L·GARDE

SMART SPACE TECHNOLOGY

L • GARDE INC. CORPORATE PRESENTATION

Lightweight Inflatable Solar Array

Patrick K. Malone and Geoffrey T. Williams

JOURNAL OF PROPULSION AND POWER
Vol. 12. No. 5, September-October 1996

Lightweight Inflatable Solar Array

Patrick K. Malone* and Geoffrey T. Williams†
L'Garde, Inc., Tustin, California 92680

A lightweight deployable solar array wing in the 200–1000-W range has been developed, on the Inflatable Torus Solar Array Technology Demonstration (ITSAT Demo) Project. The power density of a flight unit could be as high as 93 W/kg for a 200 W-class wing, including structure and deployment mechanisms. In Phase I, a proof-of-concept torus and array were constructed and deployed in the laboratory. During Phase II, a revised torus and array were constructed and tested at L'Garde and the Naval Research Laboratory. The qualification tests included random vibration, deployment in a thermal vacuum chamber, natural frequency determination, and thermal cycling. The flight design uses 2-mil-thick crystalline silicon ceils on an atomic oxygen-protected flexible Kapton film substrate folded accordion *style* for stowage. The support structure is a rectangular frame comprised of two cylinders (which are inflated then rigidized), the array stowage box, and its cover. The cylinders, flattened, folded and stored for launch, are deployed by inflating with N_2 and rigidized by straining the cylinder laminate material controllably beyond the elastic limit. The engineering protoflight array was designed for optimum power density but, because of availability, some of the components came from excess production runs. Because of this, the actual power density of the test article was 59 W/kg, or 36% lower than the baseline flight array. However, using components as designed, the projected 93 W/kg can be achieved. Because of a simple deployment mechanism, the cost of an ITSAT-type solar array is about one-half to hvo-thirds that of competing systems.

Introduction

IN today's small satellite environment, low cost and high performance determine project success. As such, it is essential to maintain high-reliability, low-cost, and high-efficiency satellite subsystems. The Inflatable Torus Solar Array Technology (ITSAT) accomplishes this by having high-power densities and minimal packaging volume at low cost. ITSAT offers an additional advantage of allowing a variety of photovoltaic media to be used.

The ITSAT program consisted of two phases. Phase I was a feasibility study of point designs for low Earth orbit (LEO), geosynchronous Earth orbit (GEO), and Molniya orbits with array sizes from 100 W to several thousand watts.' A prototype system was built and this phase was completed in October of 199 1. Phase II revised the design of Phase I, generated detailed drawings, and retested the components, producing a protoflight unit. This phase ended with a successful deployment at a temperature of −90°C in the Naval Research Laboratory's (NRL) 9 meter vacuum chamber, and subsequent dynamics testing and thermal cycling in the deployed state. Phase II was completed in January of 1994. The remaining work to be done toward flight demonstration is to refurbish the Phase II unit, finish the remainder of the qualification tests, and integrate with a sponsored spacecraft for a flight test.

Solar Array Current State-of-the-Art

This project originated from the Advanced Space Technology Program (ASTP). Its goal was to define, develop, and demonstrate high-payoff technology applications to improve space system operational support to various military and commercial disciplines.

Under the ASTP effort, a number of small satellites were designed. These satellites. typically with a mass of a few hundred kilograms or less, have stowed volumes that are less than a cubic meter, and require electrical power levels ranging from a few watts up to several kilowatts. Body-mounted solar cells can meet some of the lower power requirements. For power levels above about 100 W, deployable solar arrays are needed. Current solar array designs for more than 100 watts may exceed mass and volume restrictions, both of which are critical.

To meet the solar array requirements of these small satellites, it was clear a new and unique approach was required; one that would involve thin flexible structures and solar arrays.

Solar Array Requirements

Spacecraft performance has always been constrained by power availability. Designers have limited their designs to match the power they could supply to the load. Figure 1 presents future U.S. photovoltaic space power needs. This article concentrates on space power requirements of 0.01 – 1.0 kW, though the ITSAT can be extrapolated to larger systems.

Demand for solar arrays of the type being developed in this project increases with the increasing production and use of small satellites. Both military and civilian applications are being proposed.

Military Applications

The Gulf War showed field commanders the utility of communications by satellite, and the advantage of satellite navigation systems for ground position determination. Both are in demand. Small satellites for both applications may be more cost effective than large satellites.

