power (Switching)
  - Surge Capability (Power, Bandwidth, Processing)
- Reliability
  - Electronics Performance & Life
    - Thermal Design, Small SSPA's, Power Usage
  - On-Orbit Sparing
    - Parking Orbit Environment
    - On-Board Propulsion
- Mass Producibility
  - Quality: Process Understanding and Control
  - Design-In Test Capability
  - Use of Automation in Fabrication, Integration and Testing
- Multiple Launch on Multiple Launch Vehicles
  - Weight
  - Fairing Envelopes & Volumes
  - Envelope Shock and Vibration Environments
  - Foreign Launch Vehicles, Pre-Launch Processing, and Launch Sites
- Field of View
  - Antennas, Solar Arrays, and GN&C Sensors
- Spacecraft Autonomy
  - Management of Multi-Satellite Orbital Planes
  - System Fault Detection and Correction

In addition to cost and schedule, key performance measures which drive communications satellite system design are shown. Global coverage, link margin, time delay, power surge capability, mass producibility, multiple launch, and constellation autonomy are somewhat unique drivers for this focus mission.

For comparison, some of the GEO communications satellite market and design trends are shown. The source of this data is the 1994 edition of the Euroconsult "World Space Markets Survey - 10 Year Outlook"[3].

A key market trend is the shift from voice to video communications as a result of growing competition from terrestrial fiber optic communications systems. Another is the use of new frequency bands for increased performance and to reduce crowding. Finally, there are a few systems which compete with the Big LEO satellite constellations for future communications needs.

The GEO market is dominated by large satellites. Indeed, surveys indicate that the size, mass, power, cost, lifetime, and capability of GEO commercial communications satellites are steadily increasing with each successive design generation. In contrast to the LEO systems, the trend for GEO satellites is to use larger, lightweight, deployable, parabolic-reflector antennas. Because the current and projected trend is toward larger satellites, GEO communications satellites were not included in the focus missions for this study. Nonetheless, the equivalent relative payoffs of using selected advanced technologies to reduce weight and the associated launch cost savings to GEO were calculated.

GEO Communications Market Trends

- $16 Billion Satellite Market Potential 1994 - 2004
  - 178 Satellites in Third GEO Generation @ $90M each
  - 2/3 of Satellites Already Under Firm Contract
- Market Shift from Voice to Video
  - Estimate 88% of Transponders Currently Used for Video versus Only 45% in 1978
  - Voice Market Projection Flat to Continued Decline
  - Fiber Optic Systems Used for Point-to-Point Communications
- Gradual Shift from C- to Ku-Band
  - Most Current Transponders are C-Band
  - Ku-Band allows easier frequency coordination and smaller Earth stations
- Ka-Band Entrants in GEO
  - Ka-Band Rain Fade and Propagation Loss Previously Discouraged Use
  - AMSC/TMI MSat (1995)
    - Mobile Service to Augment Cellular Networks in North America
    - 2 MSat Satellites with Large, 20-Foot Diameter S/C Antennas
  - Hughes "SpaceWay" (2000)
    - 9 Satellites for Global Fixed Service, 26-inch Ground Antenna

Source: References 3,4,5,6

GEO Satellite Design Trends

- Technology Used to Reduce Satellite Quantity & Launch Demand
- Fewer Satellites Handle Same Amount of Traffic
- Bigger, More Sophisticated Multi-Frequency Systems
  - Hughes Communications, Inc. Invested $2 Billion in Galaxy System
- Size Increasing to Fill Increasing Fairing Volume
- Larger Deployable Antennas
  - 10 - 20 ft Diameter Lightweight Deployable Antennas
- Mass Steadily Increasing in the 90's
  - Average Launch Mass Increasing from 3,850 lb to 6,600 lb by 1996
- Power Increasing Despite Microelectronics Revolution
  - Digital Compression Technology Increasing Capacity/Transponder 5X
- Lifetime Increasing
  - 8 -> 11 -> 13 Years in 3rd Design Generation
- Cost Increasing Commensurate with Capability

Source: Reference 3

2.1.2 Remote Sensing Since the passage of the Land Remote Sensing Act of 1992, four LEO small satellite commercial remote sensing systems have been licensed by the Department of Commerce and are currently in development. Few details about these systems have been released publicly. The next few charts describe several planned NASA and Federal remote sensing satellites, along with typical characteristics and design drivers.

Remote sensing spacecraft are more likely to consist of a payload or set of payloads attached to a standard, multimission bus. A detailed description of an example state-of-the-practice multimission bus is provided in the next section.

Commercial Remote Sensing System	Launch Date (Yr)	No. of S/C	Orbit (km)	Weight (lb)	Launch Vehicle	Resolution	Estimated System Cost ($M)	DOC License Date
EarthWatch EarlyBird	1996	2	470	880 - 1100	START-1 LLV1	3m Pan 15m Multi	150 - 200	Jan-93
Space Imaging	1997	2			LLV2	1m Pan 4m Multi	500	Apr-94
Eyeglass	1997	1 - 2			Taurus	1m Pan Stereo	150 - 200	May-94
EarthWatch QuickBird	1997	2	470	880 - 1100	START-1 LLV1	1m Pan 4m Multi		Sep-94

Source: References 3, 6

- $3 Billion/Year Imagery Market Projected by 2000
  - Highly Fragmented
- Potential Military Market
  - Tactical Imaging Satellites
- Geographic Information Systems (GIS)
  - Merging of Computerized Information About Geographic Regions into Digital Maps
- Spectrum
  - Panchromatic & Multispectral
  - Hyperspectral
- Four Licenses Granted by Department of Commerce
- Commercial High Resolution Imaging Market Segment Served Exclusively by Small Satellites in LEO Sun-Synchronous Orbits

Source: Reference 6

Earth Observing System (EOS)

- Small Spacecraft Earth Probes
  - –Color (1998), Aerosols (2000), Altimetry (2002)
  - –New MTPE Earth Probes Launched Every 2 Years (FY96 Budget)

Small Spacecraft Technology Initiative (SSTI)

- Lewis (1996 LLV1 Launch)
  - –15m Multispectral, 3m Panchromatic Imaging System
- Clark (1996 LLV1 Launch)
  - –30m Multispectral, 5m Panchromatic Imaging System

Spacecraft	Launch Vehicle	Orbit (km)	Orbit Type	Payload Wt (lb)	Payload Pwr (W)	Pointing (deg)	Life (Yr)
Aerosols Probe	Pegasus	705	57°	88	15		3
Altimetry Probe	MELV	705	S.S.	660	290		5
Color Probe	Pegasus	705	S.S.	100	75		3
EOS	Delta II	705	S.S.	2,200	1,100	.03 - 0.2	5 - 7.5
Converged Weather	Delta II	~800	S.S.	~2,000	900	0.05	5 - 7

Source: References 2,6

The EOS Earth Probes and the NASA Small Spacecraft Technology Initiative (SSTI) satellites are examples of small NASA remote sensing satellites. Most remote sensing satellites are placed in a sun-synchronous LEO orbit, which provides a constant sun angle for imaging. The constant sun angle allows one to compare images of the same location taken on different passes. The EOS satellites and Earth Probes are placed in a 705 km orbit, while the advanced technology demonstrator SSTI satellites are in lower 500 - 600 Km orbits. Unlike any of the commercial communications or remote sensing spacecraft, two of the EOS Earth Probes are small enough to be launched on Pegasus rockets.

System design drivers for remote sensing satellites are shown in the next chart. Note that compared to current commercial communications satellites, there is an increased emphasis on high-performance attitude control, dimensional stability, and contamination.

Remote Sensing
System Design Drivers

- Cost
- Schedule
  - Commercial Time-to-Market
- Instrument/Optics Design
- Attitude Control and Pointing
  - Payload Agility
  - Accuracy
  - Knowledge
  - Stability, Jitter
  - Ground Truth
- Power
  - Surge Capability = Power Management
- Weight
- Data
  - Storage, Compression, and Downlink Rates
  - Minimize Antenna Size and Power
- Thermal
  - Small Spacecraft = High Power Density
- Alignment
  - Thermal Distortion
  - Shock, Vibration
- Field of View
  - Instrument, Instrument Calibration, and Thermal
  - Radiators, Solar Arrays, GN&C Sensors, HGA Antenna
- Contamination
  - Optics, Critical Surfaces

2.2 TECHNOLOGY DEVELOPMENT NEEDS SURVEY

Interviews were conducted to collect current information regarding structures and structural materials technology development needs. LMSC personnel from product centers, programs, and technology development centers were asked the same set of questions.

The results of these interviews, conducted early in the study, are incorporated throughout the report. Responses to mission-related questions are included in the focus mission discussion in the previous section. Insights into small satellite design and integration challenges are summarized in Section 2.2.1 Survey responses for structures and structural materials technology development needs are summarized in Section 2.2.2. Finally, survey responses for environmental materials technology development are summarized in Section 2.2.3.

- Interview various focus mission personnel
  - Structures Design Engineer
  - Stress Analyst
  - Dynamics Engineer
  - Materials & Processes Engineer
  - Manufacturing Engineer
  - Systems Engineer
  - Program Management

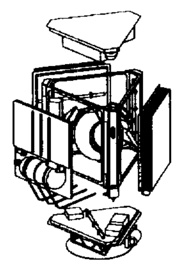
- Example questions
  - What are the key design drivers for your mission?
  - How are small satellites different than large ones for your mission?
  - What are your structures and materials technology trades?
  - If you had additional funds, how would you invest them in structures & materials technology development?
  - When is the technology need date?
  - Do you have life-certification issues?
  - Do you have material characterization issues?
  - Do you have process issues?

2.2.1 Small Satellite Design and Integration

System Requirements Flowdown

- Launch Environment
- Orbit Environment
- Payload Support
- Reliability
- Life
- Pointing
- Power

Structural Design & Integration Solves System Problems



- Environments
- Envelope
- Fields of View
- Alignment
- Stability
- Stiffness
- Strength
- Deployment
- Mechanisms

Design-to-Cost Capabilities Flowup

- Qualified Designs
- Thermal
  - Conduction
  - Radiation
- Radiation
- Random Vibration
- Shock
- EMI/EMC
- Size
- Weight

2.2.1 Small Satellite Design and Integration A central theme which emerged from the survey was the important role that the structure plays in integrating the satellite design. During the design and development process, system requirements are flowed down from the mission objectives through the system specification, while the capabilities of individual, qualified, off-the-shelf components are "flowed up" from manufacturers and vendors. Designing the structure to accommodate the performance ranges for existing components limits the amount of added qualification testing required to meet out-of-spec environments (e.g., thermal, shock, random vibration, stability), thereby reducing both cost and risk. Thus, the structure plays a key integration role in accommodating the various capabilities and restrictions at the component (box) level in a manner which meets the system performance requirements at lowest cost.

For the above reasons, the structural design is often a highly unique system solution. Many survey respondents were careful to point out that their structures and structural materials technology development needs can vary widely from mission to mission.

Small Satellite Design and Integration Challenges

- Thermal
  - Thermal Density Increased
  - Thermal Capacitance Decreased
- Shock
  - Equipment Closer to Sources
- Pointing
  - Small Bus Inertia Relative to Solar Array Size
- Radiation
  - Less Shielding
- Contamination
  - Equipment Closer to Sources
- Deployable Structures and Appendages
  - Confined Payload Envelope
  - More Restricted Field of View

Another interesting result of the technology needs survey was a list of some of the increased design integration challenges for small satellites relative to larger ones. Some of these challenges, i.e. thermal density, thermal capacitance, and radiation, make it more difficult to choose composite materials over aluminum for the bus structure. While the most difficult design challenges for specific small satellite missions may differ, this list suggests potential areas where advanced technology could be used to solve small satellite design and integration problems. The advanced technologies could be inserted directly, or added to a "toolkit" which offers the designer a wider range of approaches and solutions to small satellite design challenges.

2.2.2 Structures and Materials Needs Although they are in many cases interrelated, the results from this part of the survey were divided into four technology development needs categories. They are (1) integrated structural design, (2) deployable structures, mechanisms, and launch vehicle integration; (3) precision pointing and jitter stability, and (4) dimensionally stable structures. While the obvious technology needs are related to reducing cost and weight, particularly for the commercial missions, there are also some technology development needs for increased performance as well as design flexibility and standardization.

Integrated Structural Design The principal mission needs in the integrated structural design category are technologies for integrating improved thermal management and heat dissipation, EMI/EMC shielding, isotropic fittings and brackets, and electronics boxes into a cohesive structural unit. Several survey respondents pointed out that using technology to address the integration issues associated with composite structures would make it easier to reap the weight savings over aluminum structures, particularly for LEO missions. Thermal management and CTE matching were singled out as the most pressing technology needs.

Related technology needs include materials characterization and standardized material property databases, as well as analytical tool development. For the commercial communications missions in particular, low-cost, high-volume, production methods are a critical enabling technology. Development of these technologies could provide the designer with greater flexibility in optimizing the structure to meet small satellite design challenges and dramatically improve performance.

Integrated Structures Technology Development Needs

MISSION NEED	SOA TECHNOLOGY	RECOMMENDED DEVELOPMENT
Improved Heat Dissipation	Integrated High Thermal Conductivity Structural Materials	• Thermally conductive fibers (K1100, P120, etc.) • Diamond-coated fibers • Advanced layup methods, Kz enhancement • Low cost carbon-carbon materials processes
Improved Heat Dissipation	Integrated Heat Pipes	• Increased conductivity and reduced CTE mismatch • All-composite heat pipes • Composite radiators with heat pipes embedded in composite laminates or sandwich panels
Improved Heat Transfer Across Interfaces	Advanced Interface Materials (Films, Gaskets, Adhesives)	• Compressible thermal films such as soft metal foils • Formable polymeric gaskets with oriented conductive fillers, reinforcements • Highly improved thermally conductive adhesives
Improved Thermal Management	Lightweight Thermal Straps for Balanced Thermal Design	• Flexible, polymer-encapsulated, thermally conductive carbon fibers • Efficient conductive strap terminations • Integration into structure, co-curing
EMI/EMC Shielding	Integral, Co-Cured Materials	• Metal-coated fabrics, felts, and fibers • Metal foils and meshes • Intercalated fibers
Integration of Fittings and Brackets	High-Strength, High-Stiffness "Isotropic" Materials	• Improved Aluminum/Beryllium two-phase alloys and associated materials characterization database • Lower-cost 3D composites
Lightweight Integrated Electronics Boxes	Composite Electronic Enclosures and Cards	• Highly improved Kz composite materials • Improved box to exterior heat transfer designs • High conductivity, non-metallic card holders, thermal planes,heat sinks
Standardized Design and Analysis Approaches	Material Properties Database and Design Tools	• Standard composite materials database • Laminate thermal conductivity and CTE prediction • On-orbit characterization of newer materials
Low Cost Production	Manufacturing Processes & High-Volume Production Methods	• Automated tape or tow pre-preg layup • Resin-injection molding of fiber preforms • Co-cure integrated structure in single process step • Modular components and standardized tooling • Replace tapes with fabrics

Deployable Structures, Mechanisms, and Launch Vehicle Integration This category addresses technologies for integrating the structure with the launch vehicle and stowing it in the limited volume within a shroud. The principal mission needs are for compact, lightweight, stiffness-critical, deployable structures (e.g. solar arrays and flat-panel phased array antennas), and associated low-shock deployment mechanisms. Current T300/honeycomb rigid solar array sandwich-panel substrates are on the order of 0.5 lb/ft^2, representing about two-thirds of the panel weight. Triaxial weave fabrics for panel sandwich structures have the potential for reducing structural weight 35% while dramatically reducing labor hours for layup. For large antenna reflectors on GEO communications satellites, current technologies promise 0.16 lb/ft^2 (for the 22 x 16-ft MSAT antenna), while studies of inflatable structures indicate the potential for 0.06 lb/ft^2 based on aperture area. For small satellites, and particularly commercial communications satellites launched in stacks, technologies which facilitate compact packaging, reduced shock levels, and reduced launch loads will have large payoffs. These include shape memory metal actuators for deployment and non-pyrotechnic separation nuts for booster release.

Precision Pointing and Stability The principal mission needs in this category are technologies for meeting the precision pointing requirements for high-resolution commercial remote sensing. While current pointing requirements for commercial communications spacecraft are not stressing, the next generation of spacecraft will also have tight pointing and stability requirements associated with the operation of high-bandwidth optical crosslinks. The stability requirement is on the order of 10 micro radians. Relevant technologies which could be used to solve potential vibration disturbance problems include isolation systems and passive or active damping. Technologies for solar array damping will permit the use of lighter-weight solar arrays, significantly reducing the satellite moment of inertia and correspondingly increasing the satellite agility. In addition, on-orbit system identification will be needed for the initial development vehicles in a commercial communications constellation to verify on-orbit performance and enhance mission operations capabilities.

Dimensionally Stable Structures The principal mission needs are for remote sensing and earth science missions. Technologies which reduce the cost and improve the environmental compatibility of low CTE materials will have significant payoffs for these missions. These include silicon-based polymers, cyanate ester polymers, thermoplastics, and lightly-crosslinked, fully cured, thermosets.

Structures and Materials Technology Development Needs

MISSION NEED	SOA TECHNOLOGY	RECOMMENDED DEVELOPMENT
Deployable Structures, Mechanisms, and Launch Vehicle Integration		
Lightweight Panels	Triaxial Weave Fabrics	• High modulus triaxially-woven fabrics • Solar array and phase array panels • Associated materials database
Low-Shock Booster Release	Non-Pyrotechnic Release Mechanisms	• High-preload, simultaneous, non-explosive actuators, shape-memory alloy or spool-initiated
Low-Shock Deployment	Non-Pyrotechnic Release Mechanisms	• Shape-memory alloys, fusible links • Nitinol characterization and standardization
Reduced Stowed Volume	Deployable Struts	• Shape-memory materials • High strain carbon fibers • Lightweight orthotropic composite joint materials
Reduced Stowed Volume	Inflatable Structures	• On-orbit rigidized resins • Inflatable booms, arrays, and sunshades • Inflatable reflector antennas
Increased Reliability	Improved Bearings and Lubricants	• Longer-life bearings and lubricants • Associated database
Reduced Launch Loads	Launch Load Alleviation	• Passive and actively damped booster adapters
Precision Pointing and Jitter Stability		
Precision Pointing	Isolation Systems	• Isolation mounts • Well-developed options as design alternatives
Precision Pointing	Structural Damping	• Tuned mass dampers • Passive and active damping
On-Orbit System Identification	Health Monitoring	• Easily-integrated, standardized, flight-qualified, lightweight sensors, data acquisition system and associated software for 16 channels
Dimensionally Stable Structures		
Improved Environmental Compatibility	Advanced Materials and Resin Systems	• Resin systems with reduced moisture absorption, reduced CTE, improved microcrack resistance • Enhanced thermal conductivity
Reduced Cost	Tailored Design Approaches	• Optimized layups with lower cost carbon fibers and post-processing heat treatments • Low cost carbon-carbon materials processing • Thin-ply (0.5, 1, 2 mil) pre-pregs

2.2.3 Environmental Materials Technology Development Needs

The principal environmental issues for the focus LEO commercial communications and remote sensing orbits are atomic oxygen, debris and micrometeoroid, solar UV, vacuum outgassing, and thermal cycling effects. While this study is primarily focused on structures and structural materials, a comprehensive description of LEO environmental effects is described in Reference 7, "Space Environmental Effects on Spacecraft: LEO Materials Selection Guide".

The technology needs survey results indicate that on-orbit environments remain a significant concern for specific materials and exposures. The survey also revealed an increasing focus on reducing cost by improving the efficiency of manufacturing and processing for environmental coatings and materials. Ease of application and reduction of process steps through methods such as parts count reduction and co-curing are high-priority technology development needs for the planned large constellations of LEO satellites.

AO protection methods are needed to address the synergistic recession and undercutting effects of atomic oxygen combined with particle impacts and solar UV. They include: (1) protective coatings, (2) intrinsically-resistant composite matrix resins, and (3) advanced thermoset polymers, which during the curing process, form an integral protective barrier due to migrating fillers or surface chemical reactions. An effective replacement for Kapton® and FOSR needs to be developed for solar arrays and thermal blankets, respectively. Current practice for AO protection includes coatings and/or added "sacrificial" material thickness. The long-term effects of the AO environment are not precisely known for all of the orbits under consideration, and in the latter case, the sacrificial material may cause contamination problems for sensitive surfaces, such as solar cell covers. Qualified coating processes for protective oxide thin film coatings, such as MgF_2, Al_2O_3, and ITO, also need to be developed for LEO applications.

Contamination and dimensional stability are important issues for remote sensing spacecraft with imaging sensors and optical benches (although, since they are much fewer in number than communications satellites, may be of lower overall priority). New materials and design approaches need to be developed to reduce CTE, reduce outgassing and moisture absorption effects, increase micro-crack/thermal cycling resistance, and enhance the thermal conductivity of optical coatings and metering structures.

Environmental Materials Technology Development Needs

MISSION NEED	SOA TECHNOLOGY	RECOMMENDED DEVELOPMENT
Environmental Resistance	Protective Coatings and Intrinsically Resistant Materials	• Advanced resins - polysiloxanes, epoxy-functionalized siloxanes, other silicon-carbon thermosets, aliphatic and aromatic silanes, ceramic converted polymers, improved silicone paints • Improved, crack-free SiO2, diamond coatings, sputtered alloys
Integrated, Protective Designs	Integrated Surface Barriers	• Thermoset polymers with surface-migrating protective fillers or surface barriers • Co-cured films or foils for composites
Thermal Control Coatings and Mirrors	Advanced Processes	• Low-cost diamond coating processes • Cerium-doped optical surface reflectors • Improved integration, ease of application
Environmental Effects & Life Certification	Enhanced Understanding of Environments & Environmental Effects	• Continued flight experiments using current and evolving aerospace materials and design approaches • Continued development of materials environmental effects characterization database • Design guides

In the area of thermal cycling, development needs include: (1) higher emittance coatings, (2) higher conductivity coatings such as deposited diamond coatings, (3) radiator coatings with increased resistance to thermal and fatigue cycling effects, and (4) increased understanding of the changes in absorptivity and emittance of thermal control coatings during their lifetime in the LEO AO and radiation environment.

As implied above, the importance of understanding the environment, characterizing its effects on spacecraft performance, and establishing good design practices should not be underemphasized, and is symbiotic with the development of new materials. Flight experiments, design guides, and materials databases need to be continually updated to reflect the evolution of new materials and evolving design approaches. The research community needs to continue to work closely with the spacecraft industry to maintain the focus of this important activity on current structural and environmental design practices, materials, laminates, and application processes.

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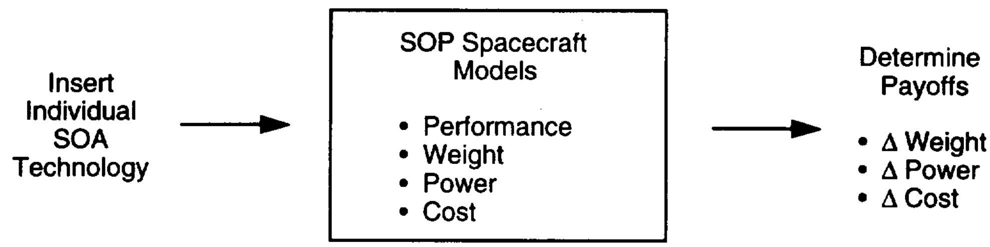
2.3 SYSTEMS ANALYSIS APPROACH FOR TECHNOLOGY TRADES



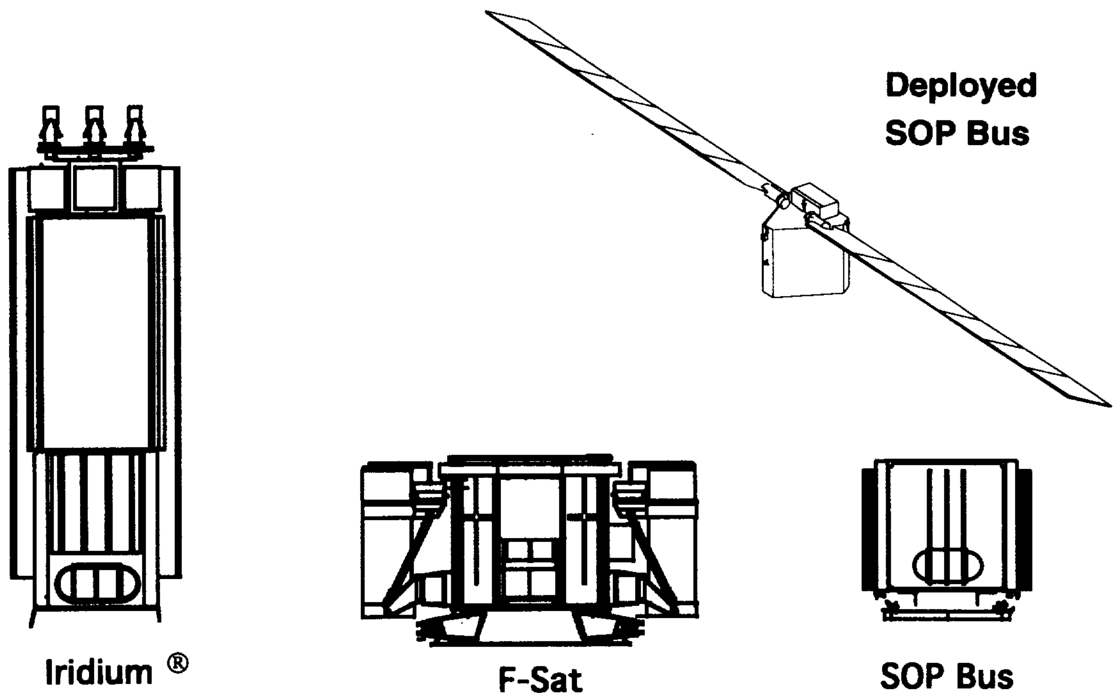
This section discusses the approach for conducting trade studies to determine the benefits of developing advanced SOA technologies to meet a selected set of the technology needs described in the previous section. Two slightly different approaches were used, based on whether the technologies are enhancing, or other, as defined below:

> *Enhancing Technology Development Trades* Determine the system benefits and incremental cost savings per satellite based on substitution of an SOA technology into a typical SOP design to meet *equivalent* system performance requirements.
>
> *Other Technology Development Trades* Identify the benefits of SOA technologies for which the SOP model does not apply that enable *increases* in performance or solve, specific, unique problems.

The approach for conducting the enhancing technology trades is based on developing a reference, SOP, multimission small spacecraft bus design capable of performing the focus missions. This reference SOP spacecraft design and its associated performance, weight, power, and cost form a baseline system model for perturbation analyses, as described in Section 2.3.1. The analyses are performed by individually substituting the SOA technologies into the reference SOP spacecraft model, and calculating the associated changes in weight, power consumption, and cost on a per satellite basis. Using metrics developed in Section 2.3.2 for the marginal value of a pound ($K/lb) and a Watt ($K/W) for each focus mission, the changes are converted into cost numbers. As described in Section 2.3.3, the cost numbers for weight, power, recurring cost, and non-recurring cost (in 1993 $) are then summed to determine the relative payoff (or loss) associated with incorporating the SOA technology into the spacecraft design. Other benefits, which could not be costed, are noted individually in Section 3.1.

For the other technology trades, the approach is based on knowledge of the needs and trends outlined in Sections 2.1 and 2.2. Assumptions for these trade studies are discussed along with each trade in Section 3.2.

2.3.1 Reference State-of-Practice Design



2.3.1 Reference State-of-the-Practice Design The chart above compares the size of the reference SOP small satellite bus design with stowed versions of the Iridium® spacecraft and the LMSC F-Sat medium-class satellite bus. Modifications can be made to the reference SOP multimission bus structure to create a remote sensing or communications spacecraft. The payload and associated payload structure are attached forward of the interface plane. The specific assumptions used to tailor the SOP design for communications or remote sensing system technology trades are described along with each trade study in Section 3.1.

The remaining charts in this section describe different aspects of the SOP multi-mission bus. The next chart lists the principal performance requirements and includes an exploded view of the reference design and associated components. The basic requirements for attitude control are based on a Big LEO communications mission, while the tighter requirements associated with the star tracker option are appropriate for a remote sensing mission. Overall, the requirements reflect the SOP for a high-performance multimission bus. The charts which follow describe the mechanical and thermal subsystems, concluding with the relative weight and cost breakdowns for the bus.

SOP Small Satellite Bus
Requirements and Reference Design

Launch Vehicles:	• LEO, single launch:	LLV-1/2, Taurus, Conestoga, Titan II
	• LEO, multiple launch:	Delta II, Atlas II, LLV-3, Proton
	• GEO, single launch:	LLV-3, Delta II, Atlas II, Proton
	• GEO, multiple launch:	Proton

Physical Characteristics
- 700 - 800 lb. dry
- Triangular shaped graphite/epoxy structure
- 3 point payload interface
- Payload weights to 1000 lb.

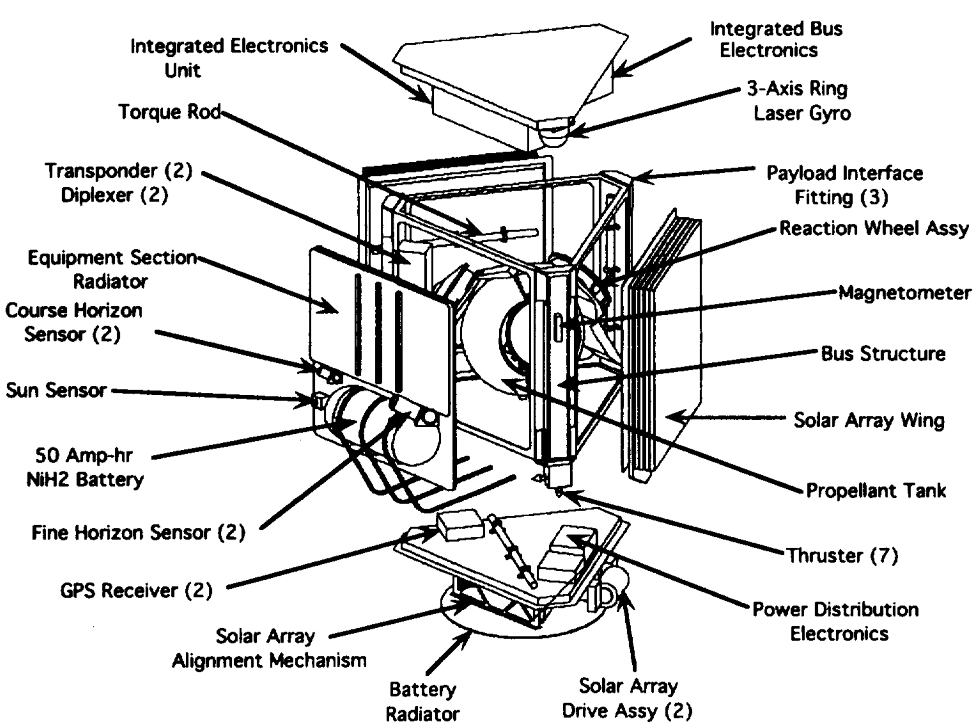
Electrical Power
- 2 GaAs solar arrays on 2-axis gimbal
- 1.5 kW at arrays, BOL
- 300 W avr. to payload EOL, LEO
- 50 Amp-hr Single Pressure Vessel NiH2 battery

Attitude Control
- 3-axis stabilized
- Attitude knowledge to 0.25° (0.01° w/ star tracker option)
- Attitude control to 0.35° (0.02° w/ star tracker option)

Propulsion
- Blowdown hydrazine monoprop, 200 lb.

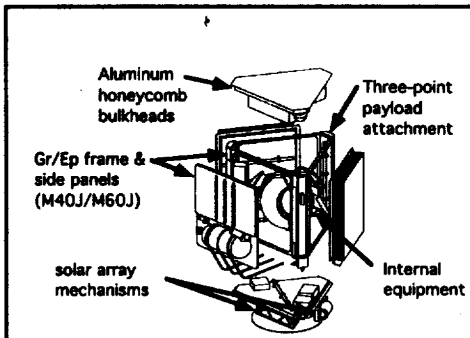
T,T & C
- MIL-STD-1750A processor (16 bit ASCM), 2.5 MIPS at 20 MHz
- 8 Mb RAM
- 2 kbps SGLS/STDN uplink commands
- 64 kbps SGLS/STDN downlink for health and status
- GPS for navigation
- MIL-STD-1553 interface to payload
- Growth capability for serial, analog & discrete interfaces

Life
- 7 year design life, 5 year MMD



Structures & Mechanisms Subsystem

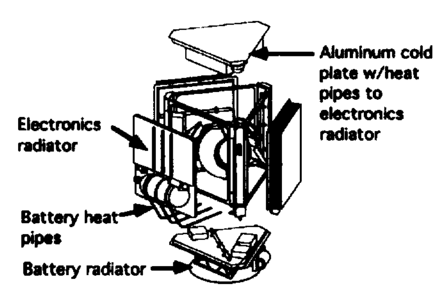
- Graphite / Epoxy frame.
  - low CTE
  - High stiffness/weight
- Graphite skinned side panels.
- Ply of copper mesh for EMI shielding.
- Aluminum honeycomb upper and lower bulkheads.
- Three point payload attachment allows for statically determinate interface.
- Mechanisms
  - Solar Array Drive
    - Two axis, elevation-azimuth
    - Stepper motors, resolvers
    - Slip ring on elevation axis
  - Solar Array Alignment Mech.
  - Solar Array Deployment
    - Paraffin actuators - low shock
    - Pantograph mechanism
  - Spacecraft Separation
    - Three sep nuts
    - Three push off springs



•Structure sized for 1000 lb. payload.
•Bus weight = 700 - 800 lb.

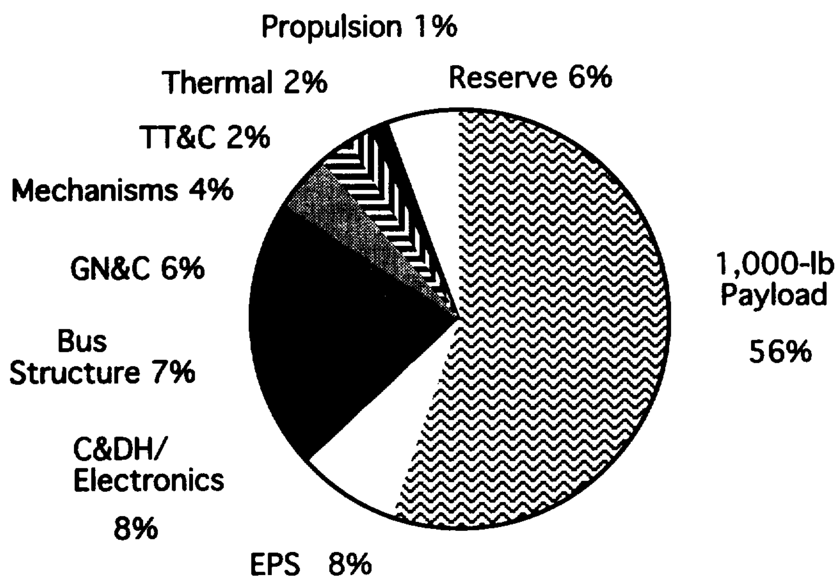
Thermal Subsystem

- Cold biased system:
  - FOSR covered radiators
  - Heaters
  - Multi-Layer Insulation
- Heat pipes move heat to radiators from:
  - Battery
  - Electronics on upper bulkhead
- Minimal thermal interaction with payload desired.

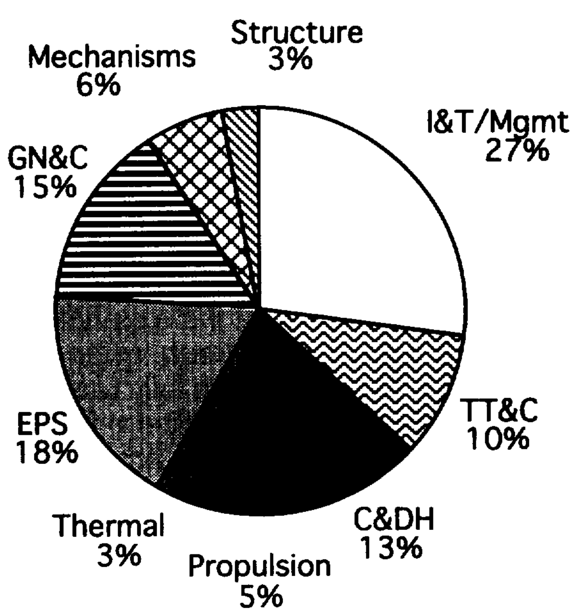


SOP Bus Subsystem Mass & Recurring Cost Breakdown

MASS



RECURRING COST



2.3.2 Metrics Development Given the wide variety of system design approaches and mission orbits for the communications and remote sensing satellites described in Section 2.1, the consistent development and use of weight and power metrics which are applicable in a general sense is challenging. Since the structure is an integrating element for the satellite, each bus can be driven by an array of different requirements. Thus, the use of structural metrics often results in "apples and oranges" comparisons. The facing charts describe the strong dependence of weight/cost metrics on individual system characteristics, and how the metrics can vary widely over the time of satellite development.

Prior to PDR, the satellite system developer will include in his/her system trades the cost and performance capability of the available launchers. The developer can then tailor the satellite system design, and the selection of any advanced technologies, to arrive at the best solution. Recent examples of this include the downsizing of the EOS spacecraft from an Atlas to a Delta to save cost, and the upsizing of the SSTI Lewis and Clark satellites from a Pegasus to an LLV1 for increased mission performance. After PDR, the cost/weight metric varies widely, and the selective use of advanced technologies to reduce weight will depend largely on the remaining margin available in the satellite weight budget. In the case where there is little margin, the payoff for technologies which can be "retrofitted" into the satellite design can be much higher than the launch cost metric used herein.

In this study, metrics were developed for use during the system design phase, prior to PDR, when the satellite design and launch vehicle trades are being conducted. The graphs which follow illustrate how cost/weight metrics were developed for LEO and GEO missions using the slope of the launch cost versus launch weight curve. Points on the graphs indicate the capabilities of the expendable launch vehicles (ELVs) listed to the right of each graph, given the orbit and insertion assumptions shown. This slope is a fully realizable metric associated with being able to downsize the satellite(s) for launch on a smaller, less costly, ELV. Nonetheless, the cost savings is only realized if enough weight is saved to switch from one ELV to another.

A different approach, which was not used in this study, derives the cost/weight metric by dividing the launch cost by the throw-weight for each launch vehicle. While this is a useful performance metric for the launch vehicle manufacturers, it is inappropriate for satellite developers. The "cost savings" associated with a weight reduction using this metric approach is intangible and cannot actually be attained.

Development of Weight/Cost Metrics

Weight/Cost Metrics Difficult to Quantify

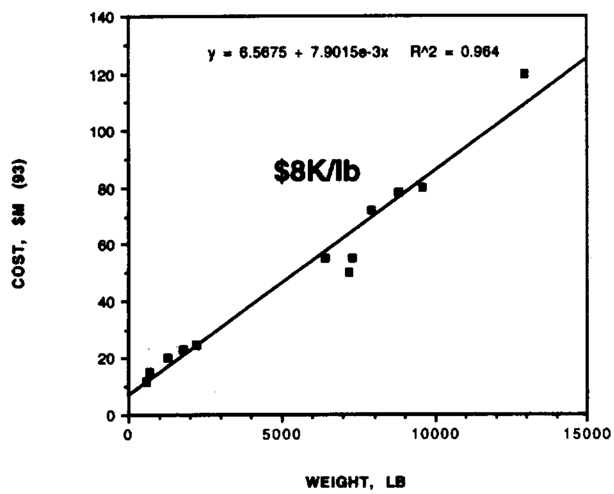
- Strong Dependence on Orbit Altitude, Orbit Type, Inclination and Design Life
- Strong Dependence on Individual System Characteristics (Including Revenue Model)
- Many Satellite Designs are Constrained by Both Weight and Volume (Shroud Diameter or Height)
- Accurate Competitive LV Pricing Data Difficult to Obtain
- Many Different Approaches to Meeting Orbit
  - Direct Insertion, Apogee Kick Motors, On-Board Propulsion
- Dividing Launch Mass by Launch Cost is Not A Realizable Metric

Development of Weight/Cost Metrics

Time Value of Weight Savings Varies

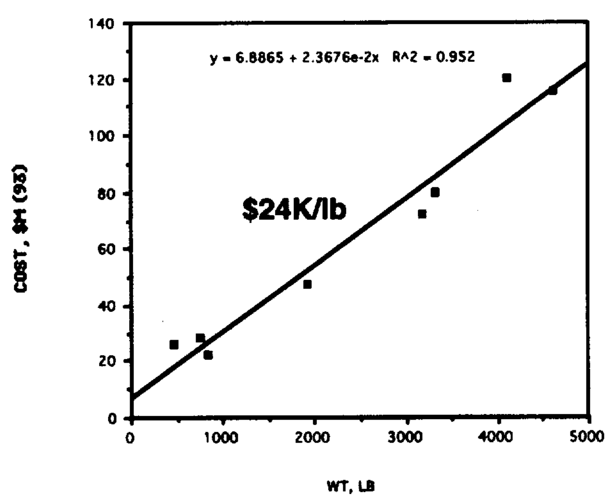
- SOA Technology and NRE Investment Trades Made Before PDR
- Launch Vehicle & Satellite Sizing Trades Made Before PDR
  - Weight & Volume Compatibility
  - Number of Vehicles per Launcher
  - Satellite Form Factor Constrained
- Realizable Weight/Cost Metrics are Based on Incremental Costs and Benefits of Re-Sizing Satellite for Next Size of Launcher
  - Cost Savings due to Downsizing on EOS from Atlas to Delta
  - Performance (Design Life) Gain due to Upsizing SSTI "Clark" from Pegasus to LLV1
- Value of Weight Savings is Highly Nonlinear After CDR
  - Near Zero if Spacecraft is Well Within Weight Budget
  - Exponentially Higher if Significant Redesign Needed

Marginal Launch Cost to LEO



- LEO Orbit Assumptions
  - Direct Insertion to 705 Km
  - Sun-Synchronous Circular Orbit (i= 98.2°)
  - Appropriate for Aries, Iridium, Teledesic, EOS, ~Globalstar
- Small ELV's for LEO
  - Pegasus XL, LLV-1, Connestoga, Taurus, LLV-2
- Medium ELV's for LEO
  - Delta, Atlas II, Atlas IIAS, Titan, Ariane 44P, Ariane 42P, Ariane 40
  - Most Small Satellites Launched on Medium LV'S

Marginal Launch Cost to GEO



- GSO Orbit Assumptions
  - Launch into GEO Transfer Orbit
  - Apogee Kick Motor Used for Circularizing Burn to Final 35,786 Km, 0° Inclination Orbit
  - Direct Insertion for Some Large LV's/Upper Stages
  - Appropriate Cost for MSat, Spaceway
  - Upper Cost Bound for Odyssey, AMSC
- ELV's for GSO
  - Taurus XLS, LLV-2, Delta 7925, Atlas II, Atlas IIAS, Ariane 42P, Ariane 44L

Other Design- and Mission-Dependent Measures:

- Additional Payload and Associated Revenue
- Use of Lower Cost, Heavier Components
- Reliability and Design Life
  - Additional Fuel (1 lb = Up to 2 Months in LEO)
  - Additional Solar Cells
  - Additional Shielding (Lower Radiation Environment for Commercial Parts)
  - Additional Redundancy (Add Extra Components)
- Performance
  - Agility
  - Stiffer Structure
- Additional Weight Margin for Reduced Risk
  - Reduced Analysis and Redesign Cost
  - Margin for Problem-Solving on Tight Schedule

There are a myriad of ways that weight savings in one area can be used for increased benefits in other areas. However, in many cases, weight savings need to be undertaken in large enough discrete increments to be worthwhile. The weight savings could be used for additional components, in which case the increment would be the component weight. In other cases, the benefits of weight savings are continuous, such as in the case of inertia reduction for agility. Finally, extra weight margin can be used early in the design phase to offset weight gains later in the program, reducing the risk of having to undertake costly weight-reduction measures.