Civilian Applications

Communications and environmental monitoring are the principal civil applications for small satellites.

The numbers of small satellites required for environmental monitoring are likely to be moderate. Third-World nations are interested because they have less access *to* Landsat and other similar satellite services. However, continuous coverage is not needed as it is for communications applications, and the need can be met with fewer spacecraft.

Received Sept. 29, 1995; revision received March 8, 1996; accepted for publication March 11, 1996.

*Mechanical Engineering Supervisor, 15 18 1 Woodlawn Avenue.
†Project Engineer, 15181 Woodlawn Avenue. Member AIAA.

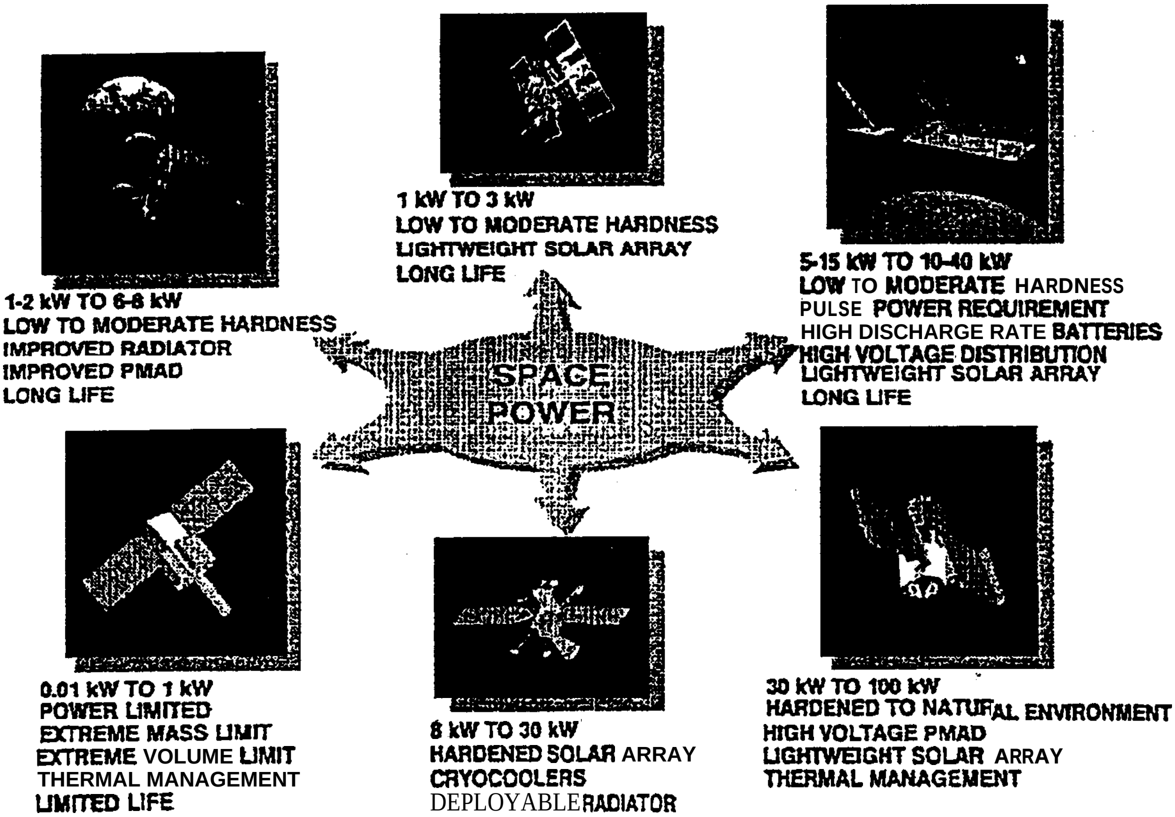

Fig. 1 Space power, the needs.

Table 1 Recent small satellites

Name	Mass, kg	On-orbit average power watts
Macsat	68	8.5
Microsat	23	6.0
Glomr	68	4.3
Ises	82	9–15
Stacksat	68	7-12
Step–0	160 experiments	
	90 spacecraft	
Step- 1	52 experiments	
	43 spacecraft	

Recent Small Satellites

Some examples of recent small satellites are given in Table 1. If approved by the Federal Communications Commission and launched on schedule, these satellites will all be in use by 1997.