Development of Power/Cost Metrics

General Quantitative Measure:

- EPS Subsystem Cost for Orbit-Average Power Demand at EOL
  - LEO Polar 705 Km $8 - 12K/W
  - Geosynchronous $10 - 14K/W
- Effects of Degradation and Eclipse Included
  - 12% Degradation Assumed over 7-Year Life
  - Maximum LEO Eclipse Duration of 35 minutes is 40% of orbit period
  - Maximum GEO Eclipse Duration of 70 minutes is 5% of orbit period
- Effects of EPS Subsystem Weight and Recurring Cost Included
  - Solar Array
  - Battery
  - Charge Control
  - Power Switching and Distribution

For completeness, power/cost metrics were developed for the reference focus mission orbits. This metric is difficult to compare for different missions, as the duration of the eclipse, the amount of solar cell degradation, and the launch cost for the weight of the Electrical Power System (EPS) must be taken into account. Since the EPS subsystem is sized for End-of-Life (EOL) performance, the type of cells and the associated degradation environment must be considered. In addition, for commercial communications missions, the ratio of peak power and orbit average power can vary widely depending on the required surge capability and the surge duration. Finally, the timing in the orbit period when the power is needed is important, as the system may be battery-limited, such that power savings during eclipse is more valuable than power savings during daylight.

The ranges above are typical for GaAs solar cells on a 1200 to 1500 Watt (BOL) rigid array using the assumptions listed.

Marginal Value of a Watt

Other Design- and Mission-Dependent Measures:

- Additional Payload Power and Associated Revenue
  - More Traffic and Better Sound Quality for Communications
  - More Pictures per Orbit for Remote Sensing
- Reliability and Design Life
  - Additional Heater Power (for Cold Spots)
  - Less Heat Dissipation (Hot Spots, Reduce Radiators and Heat Pipes)
  - Lower Max Operating Temperatures for Electronics
  - Low Power Safemode
  - Battery Life (Lower DOD)
- Smaller Solar Arrays
  - Lower Inertia for Agility, Reduced Disturbances (Kb)
  - Smaller Stowed Area, Fewer Mechanisms
  - Increased Field of View
- Additional Power Margin for Reduced Risk
  - Reduced Analysis and Redesign Cost
  - Margin for Problem-Solving on Tight Schedule

There are also many ways that power savings in one area can be used for increased benefits in other areas. As in the case of weight savings, power savings must be undertaken in large enough discrete increments in to be worthwhile. For example, solar arrays are made up of "strings" of solar cells capable of producing a fixed amount of power. To reduce the size of a solar array, the power savings must be large enough to eliminate an entire string of solar cells (or two if the spacecraft has two common arrays). In the case of components which are powered during eclipse, power savings can be converted into longer life for the battery through reduced depth of discharge. Finally, extra power margin can be used early in the design phase to offset anticipated power gains, reducing the risk of having to undertake costly design changes late in the program.

2.3.3 Calculation of Relative Technology Benefits A simplified approach for conducting the technology payoff analyses is described on the next chart. Using the SOP bus design and metrics described in the previous section, SOP system models were constructed for the two focus LEO communications and remote sensing missions. The orbit assumptions, satellite quantities, and weight and power metrics are shown. An additional GEO system model was developed for evaluating the equivalent technology payoff for GEO communications missions. This model extrapolates the SOP LEO bus design to GEO, i.e., the only differences are the added launch cost to GEO (e.g. $24K/lb instead of $8K/lb) and the reduced number of satellites. The GEO model also provides an upper bound for LEO & MEO systems which involve fewer satellites, higher orbits, and/or a higher cost/weight metric ($/lb).

The trades are conducted by individually substituting each advanced SOA technology into the SOP satellite point designs and calculating the associated changes in weight, power consumption, and cost (in 1993 $) on a per satellite basis. For example, if a radiator is chosen as the focus component, the SOP aluminum radiators on the spacecraft could be replaced with composite radiators designed to meet the same performance requirements, and the associated system weight and cost changes calculated. This step essentially amounts to a set of perturbation analyses for the incorporation of individual SOA technologies into the satellite design. Additional assumptions made to conduct a quantitative analysis are noted on a case by case basis. Other benefits which are important, but which could not always be quantified (e.g., simplified design, increased reliability, reduced integration and test time, etc.) are noted separately.

The final step is to complete the perturbation analysis by converting the weight, power, recurring cost, and non-recurring cost deltas to a combined relative payoff per satellite based on the equation at the bottom of the chart. An example payoff table is shown. The relative payoff per satellite can then be used to rank the benefit of using the advanced technologies included in this study.

The overall approach provides a structured way of quantifying the general benefits or drawbacks of incorporating specific, enhancing, advanced technologies into a typical system. However, caution is warranted in that the quantitative results presented herein are highly dependent on the assumed requirements (e.g. orbit altitude and inclination) for each specific mission.

Calculation of Relative Technology Benefits

Insert Individual SOA Technology into SOP Models

"Big LEO" Comm SOP Model
- 705 Km Polar Orbit
- 50 Satellites
- Launch Cost \$8K/lb
- Power Cost \$10K/W
- \$ REC, \$ NRE

GEO Comm SOP Model
- Geosynch (i=0) Orbit
- 10 Satellites
- Launch Cost \$24K/lb
- Power Cost \$12K/W
- \$ REC, \$ NRE

LEO Remote Sensing SOP Model
- 705 Km Sun Synch Orbit
- 2 Satellites
- Launch Cost \$8K/lb
- Power Cost \$10K/W
- \$ REC, \$ NRE

Calculate Relative SOA Technology Payoffs

- Payoff per Satellite =
  (ΔWt x \$/lb) + (ΔPwr x \$/W) + (Δ Rec Cost)+(ΔNRE/ # of Satellites)
- Other Benefits (Design, I&T, Reliability, etc.) Noted

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY Weight Savings (lb)	Cost Savings ($K)	RELATIVE PAYOFF PER SPACECRAFT LEO Comm Spacecraft (~50) ($K)	GEO Comm Spacecraft (Equiv) (10) ($K)	LEO Rem Sens Spacecraft (~2) ($K)
M40J & P120 vs. Aluminum	61	300 NRE 75 RE	570	1,570	710

In the development of a specific commercial satellite system, an additional step would be added to the trade studies to compute the return on investment. This calculation would include the development or qualification cost for the SOA technology, if any (one could assume that there is none if the technology is fully developed by NASA). Most importantly, the return on investment calculation would include the commercial satellite system revenue model to directly relate performance versus cost. This revenue model would include all important metrics for the specific system, allowing advanced technology trades to determine the effect of selecting an advanced technology on the potential profits produced by the commercial venture.

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3.0 STRUCTURES AND MATERIALS TECHNOLOGY TRADES

To gain further insight into the characterization and quantification of the benefits of developing selected advanced SOA technologies, abbreviated trade studies were conducted for 11 out of the 25 recommended SOA technology development opportunities outlined in Section 2.2. The trades were divided into two groups: "enhancing" and "other", based on whether or not the reference SOP spacecraft baseline applied. For the enhancing technology trades, the spacecraft performance requirements are held constant (as defined in Section 2.3.1) while the cost benefits are quantified. For the other trades, the SOA technologies enable increases in performance or solve specific problems (e.g. jitter) which are documented.

3.1 ENHANCING TECHNOLOGY DEVELOPMENT TRADES

The mission needs, SOA technologies, and focus components for the eight enhancing technology trades are shown below. Each trade is discussed in a separate subsection which describes the mission need, the SOA technologies included in the trade, additional trade assumptions, and recommendations for development. Focus components using the SOA technology are substituted into the reference SOP spacecraft model and the payoff, if any, is calculated using the approach described in Section 2.3. Each subsection concludes with a summary chart describing the relative quantitative payoff of each technology trade for the focus missions.

Mission Need	Enhancing SOA Technology	Focus Component(s)
Integrated Graphite/Epoxy Bus	Technologies for Lightweight Structure with Integrated Thermal, EMI/EMC, and Structural Fittings	Multi-Mission Graphite/Epoxy Bus Structure
Improved Heat Dissipation	Integration of High Conductivity Materials into Bus Structure	Spacecraft Thermal Panels and Radiators
Lightweight Panels for Stiffness-Critical Solar Arrays and Phased Array Antennas	Triaxially-Woven Fabric for Single-Ply Balanced Laminates	Rigid Solar Arrays Phased Array Antennas
Reduced Shock Environment	Non-Pyro Release Mechanisms	Booster Release Mechanisms
Balanced Thermal Design - Eliminate "Hot Spots"	Carbon Fiber Thermal Straps	Thermal Straps
Integrally EMI/EMC-Shielded Composites	Coated Fabrics, Felts, and Fibers Foils	Shielded Bus Panels
High-Strength, High-Stiffness Materials for Fittings & Brackets	New Be-Al Alloys	Booster Interface Fittings
Integrated Electronics Boxes	Composite Electronics Boxes	Integrated Electronics Box

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3.1.1 Integrated Composite Bus Structures

- Integrated Structural Design Technologies
  - Thermally Conductive Materials
  - Lightweight Radiator Panels
  - Integral EMI/EMC Shielding
  - Triaxially Woven Fabric for Panel Facesheets
  - Thermal Straps
  - Lightweight Isotropic Materials for Fittings

- Integrated Structural Design Tools
  - Materials Characterization Databases and Design Guides
  - Analysis Techniques for Laminate Thermal Conductivity and CTE Prediction

3.1.1 Integrated Composite Bus Structures This trade addresses the need for integrated structural design solutions described previously in Section 2.2.1. While it is commonplace to use composite materials in the primary structure of GEO satellites (where the payoff for reduced weight is three times as high), Aluminum is still used in many LEO commercial communications and remote sensing satellites. One reason for this is because of the difficulty of meeting the additional requirements placed on the structure beyond strength and stiffness. In particular, one of the greatest challenges is meeting the heat dissipation requirements for high performance small satellites.

It is anticipated that the biggest payoffs associated with building satellite bus structures using composite materials will be realized when a integrated design approach is taken. That is, the composite structural components should, with a minimum number of discrete manufacturing procedures, incorporate the appropriate fibers, fillers, metal mesh, heat pipes, films, or coatings to meet the applicable system requirements for EMC, thermal control, AO protection, heat transfer, balanced CTE, dimensional stability, etc. This approach will, in general, reduce costs by reducing the total number of processing steps.

Design tools and methodologies need to be developed to support this integrated approach. In addition, developing standard designs for both tooling and hardware, which can be used for multiple satellite designs (by scaling up or down) has a potential for very large payoff by reducing non-recurring costs.

Relative Weight and Cost Comparisons

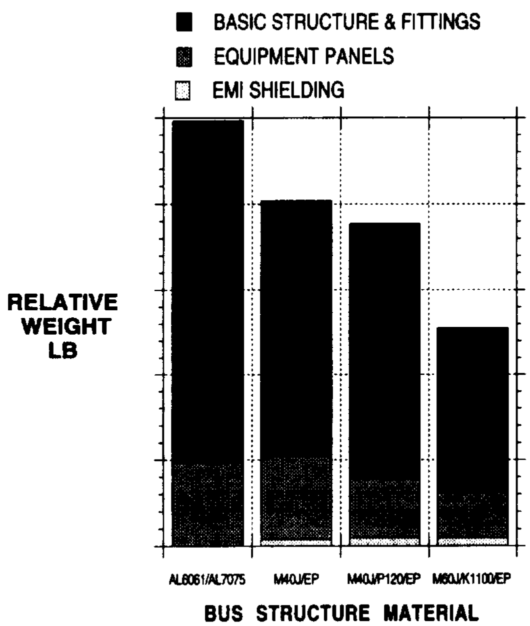


FIGURE A

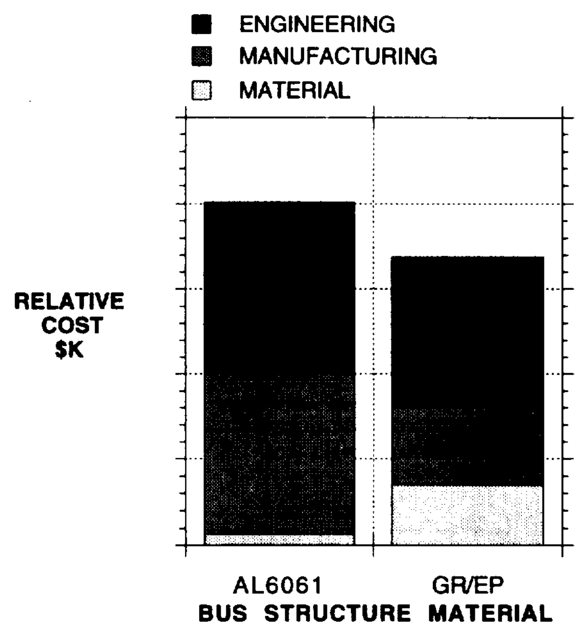


FIGURE B

Constructing a small satellite bus structure with carbon fiber composite materials offers advantages in both weight and cost compared to all metallic structures. In Figure A, it can be seen that the total weight decreases as the stiffness of the building material increases, i.e. K1100 > P120 > M60J > M40J >6061 Al. The addition of EMC materials required for composites (but not metals) does not significantly affect the weight. Figure B shows the relative cost differences between composite bus structures and metallic structures. While composite raw materials are higher than lightweight metals, this cost is more than offset by the significantly higher manufacturing costs which are related to the larger number of piece parts required and the machining costs associated with those parts.

Bus Structure Cost and Weight Trade

BUS STRUCTURE COST TRADE — GR/E VS. 6061 AL

	GR/E STRUCTURE		6061 AL STRUCTURE	
	HRS	$	HRS	$
ENGINEERING				
• TRADE STUDIES	1800		1800	
• DESIGN	2290		3200	
• ANALYSIS	3150		2700	
• TOOL DESIGN	400		1540	
• MFG SUPPORT	1210		800	
MANUFACTURING				
• DETAILS & SUB-ASSY	3190		6600	
• FINAL ASSEMBLY	370			
• TOOLING	960		2780	
MATERIAL				
• BASIC STRUCTURE	—	206,000	—	9,300
• EQUIPMENT PANELS	—	32,800	—	382
TOTALS	13,370	238,000	19,420	9,682

WEIGHT COMPARISON (Lbs)

	Gr/E M40J/P120	ALUMINUM 6061/7075	Δ WEIGHT(%) PENALTY FOR ALUMINUM
BASIC STRUCTURE	105	139	33
EQUIPMENT PANELS	56	87	55
BRACKETS & FITTINGS	6	8	33
EMI SHIELDING	6	0	—
TOTAL	173	234	35

MATERIALS COSTS

M40J/EPOXY = $140/Lb	6061 Al = $2/Lb
P120/EPOXY = $1500/Lb	7075 Al = $3/Lb
K1100/EPOXY = $2500/LB	

For small satellites, the goal is to develop and demonstrate more efficient ways to build bus structure components without sacrificing stiffness and strength.

Weight and cost comparisons are quantified in the above tables relative to a typical small satellite bus structure. Both labor hours and material costs for graphite/epoxy and aluminum are shown. For primary components such as the basic structure, the equipment panels, and even for some brackets and fittings, there is a distinct advantage in replacing metallic parts with composite parts. It should be emphasized, however, that once composites are selected for one component, then CTE mismatch avoidance requirements may dictate the use of composites everywhere.

Recommendations for Development

- DESIGN FOR REDUCTION OF PART COUNT
  - ONE-PIECE MODULAR COMPONENTS
  - CO-CURED STRUCTURES

- REDUCE HAND LABOR HOURS
  - REDUCTION OF PLY COUNT - TRIAX OR LIGHTLY FILLED FABRICS
  - BATCH PROCESSING

- AUTOMATION
  - PLY CUTTING
  - TAPE LAYUP OR TOW FIBER PLACEMENT

- INTEGRATED ASSEMBLY- ASSEMBLE AND CURE SEVERAL COMPONENTS IN ONE PROCESS STEP:
  - EMI/EMC MATERIALS
  - STIFFENERS
  - THERMAL CONTROL MATERIALS
  - HEAT PIPES
  - HONEYCOMB

- LOW COST TOOLING APPROACHES
  - OPTIMIZE TOOLING MATERIAL FOR HIGH VOLUME OR LOW VOLUME
  - STANDARDIZE TOOLING COMPONENTS

Developing manufacturing methods which result in reduced hand labor hours will be the biggest cost saver. This can be accomplished by increased use of automation, modular component design, integrated component design, and co-curing of structural components.

Lower cost tooling concepts also need to be developed and demonstrated. Standard tooling components such as corner angles and flats that can accommodate scaling in length and width as well as variable ply build-ups could help accomplish this. Also, an optimization study, which would tell the designer which tooling material to use when X number of parts is required and cured at temperature T for H hours.

System-Level Benefits
Integrated Composite Bus Structure

Analysis Assumptions

- M40J Primary Structure
- P120 Equipment Panels
- Copper Mesh EMI Shielding
- NRE for Design, Analysis, and Tooling Amortized over Number of Satellites

Other Benefits

- Cost Savings Due to Automation for Large Quantities

Relative System Payoff

	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY		RELATIVE PAYOFF PER SPACECRAFT		
TECHNOLOGY	Weight Savings (lb)	Cost Savings ($K)	LEO Comm Spacecraft (~50) ($K)	GEO Comm Spacecraft (Equiv) (10) ($K)	LEO Rem Sens Spacecraft (~2) ($K)
M40J & P120 vs. Aluminum	61	300 NRE 75 RE	570	1,570	710

The focus component of this trade is an integrated small satellite bus structure. Although the reference SOP spacecraft structure described in Section 2.3.1 is composite, this trade compares the benefits of a highly integrated design versus the case where aluminum cannot be used because of the aforementioned integration issues. A graphite epoxy structure using M40J and P120 fibers and integral copper mesh EMI shielding is compared with an all-aluminum design. Hand layup of the graphite epoxy structure is assumed, although it is noted that there could be further cost savings through the use of automation for large production quantities. The other costing assumptions are described in the previous charts.

The results indicate a substantial savings in being able to substitute graphite epoxy for Aluminum . The largest relative savings on a per spacecraft basis is in GEO, due to the higher launch cost of placing weight in GEO orbit. The LEO remote sensing cost savings is higher than that for the LEO communications spacecraft because the $300K NRE savings is amortized over 2 spacecraft instead of 50. If one were to calculate the equivalent system (constellation) savings for each of the three cases, the results would be $28.5M for LEO communications, $15.7M for GEO communications, and $1.42M for LEO remote sensing.

3.1.2 Passive Composite Radiators

PANEL REQUIREMENTS (For typical 24 in x 42 in panel):

- Ability to dissipate up to 20 watts/ft^2 power
- Meet structural load requirements: nominal 75 lb equipment weight
- Meet panel stiffness requirements: up to 50 Hz
- Meet EMI/EMC and grounding requirements
- Compatibility with heat pipes, when required (CTE mismatch effects)
- Ability to enhance matrix conductivity without reduction in laminate mechanical properties.

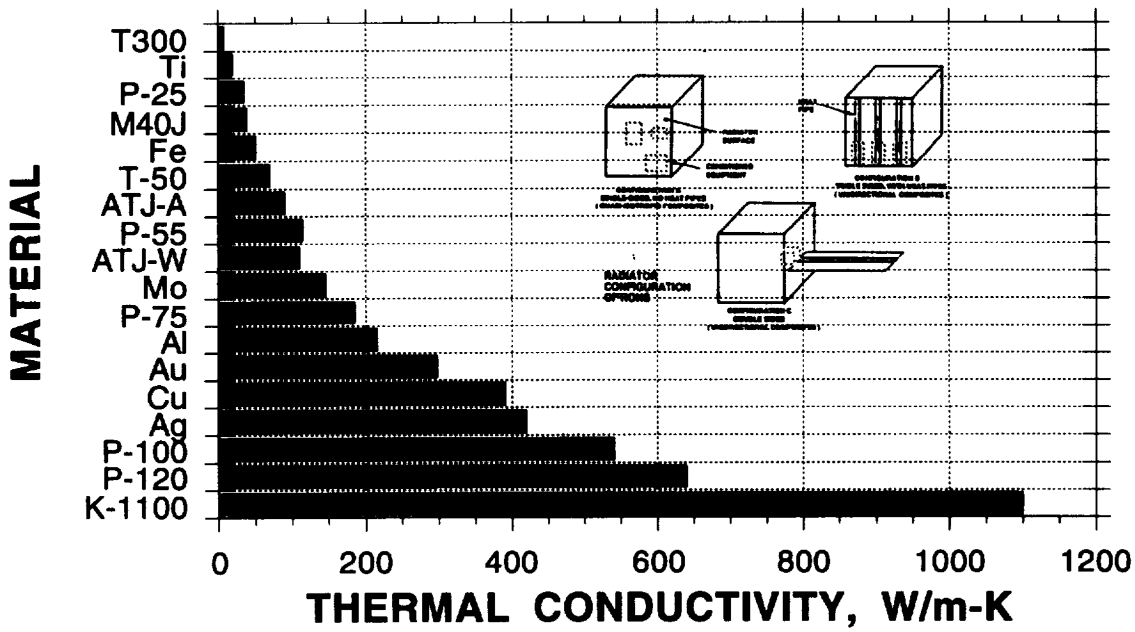
ADVANTAGES:

- Potential for eliminating or greatly reducing the need for heat pipes
- Potential for significantly increasing in-plane thermal conductivities(compared to Al)
- Significant weight savings
- CTE match to composite bus structure
- Allows the co-curing of stiffeners to reduce secondary processing

3.1.2 Passive Composite Radiators This trade addresses the need for improved heat dissipation in composite structures, particularly in smaller satellites where the relative thermal density is high. With the introduction of high conductivity carbon fibers to the market, which exhibit thermal conductivities 2 to 3 times that of copper, there now exists the potential to construct carbon fiber composite radiator and equipment panels for satellites which efficiently perform heat removal functions as well as meeting structural requirements.

A typical small satellite electronic equipment mounting panel would be required to support approximately 75 pounds of equipment and maintain a first flexural frequency of 30 to 50 Hz. High power density boxes can generate 40 to 120 watts which translates to an average heat production of about 20 watts/ft^2 for a typical 24" x 42" panel. It has been demonstrated that a panel constructed from P120S fiber in an epoxy matrix will marginally meet the above requirements for heat dissipation (without the use of heat pipes), strength, and stiffness. Average box temperatures can be maintained within + 5/-0 °F compared to boxes mounted on an aluminum panel. The potential for improved thermal performance exists with the use of higher conductivity fibers such as K1100. Composite radiator-equipment mounting panels also exhibit very low CTE, which makes them much better suited for attachment to an all composite bus structure than aluminum panels.

Comparison of Conductive Fibers and Metals

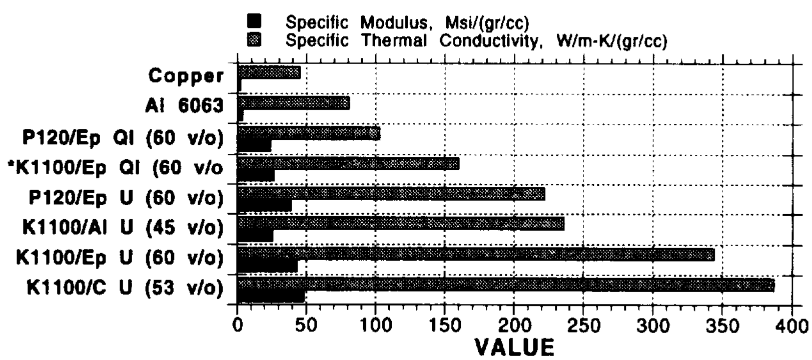


The relative conductivities of structural and thermal materials that are typically used in small satellite construction are shown above. The newest high conductivity carbon fiber, K1100, was introduced to the commercial market in 1994 and shows great potential in thermal management applications.

As indicated in the chart, thermal/structural materials can be incorporated in three different radiator configurations: (1) As a radiator mounting panel for equipment, (2) As a radiator mounting panel with embedded heat pipes that conduct the heat away from the equipment, and (3) as a cantilevered radiator.

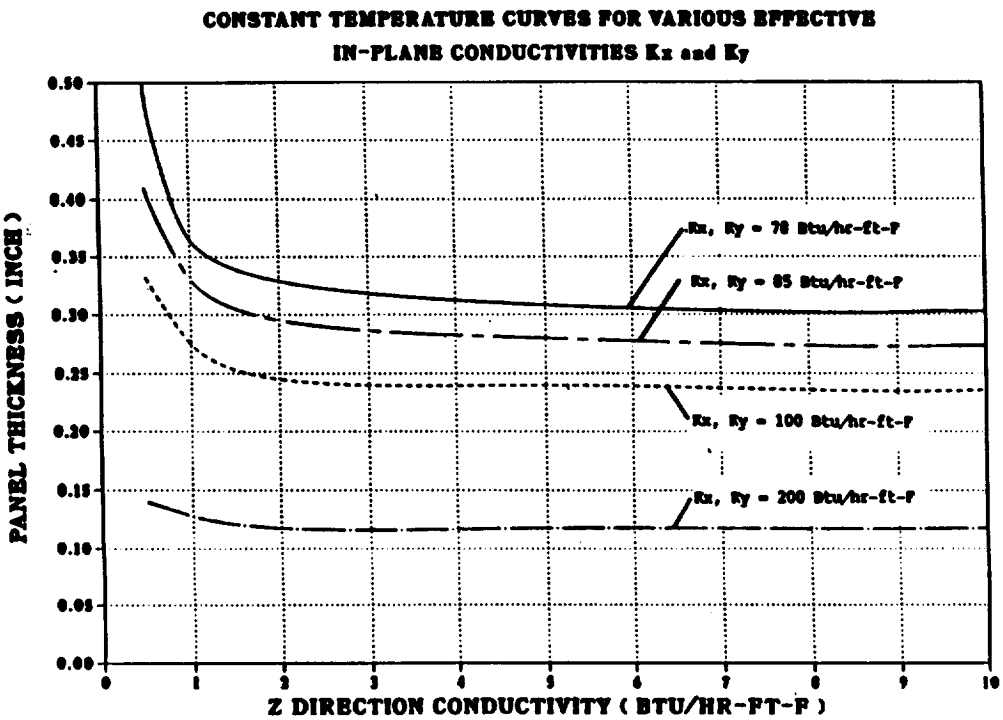
Comparison of Radiator Materials

MATERIAL	LONGITUDINAL K (W/m-°K)	MODULUS (msi)	CTE (ppm/°K)	DENSITY (g/cc)	SPECIFIC CONDUCTIVITY (W/m-°K/p)
Al 6063	218	9.9	23	2.7	81
Copper	400	17	17	8.9	45
P120/Epoxy Uni (60 v/o)	396		-0.5	1.8	222
K1100X/Al Uni (45 v/o)	590	64	0.5	2.5	236
K1100X/Epoxy Uni (60 v/o)	620	81	-0.67	1.8	344
K1100X/C Uni (53 v/o)	696	87	—	1.8	387
P120/Epoxy QI (60 v/o)	183	25	-0.3	1.8	103
*K1100X/Epoxy QI (60 v/o) (*Projected)	287	39	-0.4	1.8	160

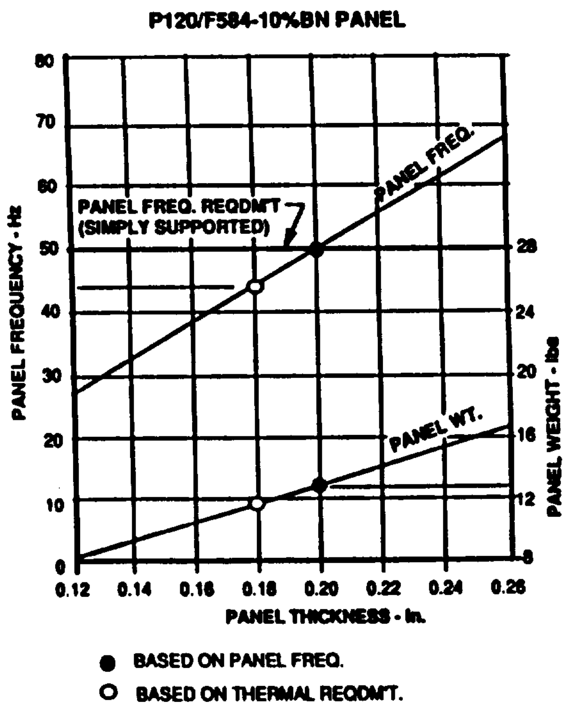


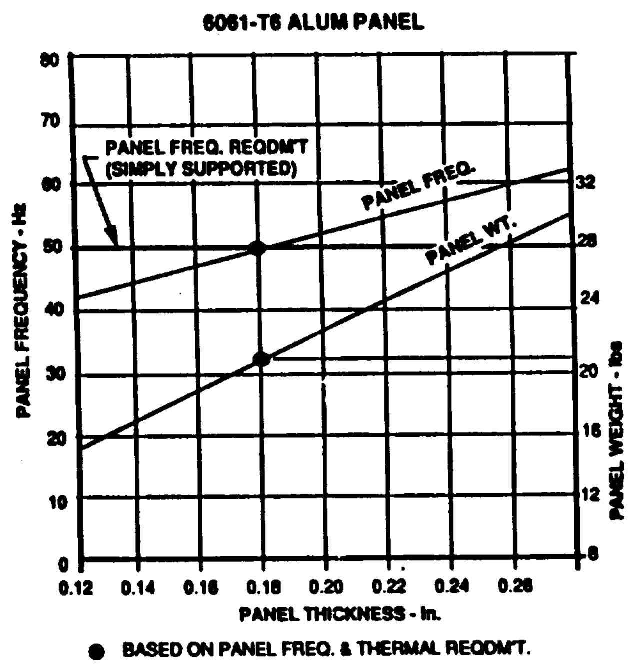
The ideal thermal/structural radiator material has high conductivity, high modulus, low CTE and low density. A comparison of these properties for selected materials is shown above. Except for pure radiator applications, composite equipment panels will almost always have a quasi-isotropic or near isotropic lay-up. Considering these lay-ups, the K1100/epoxy has the highest specific thermal conductivity. The use of carbon (C) and aluminum (Al) matrices can improve panel conductivities, particularly in the z (out of plane) direction, but these materials are typically limited by much higher processing costs, and by lay-up and size limitations. The high temperature processing of these materials can also result in reduced mechanical strength properties.

Composite Radiator Design - Conductivity



A typical thermal modelling curve for materials with different Kz (out-of-plane thermal conductivities) properties is illustrated above. The total thermal conductance of the radiator panel can be increased by increasing the panel thickness for a given material but to conserve weight this thickness should be kept to a minimum. Thicknesses(weight) can be minimized by using higher conductivity fibers. The curves (constant conductivity) show that once a Kz value of ≥ 1.2 BTU/ft-hr-°F is achieved, additional Kz enhancement does not significantly reduce panel thickness.





These figures show how the panel frequency requirement contributes to the panel thickness and weight for the specific cases of a P120/epoxy panel and a comparison 6061-T6 aluminum panel.

Radiator Panel Design Trade

PANEL MATERIAL	SIZE	STIFFNESS (MSI)	DENSITY (LB/IN.3)	THICKNESS (INCH)*	TOTAL WEIGHT(LB)		Kxy (BTU/ft-hr-°F)	Kz (BTU/ft-hr-°F)	COST PER PANEL($)		
					w/o heat pipes	with heat pipes			Material	Labor ††	Total
P120/Epoxy	24in x 42in	25	0.064	0.2	13.5†	12.5†	100	1.0-1.2**	22,500	6,000	28,500
K1100/Epoxy	24in x 42in	29	0.064	0.13	9.0†	8.4†	150	1.2-1.5**	37,500	6,000	43,500
6061 Aluminum	24in x 42in	9.9	0.096	0.18	17.4	13.6	120	120	200	5,000	5,200

*meets minimum stiffness(50Hz) and thermal conductance requirements
**matrix enhanced with conductive fillers
†includes 0.6 lb EMI shielding material(Cu mesh)
††includes fabrication and installation of required secondary stiffeners

The weight, thermal performance, and recurring cost of two candidate composite radiator panels with quasi-isotropic layups are compared with a baseline 6061 aluminum panel. For the comparison, the three panels have the same length and width dimensions but thickness is based on what is required to meet minimum stiffness (50Hz) and in-plane conductivity (100 BTU/ft-hr-°F) using each different material. The results show that the composite panels provide significant weight savings but at increased cost. The system-level evaluation of the weight/cost trade for the reference SOP small satellite is discussed at the end of this section.

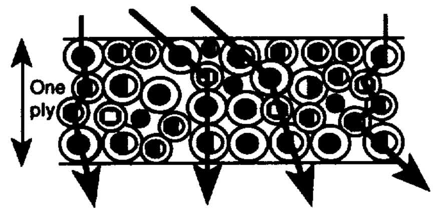
Recommendations for Development

- Develop the use of hyperconductive fibers such as K1100 and vapor grown fibers (benzene and natural gas derived) and improve data base
- Investigate the thermal advantages of intercalated fibers— A reduction in fiber E resistivity corresponds to increases in thermal conductivity
- Solve problems associated with combining heat pipes with low CTE composites— Develop composite or metal-clad composite heat pipes
- Improve Z- direction thermal conductivity with advanced matrix fillers such as helical graphite whiskers
- Improve Z- direction thermal conductivity with advanced layup methods such as braiding , Z-axis fiber piercing, and fiber precursor weaving
- Improve Z- direction thermal conductivity with advanced fiber coatings such as diamond and metal alloys
- Scale-up and cost reduction of existing processes

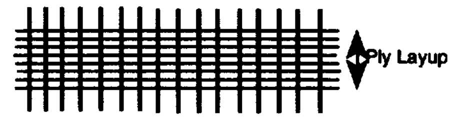
Even with the introduction of hyper-conductive fibers into composite designs, it is not expected that all thermal management requirements can be met without the incorporation of some heat pipes into the design. This will require a CTE "match" between the panel and heat pipe (usually aluminum) which is not easy to achieve for composites whose fibers have very *negative* CTEs. Technology developments to solve this problem would be very beneficial to the small satellite industry. A complementary effort would be to develop low CTE heat pipes using low CTE metals or metals combined with composites.

Improving the Z-axis thermal conductivity of organic matrix composite radiators is also very challenging. Existing concepts either result in only slight improvements in Kz or they result in a reduction of other properties such as shear strength. New, innovative concepts need to be developed.

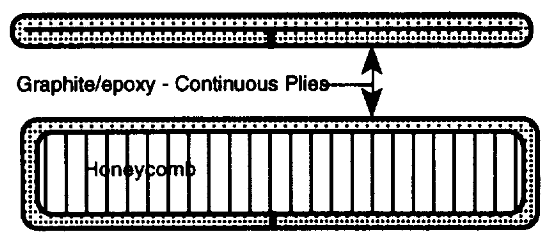
Coated Fibers



Z-Axis Piercing With High K Fibers



Wrap Around Layup



Embedded Composite Heat Pipes

The advanced concepts depicted above for the construction of radiator-equipment panels are based on the desire to enhance the thermal transfer efficiency of a standard panel which already uses the highest conductivity materials that are commercially available. The concepts assume an organic matrix composite in which the curing process yields the final product (i.e. no secondary operations such as heat pipe attachment). With the coated fiber concept, the idea is to provide a conductive coating on the fiber surfaces so that touching fibers will give added thermal paths through the thickness. In the same way, piercing the prepreg layup with high K fibers will potentially enhance Kz. In the wrap-around concept, the high K fibers are continuous from the box side of the panel to the opposite radiator side. Finally, heat pipes could be embedded in the composite laminate itself if the heat pipe CTE is tuned to that of the laminate. This would provide intimate panel to heat pipe thermal transfer. CTE tuned heat pipes could be created using an aluminum-cladding process.

System-Level Benefits
Passive Composite Radiator Panels

Analysis Assumptions

- 20 W/ft^2 Capability
- Cu Mesh EMI Shielding
- Battery & Electronics Radiators
- (a) 2 Panels, 14 ft^2
- (b) 5 Panels, 35 ft^2
- No Heat Pipes in Panels

Other Benefits

- Simplified Thermal Design
- Less Heat Pipes
- Facilitates Gr/Ep SmallSat Design
- I & T Savings

Relative System Payoff

TECHNOLOGY		SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY: Weight Savings (lb)	Recurring Cost Savings ($K)	RELATIVE PAYOFF PER SPACECRAFT: LEO Comm Spacecraft ($K)	GEO Comm Spacecraft (Equivalent) ($K)	LEO Rem Sens Spacecraft ($K)
P120 vs.	(a)	9	-46.6	25	169	25
Aluminum	(b)	22.5	-116.5	64	424	64
K1100 vs.	(a)	16.8	-76.6	58	327	58
Aluminum	(b)	42	-191.5	145	817	145

The focus components in this trade are the radiator panels on the reference SOP spacecraft. Two types of composite radiator panels are compared with the SOP Aluminum design. For each composite panel type, two cases are considered. In case (a) the battery electronics radiators are replaced with composite designs. In case (b) three additional thermal panels on the spacecraft are replaced with composite designs.

The results indicate a substantial savings in being able to substitute composites for Aluminum. The largest relative savings on a per spacecraft basis is in GEO, due to the higher launch cost of placing weight in GEO orbit. Other benefits which were not costed include simplified box design and integration using a more traditional design approach, i.e. mostly conduction through the box baseplate to the radiator panel instead of mostly radiation to the surrounding surfaces. The former allows a smaller box size for a given dissipated thermal load.

Calculating the net system (constellation) savings associated with the K1100 design for each of the three focus mission/orbit cases, the results are $2.9M - $7.3M for fifty LEO communications spacecraft, $3.3M to $8.2M for ten GEO communications spacecraft, and $0.1 to $0.3M for two LEO remote sensing spacecraft.

3.1.3 Triaxially Woven Fabrics This trade addresses the need for lower cost, lighter weight, stiffness-critical panels that could be used for rigid solar arrays, phased array antennas, and bus structures.

For thin ply faceskins used in solar arrays and other lightweight sandwich panels, the triaxially woven fabrics offer great potential benefits. Because of the 0+/- 60 planar triax weave, a single 5 mil ply laminate is balanced, resulting in both a weight and a cost savings relative to thin-ply laminates, which are labor intensive.

Products are now available constructed from ultra-high modulus PAN and pitch fibers. The very high modulus pitch fibers must be woven as a precursor and subsequently carbonized. While triax fabrics are more expensive than biax fabrics, they are typically much less expensive than thin prepreg tapes. For thicker laminates (>.020 inches) it would not be cost effective to use triax fabrics. The primary benefits will come from applications where the existing faceskin, whether thin ply composite or aluminum, can be replaced with a single ply of triax fabric.

Significant weight savings can be achieved in lightweight panels by using triaxially-woven fabrics to replace either aluminum faceskins or composite faceskins constructed from thin (1 mil/ply) prepregs. Panel designs using three different facesheet materials are compared in the figure. The first is an aluminum faceskin/aluminum honeycomb (1.6 pcf) construction in which the faceskins are bonded to the core using .030 psf film adhesive and the secondarily bonded solar cell dielectric is a 5 mil film adhesive. The second and third are composite faceskins applied to the same 1.6 pcf core using the same adhesive. One mil Kapton® is co-cured with the composite panels to meet dielectric requirements and save weight while avoiding an additional process step. For the thin ply faceskin panel, 8 plies of 1 mil M60J/epoxy prepreg tapes are laid up in a balanced (0,45, -45, 90)s quasi-isotropic sequence. For the triax faceskin, a single, balanced prepreg ply of triaxially woven Hercules UHM fiber is utilized. For comparison, all the panels had to meet minimum in-plane panel stiffness requirements. Curves comparing the weight savings of the composite panels relative to the aluminum panel are shown for both 0.75-inch and 1.50- inch honeycomb core thicknesses.

The results show that the triax panel is consistently 5 to 10% lighter than the thin-ply panel. Depending on the thickness of the aluminum facesheets, the composite panels are 10 to 70% lighter than aluminum.

1. PROVIDES UP TO 70% WEIGHT SAVINGS COMPARED TO ALL-ALUMINUM HONEYCOMB PANEL

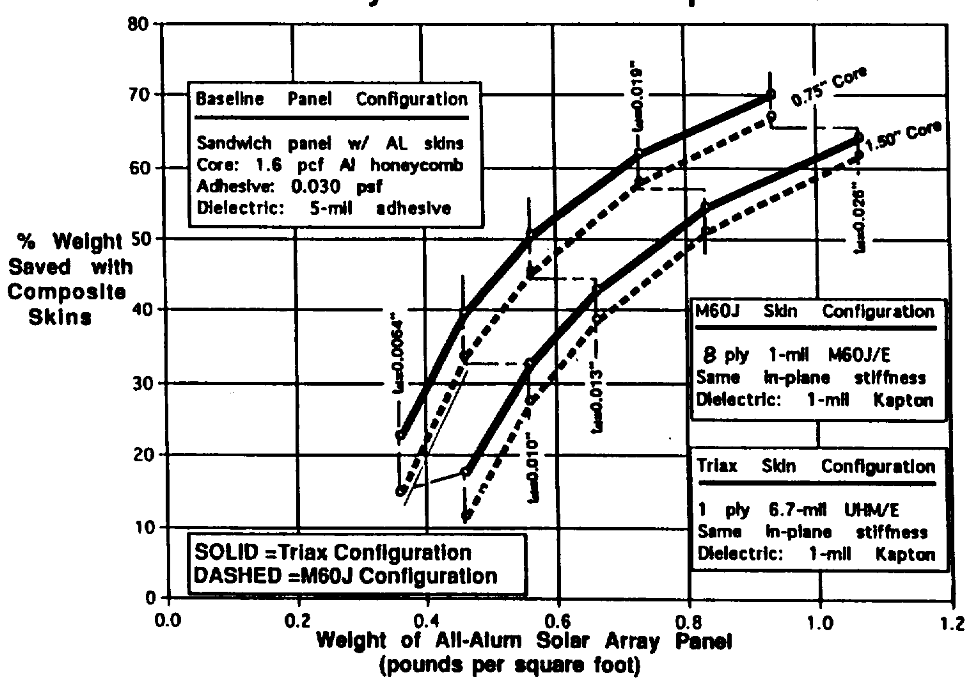
2. REDUCES LABOR COSTS BY SUBSTITUTING 1 PLY TRIAX FOR UP TO 8 PLIES OF UNIDIRECTIONAL TAPE (BALANCED QI LAYUP)AND UP TO 4 PLIES OF BI-DIRECTIONAL FABRIC (BALANCED QI LAYUP)

3. ALLOWS THE USE OF CO-BONDING SURFACE FILMS TO MEET THERMAL CONTROL AND DIELECTRIC REQUIREMENTS; THIS ELIMINATES THE NEED FOR SECONDARY BONDING OF FILMS WHICH ADD SIGNIFICANT WEIGHT

4. EACH PLY IS BALANCED AND ISOMETRIC — THIS SIMPLIFIES LAYUP DESIGN AND ELIMINATES THE NEED FOR VERY EXPENSIVE THIN PLY PREPREGS REQUIRED FOR THIN FACESKINS. IT ALSO MEANS LESS RESIDUAL OUT-OF-PLANE TENSILE STRESS IN THE BONDED FACESKIN.

5. A SINGLE TRIAX PLY WHEN USED IN COMBINATION WITH HERCULES UHM FIBER OR CERTAIN PITCH-PRECURSOR FIBERS CAN YIELD A NEAR ZERO CTE LAMINATE. THIS WOULD BE VERY VALUABLE FOR DIMENSIONAL STABILITY BENEFITS.

Weight Savings for Stiffness-Critical Solar Array Panels with Composite Skins



Recommendations for Development

1. DEVELOP TECHNOLOGY FOR TRIAXIAL WEAVING OF HIGHER MODULUS GRAPHITE FIBERS INCLUDING P75HT, M55J, AND M60J
2. GENERATE AND EXPAND DATABASE TO ALLOW INCREASE IN DESIGN ALLOWABLES
3. EVALUATE AND QUALIFY FULL-SCALE PANELS

Because triaxially woven fabrics are relatively new to industry and because the higher modulus versions have only been introduced very recently and are still considered developmental, considerable testing and evaluation needs to be done to qualify these materials. Besides their basic mechanical properties, properties such as CTE and CME need to be developed for dimensional stability applications. Their behavior with regard to thermal cycling and microcracking is also unknown.

System-Level Benefits Triaxially Woven Fabrics for Panel Facesheets

Analysis Assumptions

- 83 ft^2 of Rigid GaAs Solar Array Panels per Satellite
- 104 ft^2 of Antenna Panels per Comm Satellite
- Recurring Costs Include Labor but not Material

Other Benefits

- Lower Moment of Inertia (10 - 15%)
- Reduced Dynamic Interaction
- Increased Agility
- Simplified Integration

Relative System Payoff

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY		RELATIVE PAYOFF PER SPACECRAFT		
	Weight Savings (lb)	Recurring Cost Savings ($K)	LEO Comm Spacecraft ($K)	GEO Comm Spacecraft (Equivalent) ($K)	LEO Rem Sens Spacecraft ($K)
	Comm/R.S.				
Triax vs.	34	-3	269	813	-
Al (10 mil)	15.2	-1.6	-	-	120
Triax vs.	15.5	98	222	470	-
T300 (SOP)	6.9	39.2	-	-	94
Triax vs.	5.2	78	120	203	-
M60J	2.3	31.2	-	-	50

The focus components in this trade are the 83 sq. feet of solar array panels on the reference SOP small spacecraft. An additional 104 sq. feet of phased array antenna panels were also assumed for the communications focus mission. Recurring labor costs were included in the trade, though material costs were excluded because they are not available for the UHM triax material.

The results indicate a substantial cost and weight savings in being able to substitute triax fabrics for the reference SOP composite T300 panels, particularly for the commercial communications missions. The reduced solar array weight corresponds to a lower spacecraft inertia, which reduces both the attitude control torque requirements and the flexible dynamic disturbances.

Calculating the net system (constellation) savings associated with the UHM Triax versus T300 design for each of the three focus mission/orbit cases, the results are $11.1M for fifty LEO communications satellites, $4.7 for ten GEO communications satellites, and $0.2M for two LEO remote sensing satellites.

3.1.4 Non-Pyro Booster Release Mechanisms

- Technology Need
  - Lower Pyroshock for Small Satellites (Boxes Closer to Source)
  - NEA for Commercial Multiple Small Spacecraft Launch on a Single Booster

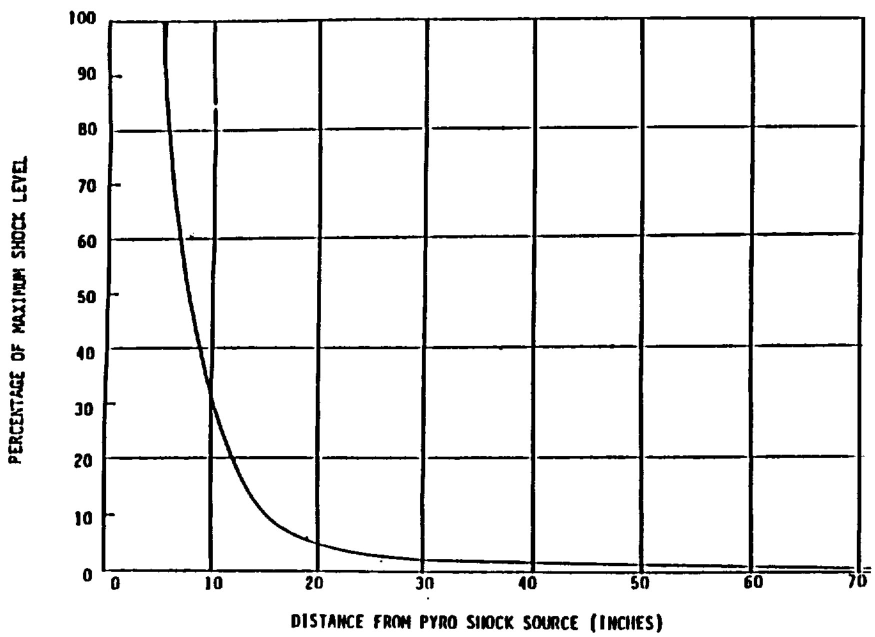
- Requirements
  - High Pre-load Release (5,000 lbs)
  - Simultaneity (milli-seconds)

3.1.4 Non-Pyro Booster Release Mechanisms This trade addresses the need for lower-shock release mechanisms, particularly for booster separation. The reason for the interest in non-pyro shock devices or non-explosive actuators (NEA) is that there are no standard analytical methods which can validate a shock-resistant design. Normally, the structure is designed according to standard practice and then later it is tested with the hope that it will pass the shock test. There are some rules of thumb but no design tools. The shock test can uncover weaknesses at surprising locations for components such as thermo-couples, solar array cells, and black box electronic parts. In many cases, these weaknesses may not be discovered until well into the satellite development program.

Pyroshock is composed of high frequency components which are damped out rapidly with distance. The shock spectrum provided in the component specification sets the testing level for verification purposes. Normally, the pyro devices are tested with qual or proto-flight hardware in place on the bus exposing the components to the actual shocks. However, if a box is being tested as a component then the level may be reduced based on the distance from the shock source using empirical curves, such as the example on the facing page. This is a particularly acute problem for small satellites, where all of the equipment is much closer to shock sources.

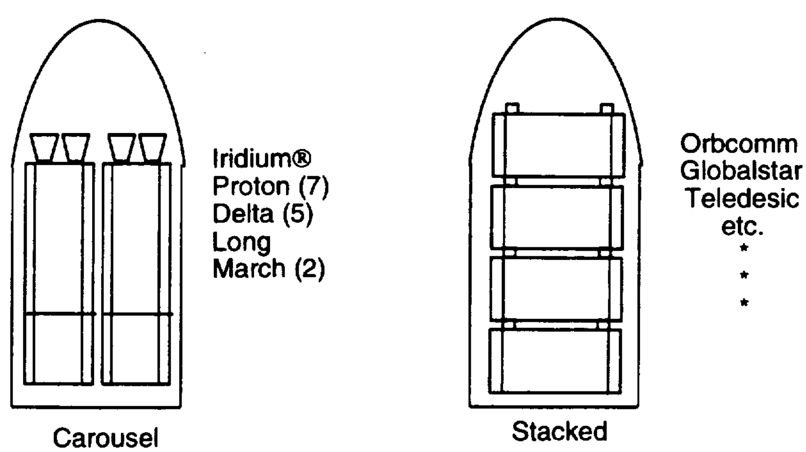
A number of new non-explosive actuators have flown or been qualified for separation of preloaded solar arrays, antennas, and payload covers. Many of these devices are currently baselined for use in remote sensing and commercial communications spacecraft. However, the benefits of non-explosive devices have not been realized for separation of the spacecraft/booster interface, an application where there are additional requirements for strength, high preload, and simultaneous firing of three or more devices within a few milliseconds. The following chart illustrates that LEO commercial communications satellites, which are launched in groups, could benefit greatly from this technology.

Pyroshock Attenuation vs. Distance from Source



Multiple Launch of Commercial Spacecraft

- Most Planned Commercial Communications Spacecraft Launched in Groups of 2 to 8 on Medium to Large ELV's
  - Large LEO Constellations
  - High Preload Required for Launch Loads
  - Simultaneity Required to Reduce Tip-Off Rates
  - Use of Marman Bands Impractical or Too Heavy



Example Devices and Potential Benefits

- Example Devices
  - Frangibolt - TiNi Alloy Company, San Leandro, CA
  - Fusible Link - BAC, Seattle, WA
  - Non-Explosive Separation Nut, G&H Technology, San Leandro, CA
- Potential Benefits
  - Reduced Pyroshock Environment
  - Reduced Safety Efforts
  - Insensitivity to EMI (Elimination of Shielding)
  - Reduced Weight (>10 lbs savings)
  - Elimination of Most Component Pyroshock Testing
  - Reuseable Mechanisms for Reliability and Performance Testing
  - Cost Savings (Typically $100K/Recurring and $450K/Non-recurring up to Several Million NRE)

The Advanced Release Technologies System (ARTS) program conducted by the Naval Research Lab is an excellent source of information on non-pyro release mechanisms. According to the ARTS report, "the non-explosive release mechanisms have a pyroshock output of about one fourth of today's pyromechanical release devices. This characteristic allows spacecraft designers to seriously look at eliminating much pyroshock testing since the levels for almost all of its components will follow this 75% reduction."

Even if pyroshock testing is still conducted at the spacecraft level, elimination of component-level tests can result in significant cost and schedule savings. Other benefits are cost savings due to the elimination of safety paperwork, reduced weight associated with eliminating the EMI shielding needed for pyrotechnic devices, and the ability to reset the devices for repeated testing.

A NASA Tiger Team led by NASA/LaRC is also currently studying NEA technology for spacecraft. For the booster interface application, further work needs to be conducted in quantifying the simultaneity requirements, designing devices for the associated higher preloads, space-qualifying the devices, and demonstrating the overall separation system performance.

System-Level Benefits
Non-Pyro Booster Release Mechanisms

Analysis Assumptions

- Weight Savings Mostly Due to Eliminating EMI Shielding
- Eliminate Qual Test for 15 Components ($30K Each NRE)
- Eliminate Acceptance Tests & Safety Paperwork ($100K RE)
- NRE Amortized Over # of S/C

Other Benefits

- Lower Specification, Design, and Analysis Costs
- Reliability
- Compact Satellite Design
- I & T Savings (Safety)
- Backup for Pyros if Components Fail Shock Test

Relative System Payoff

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY		RELATIVE PAYOFF PER SPACECRAFT		
	Weight Savings (lb)	RE &NRE Cost Savings ($K)	LEO Comm Spacecraft (~50) ($K)	GEO Comm Spacecraft (Equiv) (10) ($K)	LEO Rem Sens Spacecraft (~2) ($K)
Shockless Release vs. Pyros	10	109	189	-	-
	10	145		385	-
	10	325	-	-	405

The focus components in this trade are the three separation nuts at the booster interface for the reference SOP small spacecraft bus. The analysis was based on the assumptions shown, where the state of the practice is pyrotechnic release devices. To be conservative, only the component-level pyroshock testing was eliminated. The weight savings is based mostly on shielding for the three booster separation devices.

The payoff is greatest for the the remote sensing spacecraft, where the NRE savings from eliminating the shock testing is amortized over only two spacecraft. Many of the "other benefits" listed on the chart, which were not costed, can be significant. The potential savings for using non-pyro booster release mechanisms can be in the millions of dollars, depending on the specific spacecraft design and mission.

Calculating the equivalent system (constellation) savings associated with the non-explosive technology for each of the three focus mission/ orbit cases, the results are $9.5M for fifty LEO communications satellites, $3.9 for ten GEO communications satellites, and $0.8M for two LEO remote sensing satellites.

3.1.5 Lightweight Passive Thermal Straps

- REQUIREMENTS:

  1. High thermal conductivity
  2. Low weight, low volume
  3. Low outgassing
  4. Flexibility — Withstand shock, vibration, deployment loads; isolate dimensionally stable components
  5. Formability — Can be formed to make smooth connections between components
  6. Electrical grounding (optional)

- GOAL: ACHIEVE 10 to 20°C DROP PER STRAP FOR 15 WATTS DISSIPATED POWER

- ADVANTAGES:

  1. Create heat flow paths for the passive rejection of excess heat.
  2. Balance thermal gradients between satellite components and eliminate 'hot spots' that drive the thermal design of the entire component or system (e.g.. diode board on solar array: $\Delta 30°F$)
  3. Reduction of average component temperature results in:

     - reduced thermal cycling effects on materials
     - reduced requirement for high temperature materials
     - reduced outgassing

  4. Carbon fiber straps offer lower weight, occupy less volume than equivalent K metallic straps or heat pipes

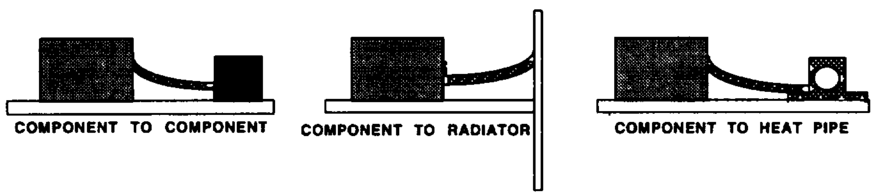
3.1.5 Lightweight Passive Thermal Straps This trade addresses the need for balanced thermal designs. Thermal straps have been used in satellite applications for many years. The Hubble Space Telescope utilized more than 40 thermal straps. Thermal straps provide a passive conduit for heat to be transferred from one location to another. They are particularly useful (1) in moving heat from a 'hot' device to a heat pipe or radiator (2) for balancing the temperature of adjacent components and (3) reducing local thermal gradients. They can be designed to fit specific applications by tailoring the size, shape, and thermal conductivity.

Conventional thermal straps have been constructed of braided copper and are very heavy. Aluminum straps have also been built by stacking sheets of aluminum foil and welding the ends. These straps are also heavy and bulky. With the availability of ultra-high thermal conductivity carbon fibers, the potential for integrating these fibers into flexible straps is very good, but additional development is required. High K fibers have been copper-coated or encapsulated in a flexible silicone polymer in early development studies. Because the high K fibers are fragile, encapsulation is necessary for durability and for survival in shock and vibration environments. Terminating the ends of thermal straps so that heat is efficiently transferred through the strap is a challenging materials and design problem. For an efficient connection, heat must move from the termination material directly to the fiber ends, picking up a high percentage of the encapsulated fibers.

Thermal Strap Trade Study

Type	Flexibility	Formability	Outgassing	Electrical Grounding	T. Cond.,K (W/m°K)	Relative Weight(2) (grams)	Relative Cost,$ (2)(3)
Copper Braid(1)	YES	YES	NONE	YES	390 [225.4*]	876 [1.93**]	150
Stacked Aluminum Foil	YES	YES	NONE	YES	210 [121.4*]	579 [1.28**]	500
K1100 Carbon Fiber/ (20% Copper)	YES	YES	NONE	YES	958 [553.8*]	142 [.31**]	800
K1100 Carbon Fiber/ (40% Silicone)	YES	YES	LOW	Development required	660 [381.5*]	105 [.23**]	600

(1)commercially available, baseline
(2)per equivalent length(30.5 cm) and equivalent thermal conductance
(3)estimated based on nominal material and manufacturing costs
*BTU/ft-hr-°F units
**pound units



A trade study was conducted using available information for copper, aluminum, and K1100 fiber straps. For equivalent thermal conductance (conductivity x thickness), the more highly conductive K1100 high K carbon fiber straps show significant advantages in weight savings. The higher cost paid for higher performance is related to the current high cost of the K1100 fiber(~ $2000/lb).

RECOMMENDATIONS FOR DEVELOPMENT

1. Bending evaluation
2. Vibration testing
3. Integral end fitting development/design and development of efficient thermal strap terminations
4. Develop metal or polymer coated high K carbon fibers that flexible, durable, and fatigue resistant
5. Optimize fiber fraction
6. Integration of straps into composite structure - co-cure processes
7. Proof of concept fabrication
8. System and sub-system qualification

System-Level Benefits
Lightweight Thermal Straps

Analysis Assumptions
- 10 Straps per Satellite
- 30 cm Length
- 958 W/m°K
- Cu Braid is SOP

Other Benefits
- Simplified Thermal Design
- Fewer Heat Pipes & Heaters
- I & T Savings (Qualification)
- Power & Volume Savings

Relative System Payoff

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY		RELATIVE PAYOFF PER SPACECRAFT		
	Weight Savings (lb)	Recurring Cost Savings ($K)	LEO Comm Spacecraft ($K)	GEO Comm Spacecraft (Equivalent) ($K)	LEO Rem Sens Spacecraft ($K)
Al Foil vs. Cu Braid	7	-3.5	53	165	53
K1100/Cu vs. Cu Braid	17	-6.5	130	402	130
K1100/Si vs. Cu Braid	17.7	-4.5	137	420	137

The focus components in this trade are ten, 30-cm thermal straps assumed for a baseline small satellite. The trade compares the Al foil, K1100/Cu and K1100/Si technologies versus the SOP Cu braid, using the data generated in the trade described on the previous chart. The results show that the higher cost of the K1100 material is offset by the relative cost value of the weight savings. The payoff is greatest for the GEO satellites, where the launch cost per pound of payload is highest.

One key potential benefit which was not costed is simplified thermal design due to the reduction of "hot case" and "cold case" temperature limits. Equipment which is designed to efficiently radiate large heat loads can sometimes require heaters during eclipse, increasing the power draw on the battery. In addition, further cost savings may be realized by not having to test off-the-shelf boxes to qualify them over a larger temperature range.

The equivalent system payoffs associated with the K1100/Cu thermal strap technology for each of the focus mission/orbit cases are: $6.5M for fifty LEO communications spacecraft, $4M for ten GEO communications spacecraft, and $0.3M for two LEO remote sensing satellites.

3.1.6 Integral EMI/EMC Shielding for Composites

CONCEPT: • CO-CURE A CONDUCTIVE MATERIAL LAYER WITH THE COMPOSITE PREPREG OR USE ELECTRICALLY ENHANCED CARBON FIBERS TO INCREASE EMI/EMC ATTENUATION

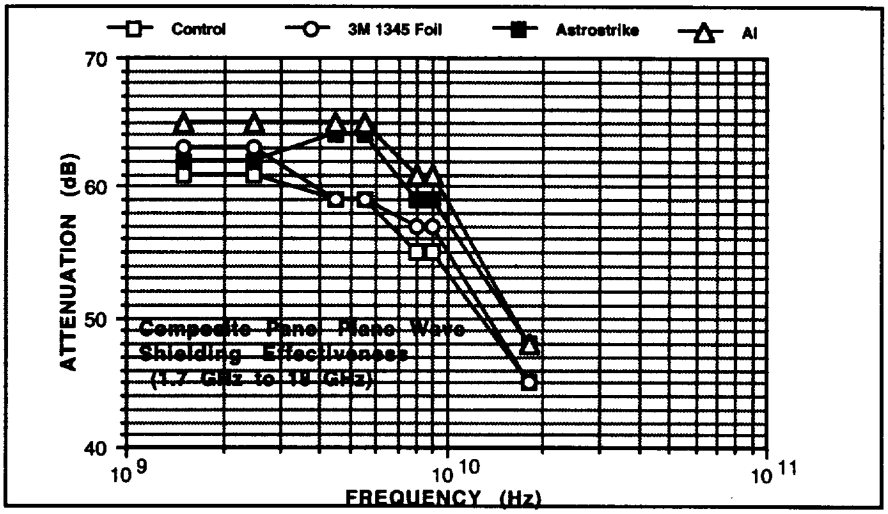
GOAL: • MEET EMI SHIELDING AND/OR GROUND PLANE REQUIREMENTS BY DESIGNING AND FABRICATING COMPOSITE PANELS WITH INTEGRATED EMC PROPERTIES, ELIMINATING THE NEED FOR SECONDARY PROCESSING SUCH AS PLATING OR SECONDARY BONDING OF FOILS

ADVANTAGES:

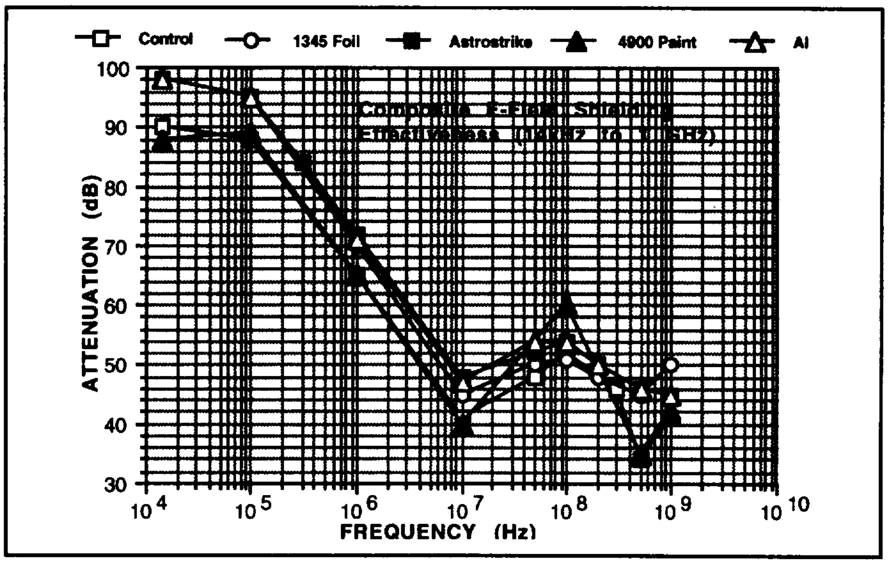
- "ONE STEP" PROCESSING REDUCES MANUFACTURING COSTS
- BY PROTECTING ELECTRICAL BOXES AT THE STRUCTURAL LEVEL, ADDITIONAL WEIGHT SAVINGS CAN BE OBTAINED BY THE ELIMINATION OF EITHER BOX COVERS OR ENTIRE ENCLOSURES
- EMBEDDED CONDUCTIVE MATERIALS ARE MORE DURABLE THAN SECONDARILY APPLIED CONDUCTIVE COATINGS, FOILS, OR TAPES
- INTEGRALLY PROCESSED EMI/EMC MATERIALS CAN POTENTIALLY BE DESIGNED AND POSITIONED TO MEET GROUND PLANE REQUIREMENTS

3.1.6 Integral EMI/EMC Shielding for Composites This trade addresses the need to reduce processing steps and manufacturing costs associated with adding EMI/EMC shielding to composites. As shown in the accompanying graphs, graphite/epoxy by itself has fairly good EMI attenuation characteristics for plane waves, E-fields, and H-fields, respectively. The other materials shown are the same "control" graphite/epoxy material modified with either 'co-cured' Astrostrike (expanded copper mesh), 3M 1345 Al foil tape, or a conductive paint, Chomerics 4900. Aluminum is also shown for reference. While shielding effectiveness equivalent to aluminum is not always required, designers are often reluctant to 'settle for less'. The results show that Astrostrike modified Gr/Epoxy is nearly equivalent for plane wave and E-field frequencies, and superior for H-field shielding.

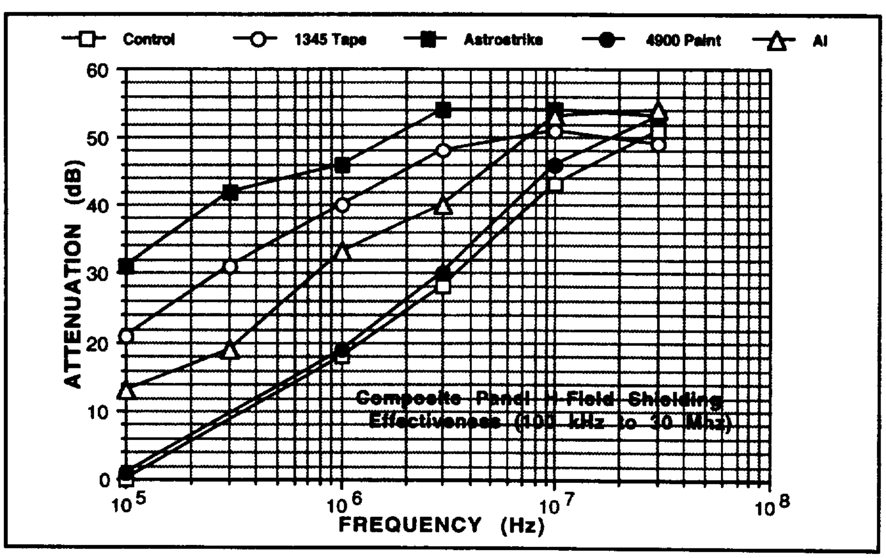
Enhancing Gr/Epoxy to meet shielding requirements is simplified when the modifying material, such as Astrostrike (or other conductive felts, fabrics or foils), is co-cured with the composite in one process step. Such materials then become somewhat embedded in the composite, but with proper masking, can leave one surface exposed for ground path connections.



Control
3M 1345 Foil
Astrostrike
Al
Composite Panel Plane Wave Shielding Effectiveness (1.7 GHz to 18 GHz)
ATTENUATION (dB)
FREQUENCY (Hz)



Control
1345 Foil
Astrostrike
4900 Paint
Al
Composite E-Field Shielding Effectiveness
ATTENUATION (dB)
FREQUENCY (Hz)



Control
1345 Tape
Astrostrike
4900 Paint
Al
Composite Panel H-Field Shielding Effectiveness (100 kHz to 30 Mhz)
ATTENUATION (dB)
FREQUENCY (Hz)

EMI/EMC Shielding Trade Study

EMI/EMC MATERIAL	WEIGHT ADDED (LB/FT2)	COST ($ per ft2)	AVERAGE ATTENUATION(dB)	COMMENTS
• Expanded Foil Mesh (copper or aluminum)	.022 to.085	LOW(3 to 4)	60-80	easy to apply
• Metal Coated Graphite Felts and Fabrics	0.002	LOW (.81)	50-75	easy to apply
• Metal Foils	.02 to.03	LOW(.05 to .50)	65-95	difficult to apply
• Metal Fabrics	.06 to .20	LOW(1 to 2)	40-60	heavier than mesh or felts
• Metal or Diamond Coated Fibers	N/A	MEDIUM-HIGH	N/A	diamond coatings are not mature
• Intercalated Fibers	N/A	NOT COMMERCIALIZED	N/A	can reduce elect. resist. by 5X

RECOMMENDATIONS FOR DEVELOPMENT

1. Develop low cost metal and diamond coated fibers — scale up existing processes
2. Demonstrate the use of integrated conductive EMI/EMC layers as an effective ground plane
3. Bring the development of fiber intercalation processes to maturity — scale up and commercialize

In the table above, candidate EMI/EMC enhancement materials are compared for weight, cost, and performance. While the use of metal mesh materials has been shown to solve shielding problems with little weight penalty, the use of these materials to meet total equipment panel ground plane requirements has not been demonstrated and needs to be developed and tested. Note that the diamond-coated and intercalated fiber approaches also offer through-the-thickness thermal conductivity (Kz) benefits as discussed in Section 3.1.2.

System-Level Benefits
Integral EMI Shielding for Composites

Analysis Assumptions

- SOP Is Copper Mesh
- Shielded Panel Areas
  - Comm 145 ft^2
  - Remote Sensing 100 ft^2

Other Benefits

- Enhanced Durability
- Ground Plane (Metal Foil Only)
- Enables Integrated Box Design

Relative System Payoff

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY: Weight Savings (lb)	Recurring Cost Savings ($K)	RELATIVE PAYOFF PER SPACECRAFT: LEO Comm Spacecraft ($K)	GEO Comm Spacecraft (Equivalent) ($K)	LEO Rem Sens Spacecraft ($K)
	Comm/R.S.				
Metal-Coated	7.5	3.9	64	185	-
Graphite Felt	5.2	2.7	-	-	44
Metal Foils	1.9	4.6	20	50	-
	2.8	3.2	-	-	25
Metal Fabrics	-11.0	2.9	-85	-262	-
	-7.6	2.0	-	-	-59

The focus components in this trade are the graphite/epoxy panels in the reference SOP small spacecraft bus and the attached representative commercial communications or remote sensing payload. The assumed panel areas for each focus mission are shown. The graphite/epoxy panels in the reference SOP bus include an embedded ply of copper mesh, so this was considered the baseline, although it is still somewhat advanced. The data shown on the previous chart were used in calculating the weight and material cost deltas. Processing costs were not included because they are not available.

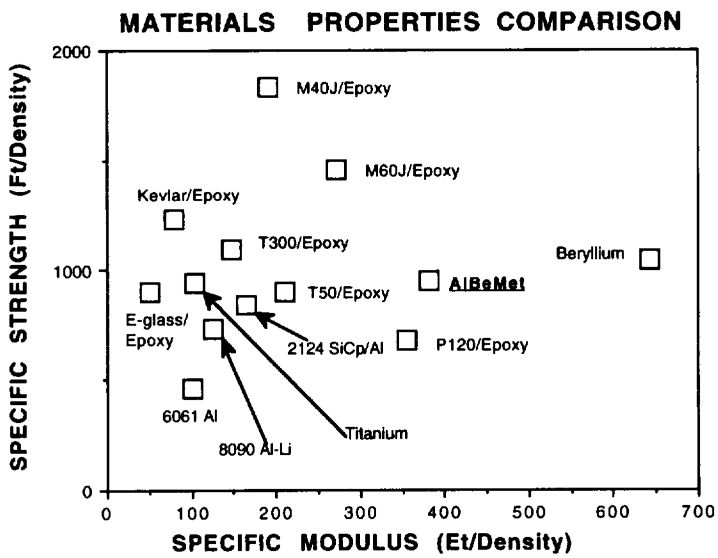
Given that both the SOP copper mesh and metal-coated graphite felts were rated as easiest to apply, they would be likely to have the lowest processing cost. The results show that the metal-coated graphite felts and fabrics offer the best combination of weight and material cost savings, as well.

Calculating the equivalent system payoff associated with the metal-coated graphite fabric technology for each of the focus mission/orbit cases, the results are $3.2M for fifty LEO communications spacecraft, $1.9M for ten GEO communications spacecraft, and $0.1M for two LEO remote sensing satellites.

3.1.7 Lightweight Isotropic Materials for Fittings

GOAL:
- Replace Existing Metal Components with New Lightweight Metal Alloys

SELECTED MATERIAL FOR EVALUATION:
•Beryllium(62%)/ Aluminum(38%) Alloy



3.1.7 Lightweight Isotropic Materials for Fittings This trade addresses the need for combined high-strength/high-stiffness materials for highly-loaded fittings and brackets, such as the booster interface fittings for spacecraft stacked in the launch vehicle. As shown above, structural composite materials such as M40J/Epoxy and M60J/Epoxy have properties superior to many structural metals. These properties, however, are 'in-plane' only for typical layups. For applications where high performance is required in all directions (i.e. orthotropic), 2D composites fall short, usually because of poor interlaminar shear or tensile strength in the 'out-of-plane' direction. Composite bus structures invariably require numerous fittings and attachments which must be bonded and/or bolted to the basic structure. Ideally, these fittings would also be composite, but, because of their weak out-of-plane properties, cannot be used. 3D composites have been developed for specialized applications but their high manufacturing costs limit their usage.

Metals such as titanium and aluminum are typically used for composite bus structural fittings; but for many designs, they are either marginal or do not meet stiffness requirements. Aluminum/Beryllium two-phase alloys show promise for meeting this need.

Metal Fitting Trade Study

Material	Weight/Fitting (lbs)	Weight/Vehicle (lbs)	Material Cost††/Fitting ($)	Machining Cost/Fitting ($)
• Beryllium/Aluminum	2.21	6.63	3937	2300
• Titanium Alloy*	4.77	14.31	487	2920
• Aluminum Alloy	2.94	8.82	30	1900

*baseline

Material	Density (lb/in3)	Thermal Conductivity (BTU/ft-hr-°F)	CTE (ppm/°F)	Tensile Modulus (Msi)	Tensile Strength (Ksi)
• Beryllium/Aluminum	0.076	121	7.7	28	60
• Titanium Alloy*	0.164	4.5	5	16**	170
• Aluminum Alloy	0.101	100	12.5	10.3†	76

*baseline
**marginally meets design requirement
†does not meet design requirement

RECOMMENDATIONS FOR DEVELOPMENT

- Improve Database for Existing BeAl Alloys
- Develop New BeAl Type Alloys (Magnesium and Other Additions) to Achieve Higher Fracture Toughness and Improved Tensile Strength
- Develop thin-gage (60-mil) as well as plate forms

A trade study was conducted for the redesign of the three metal booster interface fittings on the reference SOP spacecraft bus. The results indicate that the titanium alloys marginally meet the design requirements, while the aluminum alloys do not. Although beryllium/aluminum alloys are now significantly more expensive than aluminum or titanium alloys, they offer significant weight savings. The Be/Al alloys that are now commercially available are also limited somewhat by their fracture toughness properties. Development in addition to alloying is required to improve this performance.

System-Level Benefits
BeAl Technology

Analysis Assumptions

- Three to Four Fittings per Satellite
- Titanium is SOP for this Application - Marginally Meets Requirements

Other Benefits

- High-Strength Fittings for Payload Carriers or Satellite Stacking for Multiple Launch

Relative System Payoff

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY Weight Savings (lb)	Recurring Cost Savings ($K)	RELATIVE PAYOFF PER SPACECRAFT LEO Comm Spacecraft ($K)	GEO Comm Spacecraft (Equivalent) ($K)	LEO Rem Sens Spacecraft ($K)
BeAl vs. Aluminum	2.6	-15.1	5	46	5
BeAl vs. Titanium	9.0	-9.9	62	205	62

The focus components in this trade are the three or four metal fittings used to connect the spacecraft bus to the booster at the booster interface. The results indicate that the weight savings can outweigh the material and processing costs of BeAl in weight-critical applications.

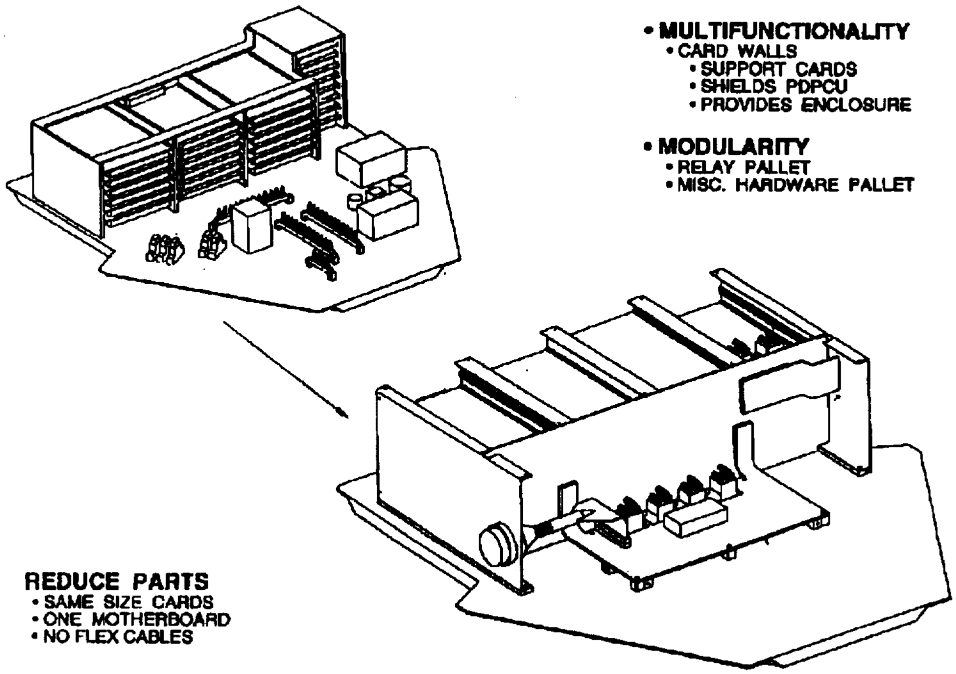
Calculating the equivalent system payoff associated with substituting BeAl for Titanium in the booster interface fittings for each of the focus mission/orbit cases, the results are $2.6M for fifty LEO communications spacecraft, $2.1M for ten GEO communications spacecraft, and $0.1M for two LEO remote sensing spacecraft.

3.1.8 Integrated Electronics

- A detailed trade study was conducted to explore an integrated electronic enclosure
- The evolution of the System Configuration Permits Integration of Electrical Power System Electronics Into a Single Box Including:
  - o PDU - Power Distribution Unit
  - o CCU - Charged Control Unit
  - o BDIU - Bus Data Interface Unit
  - o ADE - Actuator Drive Electronics
- A Concurrent Engineering Team was formed using an integrated team approach to
  - o Reduce Cost,
  - o Reduce Weight and
  - o Improve Reliability / Reduce Chance For Build Error

3.1.8 Integrated Electronics This trade addresses the need for integrated structural design approaches. The concept of combining several electronics boxes into a single box which is integral to the structure is particularly attractive for very small satellites.

This section reviews a trade study conducted by a multi-functional team at LMSC to examine the benefits of an integrated electronic box for small satellites. The primary focus was on the reduction of cost and weight by reducing part count and type, while increasing modularity and multi-functionality. The team also addressed several other topics including functionality, serviceability, maintainability, manufacturability, reliability, testing, and marketability.



Typical electronic functionality for the boxes to be integrated can be defined as follows:

PWA Cards: Holds components, holds connector, provides interconnect, provides rigidity and provides a thermal path.

Card Frames: Provides front cover, mounts PWAs, and holds front to chassis.

Enclosure: Allow growth, conducts heat, shields EMI / Radiation, holds wedgelocks, protects cards, and provides mounting.

Motherboard: Provides interconnect, holds XA connectors, and provides ruggedness.

Backplate: Covers back, provides shield and provides rigidity.

Of particular interest from a structural point of view is the enclosure and the printed circuit cards. These are candidates for composite materials. The flat panel in the figure is a structural panel onto which the electronic components are mounted.

Advantages in Integration

- •Reduce Required REE (Responsible Equipment Engineer) Effort, Fewer Independent Items to Track
- •Reduction in Power Supplies Per Unit
- •Expand Control Functions of BDIU Microprocessor, Eliminate ADE / CCU State Machines
- •Reduce Interface Quantity and Signal Conditioning
- •Reduce Card Quantity and Card Type Quantity
- •Eliminate Individual Unit Top Assembly and Enclosure Drawings
- •Eliminate Several Box - to - Box Cables
- •Reduce Quantity of Box Level STE (Special Test Equipment) Hardware, Software, Overall Test Time

LEVEL OF INTEGRATION	WEIGHT	POWER	COST SAVINGS	ISSUES / REMARKS
BASELINE DISTRIBUTED	35.5 lbs	50 watts	0	Examined 4 boxes
FUNCTIONAL INTEGRATION	31.3 lbs	50 watts	25%	Single Alum Enclosure
INTEGRATED ENCLOSURE	28.3 lbs	50 watts	25%	5 Sided Alum Enclosure
COMPOSITE ENCLOSURE	24.1 lbs	50 watts	25%	5 Sided Enclosure Shielding and Radiation

The study results showed that functional integration of the four boxes into one produced a 25% cost savings. From a purely mechanical point of view, the elimination of the enclosure common walls and the reduction of cable routing produces the greatest benefits in terms of weight savings. The use of composite enclosures saves an additional 4.2 lbs.

During the trade study, it became clear that the term integrated electronics has a different meaning for the electronics engineer than it does for a structures engineer. While they maintained that they had integrated the box into the structure, it was definitely not to the extent that a structures engineer would consider possible.

Further work is needed to incorporate composites at both the card and enclosure level for integrated electronics boxes. Of particular interest is the use of composite cards to reduce weight and enhance thermal conductivity. Further development should be a cooperative effort between the electronics and structures groups to ensure that all of the issues are addressed, particularly the environmental aspects.

System-Level Benefits
Integrated Gr/Ep Electronics Boxes

Analysis Assumptions

- Combined 4 Boxes in Single Gr/Ep 5-Sided Enclosure
- 2 Integrated Boxes per Satellite
- \$150K NRE per Box Included for Added Integration Cost (Amortized Over # of S/C)
- Assumes Heat Transfer, EMI, Grounding, and Radiation Issues Resolved

Other Benefits

- Better Suited to Very Small Spacecraft or Very Large Production Quantities
- Enabling for Very Mass-Critical Applications
- Weight and Thermal Conductivity Improvement Associated with Composite Cards, Holders, and Thermal Planes

Relative System Payoff

TECHNOLOGY	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY		RELATIVE PAYOFF PER SPACECRAFT		
	Weight Savings (lb)	NRE Cost Savings ($K)	LEO Comm Spacecraft (~50) ($K)	GEO Comm Spacecraft (Equiv) (10) ($K)	LEO Rem Sens Spacecraft (~2) ($K)
Integral Gr/Ep Box vs Aluminum	8.4 8.4	-6 -300	61 -	162 -	- -233

The focus components in this trade are two integrated electronics boxes that perform the electrical power distribution and data processing functions on a reference SOP small satellite. Because the majority of the benefits from integration can be achieved without advanced structures technology (i.e., with aluminum enclosures), this analysis addresses only the benefits of using composites instead of aluminum for the enclosure. The boxes are assumed to share a single common face with the bus primary structure. Integration costs associated with coupling the box and structure design are estimated.

The results indicate that there is a payoff for satellites that are fabricated in larger quantities. For small quantities, the non-recurring integration costs exceed the potential benefits. Coupling of the box and structure design activities presents organizational, schedule, and testing challenges that could be reduced for very small satellites which are produced by small, cross-functional teams.

The equivalent system payoffs associated with substituting structurally integrated graphite/epoxy electronics boxes into the spacecraft bus for each of the focus mission/orbit cases are \$3.1M for fifty LEO communications spacecraft, \$1.6M for ten GEO communications spacecraft, and -\$0.5M for two LEO remote sensing spacecraft.

3.2 OTHER TECHNOLOGY DEVELOPMENT TRADES

This section addresses three additional trade studies which did not conform to the "reference SOP spacecraft design" trade study approach used in the previous section. They are disturbance isolation, inflatable structures, and launch load alleviation. These trades represent special cases where the reference SOP spacecraft design and performance requirements discussed in Section 2.3.1 are not applicable. In the near term, these technologies are unlikely to be incorporated in a standard spacecraft design, but are more likely to be used to solve specific problems for which there is no other solution or to meet highly specialized requirements. The mission needs, SOA technologies, and focus components for these technology trades are shown below. The parameters and specific assumptions for each trade are discussed within each section.

Mission Need	Enhancing SOA Technology	Focus Component(s)
Precision Pointing and Jitter Stability for High Bandwidth Optical Communications Crosslinks or High Resolution Imaging Sensors	Passive Disturbance Isolation	Momentum Wheel or Reaction Wheel Isolator
Lightweight Deployable Structures for Solar Arrays	Inflatable Structures	Inflatable Struts
Reduced Launch Load Environment for Multiple Launch of Multiple Satellites	Launch Load Alleviation	Spacecraft Booster Adapter

3.2.1 Disturbance Isolation This trade addresses the need for enhanced pointing accuracy and stability in the presence of harmonic disturbances. Normally disturbance isolation requirements are met through good systems engineering and spacecraft design practices. However, isolation technology may be used in specific cases where the excitation levels are high or the disturbance is located in close proximity to sensitive equipment. Thus, isolation technology provides an alternative approach for solving system design problems. Example applications are high-bandwidth optical crosslinks for commercial communications or enhanced vehicle stability for precision remote sensing and imaging systems.

Passive Disturbance Isolation

- Problem
  - Optical Device - about 10 lbs
  - Jitter Sensitivity - about 10 micro radians
  - Momentum Wheel Disturbance Most Significant - 100 Hz
  - Requirement: Reduce Jitter by A Factor of 10
- Approach: Isolation of 23 lb Momentum Wheel Assembly
- Isolator Trade Space
  - Non Linear Leaf Spring
  - Non Linear Nitinol Coil Spring
  - Spring / Viscous Damper
  - Elastomeric Isolators

It is difficult to define the benefits associated with a payload isolator because it is so case dependent. The results presented here are based on a study performed for another spacecraft program, wherein the "payload" is an optical crosslink.

The problem examined was to replace an existing RF inter-satellite crosslink with a laser crosslink. The transmitted beam is considerably more narrow and drove the jitter requirement. The first trade question was whether to isolate the cross link or the disturbance source. Because of the existing design constraints it was decided to trade different approaches for passively isolating the disturbance source.

The primary disturbance source was a momentum wheel which rotated between 95 and 110 Hz. Although some consideration was also given to balancing the wheel to closer tolerances, this approach was rejected because it could not be proven that it would meet the specification.

Isolator Requirements

- Packaging Envelope (Baseline/Max)
  - –15.937"/17"dia and
  - – 6.8"/5.5" height
- Weight Increase Limited to 5.0 lbs
- Survive Launch Environments
  - – Random Vibration - 12.1 Gs RMS/Lateral and 18 Gs RMS/Axial
  - – Sine Vibration - 9.8 Gs Lateral from 17-100 Hz
- Survive On-Orbit Thermal Environment
  - – Operational -20 to 60 degrees C
  - – Non Operational -40 to 65 degrees C

Once the decision was made to pursue an isolator, the momentum wheel supplier was asked to design an isolator which could meet the itemized requirements. The original weight goal was around 3 pounds and had to be raised to reflect the actual design.

During the initial mechanism design studies, the spring/damper approach was eliminated to avoid the complication of a launch lock which had to be released after the satellite was inserted into orbit. The viscous damper approach was eliminated because of the sensitivity to temperature extremes. Thus, the selected approach was a non-linear spring.

The principal recommendation for development is to demonstrate and qualify an appropriate isolator design. While the industry is making some progress with the design of non-linear isolators which can be packaged in a small volume, it has yet to prove that these designs actually perform and are flight qualifiable.

For active isolators, the industry has a few designs which hold promise in meeting micro-gravity isolation requirements. The major problem is packaging and the need to have launch locks.

Isolator Benefits

ITEM	BASELINE	ISOLATED	REMARKS
WEIGHT	24 lbs	39 lbs	Assumed 3 Reaction Wheels and each Isolator weighs 5 lbs.
POWER	15 watts	15 watts	Assumed Isolators are passive
COST			
o Non-Recurring	-	$700K - 1000K	This includes Engineering design and flight qualification
o Recurring	$500K - 1000K	$210K - 400K	The smaller amount is typical of Communication Satellites (large buy) and the upper amount is typical of Earth Sensing Satellites.

	SAVINGS PER SPACECRAFT WITH SOA TECHNOLOGY		RELATIVE PAYOFF PER SPACECRAFT		
TECHNOLOGY	Weight Savings (lb)	RE &NRE Cost Savings ($K)	LEO Comm Spacecraft (~50) ($K)	GEO Comm Spacecraft (Equiv) (10) ($K)	LEO Rem Sens Spacecraft (~2) ($K)
Passive Isolator	-15 -15	-1000 NRE 300 RE	160 -	-160 -	- -320

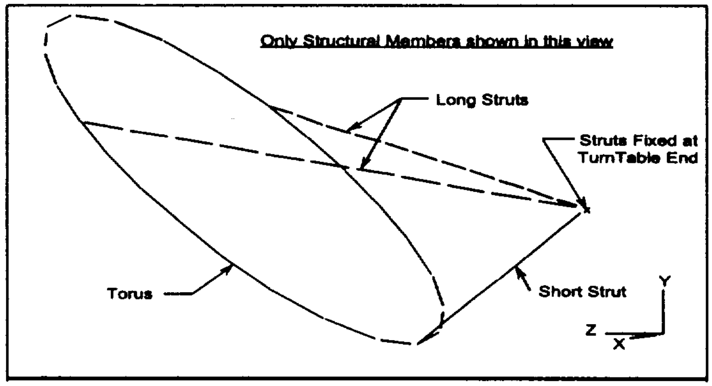
The focus components in this trade are the attitude control actuators. The baseline weight estimate is for a SOP design which replaces the momentum wheel with three standard reaction wheels without isolators. The SOA technology design uses the momentum wheel with a passive isolation system. The costs are projected from the detailed estimate provided by the wheel supplier. Unlike in the previous section, the NRE development cost is included here to reflect the specific nature of each isolation design problem.

The results indicate that for systems involving large numbers of spacecraft, such as Big LEO communications missions, the technology benefits can outweigh the implementation costs. For the other missions, the value of a pound of weight savings would have to be much greater than the launch cost metrics for the technology to show a payoff.

3.2.2 Inflatable Structures

APPLICATIONS

- Booms
- Trusses
- Sun Shades
- MLI Support Structures
- Solar Cell Support Structures
- Planar Array Antennas
- Solar Concentrators
- Reflector Antennas
- Instrument Support Structure



Solar Concentrators

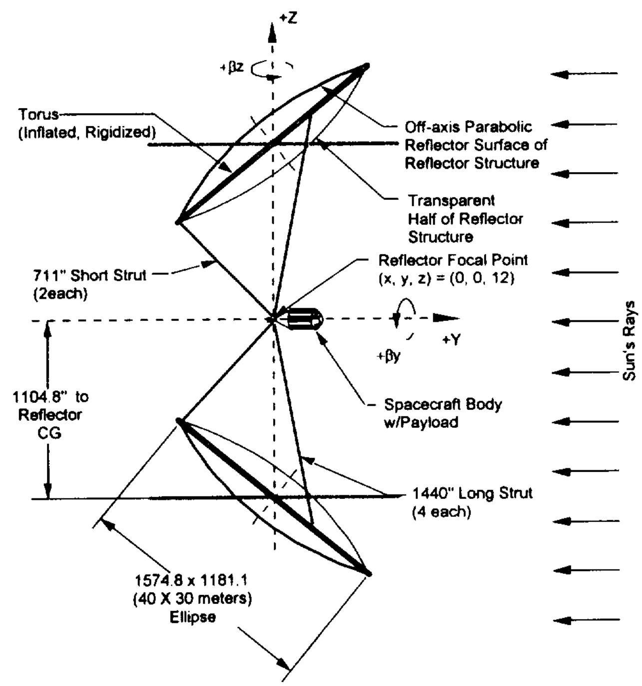
- 1993 Solar Astromast Torus Study
- Based on a Spacecraft used to transfer a 26,000 pound payload from LEO to GEO
- Inflatable Solar Reflector Concept for Satellite Power Structure
- Two Off-axis Parabolic Solar Reflectors (100 x 130 ft)

3.2.2 Inflatable Structures This trade addresses the need for lightweight deployable structures and mechanisms. The Teledesic Mega LEO communications satellite is one example of a planned system with very challenging deployable structures requirements that could potentially be addressed by inflatable structures.

Large structures are ideal candidates for inflatable structures. The larger the structure the greater the advantage. Typical applications fall into two categories: the first is large surfaces such as solar arrays, sun shades, solar concentrators, and reflectors. The second category is booms, trusses and instrument support structures. References 8 and 9 were used to estimate the benefits associated with the strut applications. This reference includes a detailed trade study previously conducted by LMSC for deployable booms, part of which is excerpted here.

APPLICATIONS CRITERIA

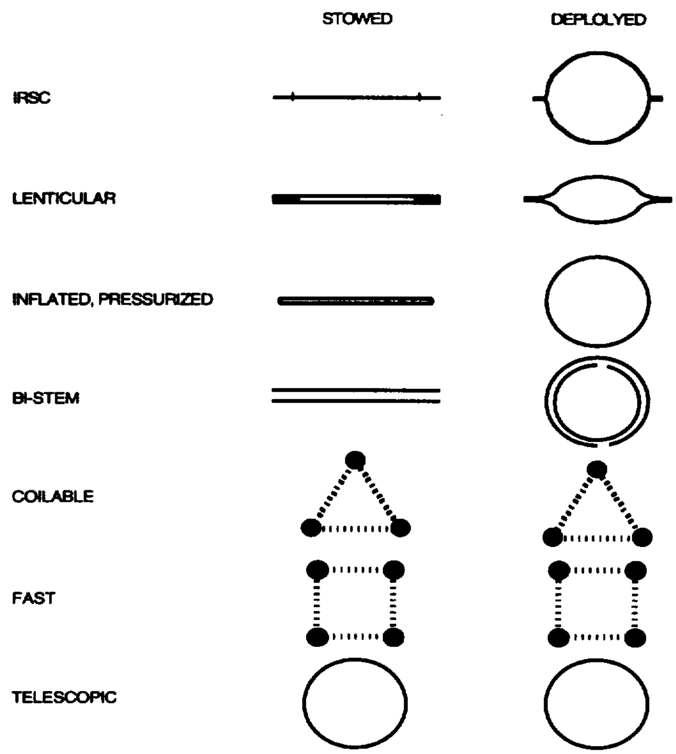
- Low Cost
- High Reliability
- Low Weight
- High Surface Precision
- High Mechanical Packaging Efficiency
- Dimensional Stability
- Good Dynamic Characteristics



The Solar Propulsion Concept (Reference 8) proposed by K. Laug of Phillips Laboratory, Edwards Air Force Base is based on a spacecraft powered by a solar powered propulsion system which will transfer a space vehicle with a 26,000-pound payload from LEO to GEO within 30 days. Two 100 x 130 foot off-axis parabolic solar reflectors are required to provide energy to the rocket motor. Each reflector consists of an accurately shaped, inflated structure. One side of the inflated structure is transparent while the inside of the other half is coated with a highly reflective film. The sun's rays pass through the transparent side, reflect off the inner surface of the other side, and are focused on the motor located on the centerline of the spacecraft.

The area of interest for this application is the struts which support the solar reflector. There are two different length struts. One is 1440 inches long and the other is 711 inches long. These are candidates for inflatable struts.

Inflatable Strut Trades



Seven strut concepts were evaluated in this study. This chart shows cross sections of each of the seven concepts as described below.

IRSC (Inflated, Rigidized, Space-Cured Composite) Tubular Struts. These are thin-wall tubes consisting of fiber reinforced composite materials. When filled with a gas, they are deployed and then cured.
Lenticular Struts The lenticular truss member concept consists of a thin walled shell with a lenticular cross section.
Inflated, Pressurized Tubular Struts
BI-STEM (Storable, Tubular, Extendible Member) Struts.
Coilable Struts The coilable strut concept consists of three continuous composite longerons joined by battens and diagonals.
FAST (Flexible Articulated Square Truss) Struts.
Telescopic Struts A number of concentric tubes are nested in the stowed position and sequentially extended to form the full length strut.

Strut Configuration			Total Weights of Sruts & Components per Spacecraft (lb.)										
Concept Type	Concept No.	Material	Struts	Strut 1-mil Internal Coatings	Unheated Thermal Blankets	Heated Thermal Blanket	Strut Fittings at Torii	Turntable Fittings	Strut Cannisters	Deploy Drives	Total Weight		
											Min.**	Nom.	Max.**
IRSC	IRSC-2	Composite	133	17	N/A	N/A	2	9	43	75	213	279	345
Lenticular	Lent-4	Composite	161	N/A	N/A	N/A	2	12	49	75	230	299	368
	Lent-6	301 CRES	668	N/A	83	N/A	3	42	91	75	848	962	1076
Pressurized	Press-1	301 CRES	514	N/A	63	N/A	3	32	60	75	656	747	839
	Press-4	Composite	111	17	N/A	N/A	3	32	60	75	211	298	385
Bi-STEM	STEM-1	Be-Cu	1306	N/A	62	N/A	4	73	89	75	1482	1609	1736
	STEM-2	301 CRES	698	N/A	N/A	116	4	39	51	75	887*	983*	1079*
	STEM-3	Composite	206	N/A	N/A	N/A	3	13	40	75	272	337	403
Coilable	Coil-1	Composite	86	N/A	N/A	N/A	5	11	134	75	199	311	424
FAST	FAST-1	Composite	108	N/A	N/A	N/A	5	12	135	75	222	335	449
Telescopic	Tele-1	Composite	140	N/A	N/A	N/A	2	9	39	75	203	265	328

* Also requires 1 kw power source for heater.

** Min. and Max. weights are estimated (based on deviations from Nominal).

N/A = Not Applicable

Comparisons of the seven strut concepts are included in this table as well as the combinations of strut type, thermal protection, and the materials considered in the trade study.

The outside diameter (OD) of the tubular strut concepts ranged from 6.9 to 8.3 inches. Wall thicknesses varied from 0.010 to 0.015 inch. The total weight of the six struts required for the spacecraft ranged from 86 pounds for the Coilable concept to over 1300 pounds for a metallic Bi-STEM concept.

The weight of the fittings at each end of the struts varies with the strut concept, material, and geometry. The strut-to-turntable fittings are much heavier than the strut-to-torus fittings due to their higher mechanical loads and the need to insure high rigidity.

The weight of a strut canister varies with the strut concept and the volume of the strut in the stowed position ranging from 6 to 25 cubic feet.

Inflatable Struts Summary

Strut Type	IRSC	Lenticular		Pressurized		Bi-STEM			Coilable	FAST	Tele.
Deploy Method	Pneum.	Mechanical		Pneumatic		Mechanical			Mech.	Mech.	Pneum.
Strut Composite Fiber or Metal	M40J	M40J	301 CRES	301 CRES	M60J	Be-Cu	301 CRES	M40J	IM7	IM7	M40J
Composite Mat'l Form (stowed)	Prepreg	Cured	N/A	N/A	Prepreg	N/A	N/A	Cured	Cured	Cured	Cured
Thermal Protection System	None	None	Blanket	Blanket	None	Blanket	Heat Bl.	None	None	None	None
Support Structure Minimum	213	249	848	656	211	1482	887**	284	199	222	203
Weight (lb.) Maximum	345	391	1076	839	385	1736	1079**	418	424	449	328
Deflection M.S. (Limit)	0	0	0	0	0.9	0	0	0	0	0	0
Strength M.S. (Ultimate)	1.1	3.8	4.4	1.1	0	6.7	6.7	10.6	0.6	2	1.1
Stowed Volume (cubic ft)	6	9	9	6	6	22	22	22	25	9	10
Material Shelf-Live (stowed)*	P	H	H	H	P	H	H	H	H	H	H
Ground Test Deploy Simulation*	P	G	G	G	P	G	G	G	H	H	G
Deployment Location Accuracy*	P	G	G	P	P	P	P	P	E	E	G
Mat'l Dim. Stability in Space*	G	G	H	H	G	H	H	G	G	G	G
Damage Tolerance (in space)*	G	G	G	P	P	H	H	H	H	H	G

M.S. = Margin of Safety

*Ranking Factors: P = Poor (may not meet requirements)
G = Good (probably meets requirements)
H = High (definitely meets requirements)
E = Excellent (Exceeds requirements)

** Concept also requires 1 kw power source for blanket heating system.

Weight is not the only evaluation criterion. An assessment of some of the most important factors is included in the table. Other factors not addressed in the table include manufacturing producibility, deployment reliability, development cost, flight article cost, and maintenance cost (if any).

Of the seven strut concepts included in the trade study, the Flexible Articulated Square Truss (FAST) concept was selected as the most promising. Carbon fiber composite material with a near-zero coefficient of thermal expansion (CTE) is selected to minimize deflections due to heating and cooling. The FAST is one of the lightest design concepts and was judged to be the most accurate during deployment.

As can be seen in the table the inflatable designs are about equal to the current technology and need a second generation approach just as black aluminum evolved into integrated co-cured structures. Thus, when all design details are considered, the inflatable alternatives do not deliver the projected benefits. Additional work is needed to advance the design and manufacturing techniques to realize the projected benefits.

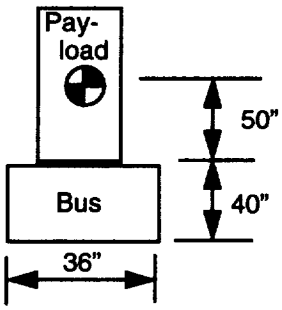
3.2.3 Launch Load Alleviation

- Reduced Loads for S/C Primary Structure Leading to One or More of the Following:
    - Reduced Weight Allowed by Lower Design Loads Provides Opportunity To Add Extra Fuel/Equipment
    - Reduced Testing Allowed by Higher Factors of Safety
    - Reduced Cost by Lessening Structural Sophistication
- Reduced Vehicle Loads at S/C Interface
- Potential for Reduced Shock into Launch Vehicle due to S/C Pyrotechnic Separation
- Insensitivity of S/C Loads To Launch System
    - Lessens Impact of Switching From One Vehicle to Another
    - Provides for Potential Streamlining of Dynamic Loads Prediction/ Verification
- Reduced Non-Acoustic Portion of Random Vibration Requirements

3.2.3 Launch Load Alleviation With the recent developments in smart structures coupled with the emphasis on small launch vehicles and spacecraft, there is renewed interest in the potential of designing a launch load alleviation system. Potential benefits include reduced static load factors and random vibration environments resulting in lighter primary / secondary structure and lighter black box designs. One side benefit is the ability to fly more ground instruments without making modifications for launch loads. Unfortunately at this time these studies cannot quantify the payoffs or suggest an isolator design which could be used for planning purposes.

The financing of large communication satellite constellations is requiring international consortiums which dictate the use of their boosters. This means that multiple satellites are being planned to be launched in groups on a number of different boosters. In addition, there is a desire to postpone for competitive reasons the decision of which booster to use in order to obtain the best launch price. A universal launch alleviation system that could be programmed to permit the use of any booster would be a competitive advantage for the industry.

Launch Load Alleviation Trade



PAYLOAD DESCRIPTION
- Rigid Structure
- Weight = 1000 - 2500 lbs

BUS DESCRIPTION
- Cylinder Radius = 18"
- Thickness = 0.12"
- Fixed Base Freq = 16 Hz
- Weight = 700 lbs

LOAD PARAMETERS

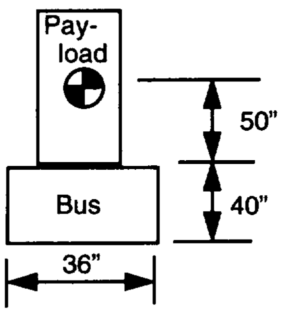
• Lateral Load Factor	2.5 g's
• Axial Load Factor	4.0 g's
• Factor of Safety	1.4
• Stress Concentration Factor	3.0
• Yield Allowable	35 ksi
• Stability Allowable	26.7 ksi

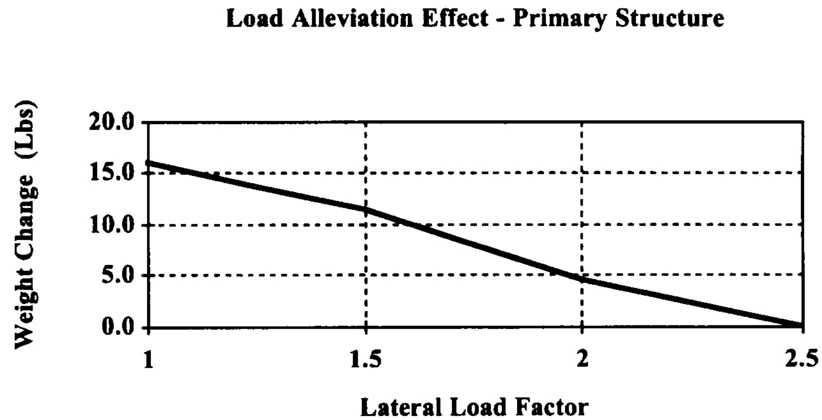
A simple trade was performed to evaluate the benefits of a load alleviation system for a satellite structure mounted atop the booster. The payload is assumed to be rigid and weigh 2500 lbs, and could be representative of a larger single satellite payload, or a stack of small communications satellites launched together.

For expediency, the bus structure is represented as a thin walled aluminium cylinder sized to have a fixed based frequency of 16 Hz which is the typical requirement. The total bus weight was set at 700 lbs to reflect the SOP reference bus design weight.

There have been a number of studies which have examined the benefits of using a load alleviation system starting in 1990 by the Aerospace Corp for the Phillips Lab. That study examined large satellites and was not conclusive in its recommendations. Recent SBIR contracts are studying the benefits for small satellites, but have not yet been completed.

Launch Load Alleviation Effect





The graph shows the weight savings versus lateral load factor, which is driven by the stability allowable. A stress concentration factor was included in the calculations in order to account for local design details.

These results show that if the lateral load could be reduced by a factor of two, then a weight savings of approximately 14 lbs could be achieved for the 700 lb bus. The structure weight is on the order of 20% of that value, or 140 lbs. Thus the weight saving is approximately 10% of the structure.

Drawbacks / Concerns / Limitations

- DRAWBACKS
  - Space / Weight / Cost Consequences of Isolation Hardware
  - Increase Rattlespace Requirements
  - Potential For Malfunction
  - Additional Preflight Activation / Checkout
- CONCERNS
  - Impact on S/C Separation Effectiveness
  - Consequence of Active Aspects on S/C Loads Analysis Methodology
  - Interaction with Launch / Space Vehicle Slosh Modes
  - Interaction with Launch Vehicle Control System Stability
- LIMITATIONS
  - Ineffective at High Frequencies Due To Dominance of Acoustic Excitation
  - Quasi-Steady Accelerations Remain

There are many concerns which need to be addressed before a load alleviation system could be used. These are some of the challenges which must be addressed during the technology development.

4.0 SUMMARY

Fueled by the growing number of commercial communications and remote sensing ventures in Low Earth Orbit and the current NASA trend toward lower-cost missions, the small spacecraft industry is in the process of explosive growth.

The objective of this study is to conduct trade studies from the perspective of small spacecraft designer/developer to determine and quantify the structures and structural materials technology development needs for future commercial and NASA small spacecraft. Technology development needs for about the next five years are emphasized, corresponding to technology insertion points for planned missions from now until 2001, and launch dates through 2005. While many of the initial constellations of LEO communications and remote sensing satellites will be launched by 1997 or 1998, there are many "block upgrades" and 2nd-generation replacement systems that will be required, particularly once the 5-year mean mission duration has been reached around 2002 or 2003.

The primary tasks accomplished as part of this study are shown below, each of which is briefly summarized in this section. The small satellite systems analysis activities described in the first three bullets are reviewed in Section 4.1. The results of the technology development trade studies are summarized in Section 4.2, which includes a ranking by relative payoff and priority. Finally, the conclusions and recommendations are presented in Section 4.3.

- Characterized Focus Commercial Communications and Remote Sensing Missions in Terms of Orbits, Sizes, Performance, and Design Drivers
- Identified Structures and Materials Technology Needs for Development in Next Five Years Based on Interviews with Personnel from Small Spacecraft Programs
- Developed Systems Analysis Approach, Including Definition of Example SOP Spacecraft Bus Design, Performance Requirements, and Metrics for Conducting Technology Development Trades
- Conducted Abbreviated Systems Analyses to Quantify Payoffs of Using Selected Advanced SOA SmallSat Technologies
- Component and System Level Benefits Identified, Quantified, and Ranked for Enhancing Technologies
- Other Technologies Identified Based on Future Needs, Key Trends, and Anticipated System Trades

4.1 SYSTEMS ANALYSIS SUMMARY

Throughout this study, public sources of information (e.g., press releases, conference papers, and especially FCC license applications) were used to develop an understanding of the markets, missions, orbits, physical characteristics, and future trends for the two focus missions. Typical characteristics of commercial communications and remote sensing spacecraft, summarized from Section 2.1, are shown. The assessment revealed that all of the *small* spacecraft planned for the two focus missions weigh 1,800 lb or less, corresponding to anywhere from 300 to over 1,000 small spacecraft planned for launch between now and 2005. Well over 90% of the planned missions are for commercial communications spacecraft, which are typically launched in groups on medium or large ELV's.

In addition to the literature survey, interviews were conducted with LMSC personnel involved in various aspects of small satellite development and fabrication in order to gain a broad perspective on structures and materials technology development needs. The table on the opposite page provides an overview of the key trends and development needs. A more thorough summary of recommendations for technology development is provided in Sections 2.2.2 and 2.2.3, while detailed recommendations are addressed in Section 3.

The interviews also included a discussion of the challenges involved in high-performance small satellite design and integration. Many of these are associated with commercial design-to-cost approaches where the structure is used as an integrating element to accommodate the space-qualified performance ranges of existing, off-the-shelf components (i.e. thermal, random vibration, shock, radiation).

Other challenges are simply a result of the reduced size of the spacecraft. For example, increased thermal density and reduced thermal capacitance make it more challenging to reject heat and minimize temperature swings, respectively. Shock and contamination levels are increased, as sensitive equipment is now closer to the offending sources. Pointing stability can be a challenge, as the relative inertia ratio between the bus and solar arrays is decreased. Smaller spacecraft provide less radiation shielding for electronic components. Finally, deployable structures and appendages compete for more relative space when stowed in the launch vehicle shroud, and for more of the relative thermal and optical fields of view on orbit.

While a different mix of challenges faces each small spacecraft design, each represents a potential area for SOA technologies to be considered in addition to the standard SOP design approaches.

Focus Mission Characteristics & Technology Needs

- LEO Commercial Small Satellite Weight Range 880 - 1,800 lb
  - NASA EOS Earth Probes, SMEX and SSTI Missions Smaller
  - MEO Communications Satellites 5,500 lb
  - GEO Communications Satellites 6,600 lb and Growing
- LEO Small Satellite Commercial Communications Constellation Size is 46 to 66 Spacecraft
  - 840 Small Spacecraft for "Mega LEO" Teledesic System
  - 10 or 12 Medium to Large Spacecraft for MEO or GEO Systems
- LEO Remote Sensing Constellation Size is 2 Spacecraft
- Most Small Satellites Launched In Groups on Medium ELV's
  - Launch Dates 1997 to 2001
  - 2nd Generation Launch Dates 2002 to 2005
- Broad Mix of Aluminum and Composite Structures Designs
  - Composites More Common in GEO than LEO
- On-Orbit Life 7 Years, 5 Year MMD
- Rigid GaAs Solar Arrays

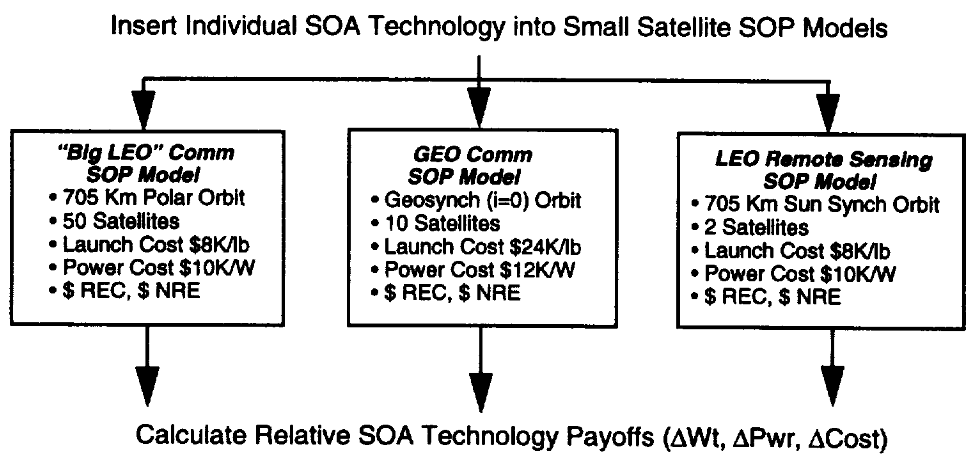
TREND/ MISSION NEED	DEVELOPMENT OPPORTUNITY	C	R
Improved Thermal, EMI/EMC, and Structural Integration - Including Brackets, Fittings, CTE Matching	Conductive Fibers, Composite Heat Pipes, Interface Materials, Thermal Straps, Co-Cured EMI/EMC Materials, BeAl Alloys	√	√
High Volume, Low-Cost Structure Production, Process Control	Modular Components, Standard Tooling, Co-Cured Assemblies, New Processes, Automation	√	
Flat, Lightweight, Compact, Deployable Structures for Solar Arrays and Phased Array Antennas	Triaxial Weave Fabrics, Low-Shock Release Mechanisms, SMA Mechanisms, Inflatable Booms and Potentially Inflatable Solar Arrays	√	√
Spacecraft Packaging for Multiple Launch on Multiple LV's	Non-Pyrotechnic, Low-Shock Booster Release Mechanisms; Possibly Launch Load Alleviation	√	
Precision Pointing/Jitter Stability for Optical Links or Imaging Sensors, Spacecraft Autonomy	Toolkit of Qualified Hardware and Methods for Isolation, Damping, System Identification, Health Monitoring	√	√
Dimensionally Stable Structures: Improved Environmental Compatibility & Reduced Cost	Advanced Materials and Resin Systems, Low-Cost Carbon-Carbon Materials Processing, Thin-Ply Pre-Pregs		√
Environmental Materials for AO/Debris/UV Protection, Thermal Control, Reduced Contamination	Integrated, Protective Designs and Coatings; Improved Understanding of Environmental Effects, Design Guides, Flight Experiments	√	√

Notes: C= Communications, R = Remote Sensing

A simplified approach for conducting the technology payoff analyses is shown below. Using the SOP bus design and metrics described in Section 2.3, SOP system models were constructed for the two focus LEO communications and remote sensing missions. The metrics shown assume that the spacecraft developer is in the process of conducting system definition trades and that the launch vehicle has not yet been selected. An additional GEO system model was developed for evaluating the equivalent technology payoff for GEO smallsat missions. This model assumes that the SOP LEO bus is launched into GEO orbit, i.e., the only differences are the higher launch costs and the reduced number of satellites. The requirements and reference design for the SOP spacecraft bus model are shown on the opposite page.

The trades are conducted by individually substituting each advanced SOA technology into the SOP satellite point designs and calculating the associated changes in weight, power consumption, and cost (in 1993 $) on a per satellite basis. Up-front research and development costs were excluded. The final step is to convert the weight, power, cost deltas to a combined *relative* payoff per satellite based on the equation shown below. Other assumptions and benefits are noted in Section 3.

The overall approach provides a structured way of quantifying the relative benefits or drawbacks of incorporating specific, enhancing, advanced technologies into a typical system. However, caution is warranted in that the quantitative results presented herein are highly dependent on the requirements for each specific mission. Furthermore, commercial system trades will usually involve revenue models.

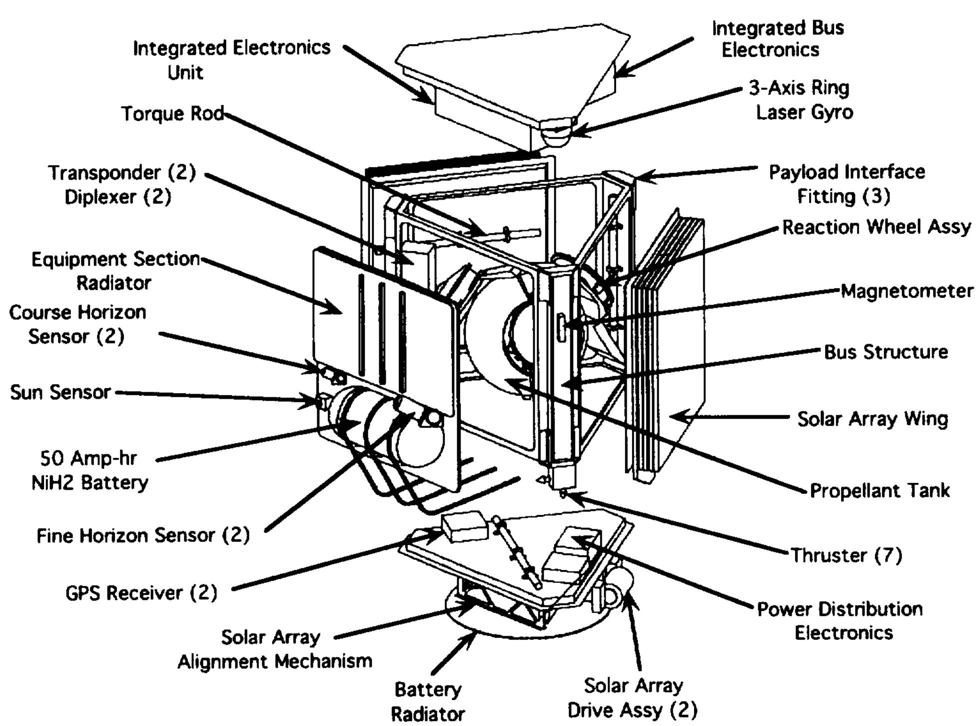


- Payoff per Satellite =
  $(\Delta Wt \times \$/lb) + (\Delta Pwr \times \$/W) + (\Delta \text{ Rec Cost}) + (\Delta NRE/ \# \text{ of Satellites})$

- Other Benefits (Design, I&T, Reliability, etc.) Noted

SOP Small Satellite Bus
Requirements and Reference Design

Launch Vehicles:	• LEO, single launch:	LLV-1/2, Taurus, Conestoga, Titan II
	• LEO, multiple launch:	Delta II, Atlas II, LLV-3, Proton
	• GEO, single launch:	LLV-3, Delta II, Atlas II, Proton
	• GEO, multiple launch:	Proton
Physical Characteristics	• 700 - 800 lb. dry • Triangular shaped graphite/epoxy structure • 3 point payload interface • Payload weights to 1000 lb.	
Electrical Power	• 2 GaAs solar arrays on 2-axis gimbal • 1.5 kW at arrays, BOL • 300 W avr. to payload EOL, LEO • 50 Amp-hr Single Pressure Vessel NiH2 battery	
Attitude Control	• 3-axis stabilized • Attitude knowledge to 0.25° (0.01° w/ star tracker option) • Attitude control to 0.35° (0.02° w/ star tracker option)	
Propulsion	• Blowdown hydrazine monoprop, 200 lb.	
T,T & C	• MIL-STD-1750A processor (16 bit ASCM), 2.5 MIPS at 20 MHz • 8 Mb RAM • 2 kbps SGLS/STDN uplink commands • 64 kbps SGLS/STDN downlink for health and status • GPS for navigation • MIL-STD-1553 interface to payload • Growth capability for serial, analog & discrete interfaces	
Life	• 7 year design life, 5 year MMD	



4.2 SUMMARY OF TECHNOLOGY DEVELOPMENT TRADES

Abbreviated trade studies were conducted for a subset of 11 out of the 25 recommended technology development areas identified in the survey. Emphasis was placed on technology development needs for spacecraft buses, solar arrays, thermal control, and launch vehicle integration for both focus missions. Greater emphasis was placed on technologies for the more numerous communications missions (e.g., phased array antennas) than those for remote sensing missions (e.g., dimensional stability, contamination).

Based on the results of Section 3.1, the top table on the opposite page ranks the relative enhancing SOA technology payoffs in $K per spacecraft for each of the focus missions and the extrapolated GEO smallsat mission model. The GEO model provides an upper bound for systems with a higher sensitivity to launch cost (3X) and a smaller production run. Since the resolution of the cost numbers in the payoff analysis is not really that fine, the bottom table ranks the same technologies in rough order of priority for each SOP model.

The results show that for the assumptions used in the trade studies, the integrated composite bus structures technology offers the greatest relative payoffs. Developing integrated structures technologies will break down technology acceptance barriers and enable composites to be used in LEO more often, more cost-effectively, and with higher performance. The goal is to integrate passive thermal control, thermal straps, minimal heat pipes, EMI/EMC shielding, fittings, brackets, and ground planes into co-cured bus assemblies with fewer parts and process steps. Passive composite radiator and conductive thermal panel technology is ranked second, and is a major contributor to enabling the selection of composites over aluminum for high-performance small spacecraft. Triaxially woven fabrics for lightweight solar arrays and phased array antennas with reduced layup costs ranked next. Non-pyrotechnic booster release mechanisms which reduce shock levels and the cost of shock testing ranked high for both focus missions. Thermal straps for passive thermal control rounded out the top three priorities for each of the three models.

The results of the other, more qualitative technology development trades described in Section 3.2 indicate that isolation, damping, and health monitoring technologies are highly desirable as backup problem-solving tools for both focus missions. Inflatable structures show promise, and could be considered for a few systems (e.g., Teledesic) as a potential alternative. However, the performance and qualification efforts still have a long way to go. Finally, launch load alleviation technology appears promising, but is too new to adequately assess.

Enhancing SOA Technology Ranking

SOA Technology	SOP	LEO Comm (~50) ($K)	GEO Comm (Equiv) (10) ($K)	LEO Rem Sens (~2) ($K)	Other Benefits (Not Costed)
Integrated Gr/Ep Bus Structure	Aluminum	570	1570	710	Automated Assy
Passive Composite Radiators	Aluminum	145+	817+	145+	Simplified Thermal Design, Enables High Perf Bus
Triaxially Woven Fabrics for Stiffness-Critical Antennas & Arrays	T300 Al Core	222	470	94	Reduced Dynamic Interaction, Lower Moment of Inertia
Non-Pyro Booster Release Mechanisms	Pyro Sep Nuts	189	385	405	Reliability, Safety, Compact Satellite Design
Thermal Straps for Passive Thermal Control	Copper Braid	137	420	137	Simplified Thermal Design, I&T Savings, Power & Volume Savings
Integrally EMI/EMC-Shielded Composites	Copper Mesh	64	185	44	Enables Integrated Box Design, Ground Plane
Lightweight Isotropic Materials for Fittings	Titanium	62	205	62	High Strength Fittings for Payload Stacking, Multiple Launch
Integrated Electronics	Aluminum	61	162	-233	Very Small Spacecraft

SOA Technology	SOP	LEO Comm	GEO Comm	LEO Rem Sens	Other Benefits (Not Costed)
Integrated Gr/Ep Bus Structure	Aluminum	1	1	1	Automated Assy
Passive Composite Radiators	Aluminum	2	2	2	Simplified Thermal Design, Enables High Perf Bus
Triaxially Woven Fabrics for Stiffness-Critical Antennas & Arrays	T300 Al Core	2	3	3	Reduced Dynamic Interaction, Lower Moment of Inertia
Non-Pyro Booster Release Mechanisms	Pyro Sep Nuts	3	5	2	Reliability, Safety, Compact Satellite Design
Thermal Straps for Passive Thermal Control	Copper Braid	4	4	3	Simplified Thermal Design, I&T Savings, Power & Volume Savings
Integrally EMI/EMC-Shielded Composites	Copper Mesh	5	7	5	Enables Integrated Box Design, Ground Plane
Lightweight Isotropic Materials for Fittings	Titanium	5	6	4	High Strength Fittings for Payload Stacking, Multiple Launch
Integrated Electronics	Aluminum	5	8	-	Very Small Spacecraft

4.3 CONCLUSIONS & RECOMMENDATIONS

- High-Priority Enhancing SOA Technologies with Significant Payoff for Both Communications and Remote Sensing Missions
  - Integrated Composite Bus Structure
  - Passive Composite Radiators
  - Triaxially Woven Fabrics for Solar Arrays and Antennas
  - Non-Pyro Booster Release Mechanisms
  - Thermal Straps for Passive Thermal Control
- Moderate-Priority Enhancing SOA Technologies with Approximately Equivalent Payoff for Communications or Remote Sensing Missions
  - Integral EMI/EMC Shielding
  - Aluminum-Beryllium Material Development
  - Integrated Composite Electronics Boxes
- Other Important SOA Technology Development Areas Identified and Recommended for Further Study
  - Isolation, Damping, and On-Orbit System Identification/Health Monitoring
  - High-Volume, Low Cost Manufacturing and Test Methods
  - Lightweight Deployable Structures & Low-Shock Release Mechanisms
  - Low-Outgassing, Dimensionally Stable Structures
  - Space Environment-Resistant Materials, Characterization of Environmental Effects, On-Orbit Materials Testing

In conclusion, the first five technologies listed have high payoffs for both focus LEO missions. On a net payoff basis (i.e., when the payoffs per spacecraft are multiplied by the constellation size), the sheer numbers of commercial communications spacecraft translate into tens of millions of dollars in technology benefits. Since all of these technologies also benefit remote sensing missions as well, there is a strong potential for significant return on investment. The large quantities of planned small spacecraft make these technologies attractive even though the structure typically accounts for only 5 - 10% of the weight and 3% of the cost of a small spacecraft.

The next three technologies have medium payoffs and the potential for a moderate return on investment, with significant return on investment for specific systems or applications, such as Teledesic.

Finally, other important SOA technology development areas were identified in the initial survey, but were not traded in a quantitative sense, either because of the limited scope of this study or because the applications were too design-specific. These are recommended for further study and selective evaluation.

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13. ABSTRACT ***(Maximum 200 words)***
This report documents trade studies conducted from the perspective of a small spacecraft developer to determine and quantify the structures and structural materials technology development needs for future commercial and NASA small spacecraft to be launched in the period 1999 to 2005. Emphasis is placed on small satellites weighing less than 1800 pounds for two focus low-Earth orbit missions: commercial communications and remote sensing. The focus missions are characterized in terms of orbit, spacecraft size, performance, and design drivers. Small spacecraft program personnel were interviewed to determine their technology needs, and the results are summarized. A systems-analysis approach for quantifying the benefits of inserting advanced state-of-the-art technologies into a current, reference, state-of-the-practice small spacecraft design is developed and presented. This approach is employed in a set of abbreviated trade studies to quantify the payoffs of using a subset of 11 advanced technologies selected from the interview results. The 11 technology development opportunities are then ranked based on their relative payoff. Based on the strong potential for significant benefits, recommendations are made to pursue development of 8 and the 11 technologies. Other important technology development areas identified are recommended for further study.

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