A small satellite requiring up to 100 W can use body-mounted photovoltaic (PV) cells. From 100 to about 350 W, deployed but nonarticulating panels or wings can be used. Above about 350 W, articulated panels are preferred. The controlling factors in the choice are the mass and the stowed volume.

When using the Delta II or larger launch vehicles and launching a number of satellites (frequently piggy-backing small spacecraft on the launch of a larger one), weight was not actually a major consideration for the small satellite. With small, less costly launch vehicles, such as the Scout or Pegasus, weight and volume become much more important. ITSAT provides the satellite designer with an option that will allow higher power for a given satellite need, or a lighter system with a smaller packaging volume for a given power need.

StatusofPbotovoltaic Types

For the ITSAT Phase I prototype, several state-of-the art solar cells were considered: 1) thin crystalline silico(Cryst–Si), 2) gallium arsenide on germanium (GaAs/Ge), 3) cleft gallium arsenide (cleft GaAs), 4) copper indium diselenide (CIS), and 5) amorphous silicon (ASi).

Each type of cell has distinct advantages and disadvantages. For example, fabrication processes for thin crystalline silicon and GaAs/Ge cells are considered more mature technologies. As such. the behaviors of these cells are well documented and predictable: their manufacture is relatively routine, they have been through thorough radiation and other qualification testing, and they have been flown in space. Crystalline silicon, usually 0.2-0.3 mm thick, can now be etched to 0.05 mm to save weight. The overall thickness of GaAs, also 0.2 mm, has been reduced to 0.09 mm for the same reason. However. the cells themselves are brittle. An array of them is flexible only in the sense that they can be attached to a flexible substrate that can be accordion-folded in such a way that the cells themselves are not creased.

Although amorphous silicon and CIS cells have the potential of being individually flexible (by deposition of tlphotovoltaic material directly on foils), their disadvantage is that the technology is less developed, and inherently less efficient, requiring larger sizes.

ITSAT Design and Development

Phase I Effort

The prototype configuration is shown in Fig. 2. The flexible array was folded into three panels along the axis of the array, then subsequently rolled up along the same axis. This foldable design had the advantage of allowing the array to be packaged into a canister approximately one-third the width of the array. However, during testing some anomalies were uncovered.

First, this concept required a four-sided torus with comer joints. The stress in the comer joints exceeded that of the material, and necessitated additional stiffening layers of the laminate. While the now stronger torus was able to handle the pressure, the material thickness had grown to the point of being inflexible: it was difficult to package into the canistevolume, negating one of the primary advantages of inflatables, their ability to be packaged into small conformal volumes.

The second problem was that of unfolding the torus in a repeatable, reliable manner. Too many degrees of freedom for the unfolding torus lead to a possible overstress situation for the solar blanket.

The three most important lessons learned from Phase I were as follows:

1) The material thickness must be kept to a minimum to enhance packagability (not to mention weight limitations).

2) A sharp 90-deg comer joint is undesirable because of the stress concentration.

3) The simpler the torus, the more reliable it is.

Phase II Effort

Parallel to the Phase I design effort, other cell technologies were carefully investigated. While there exist several advanced cell technologies with higher efficiencies, none is so well developed or as inexpensive as crystalline silicon. The latter has been used on almost every solar panel satellite since the beginning of the space program and possesses a very large knowledge base to draw from.

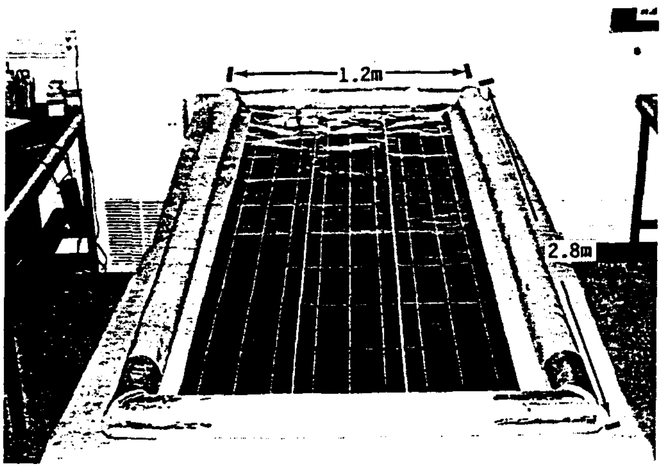

Fig. 2 Pmtotype unit.

Flexible Blanket Technology

The technology on the Advanced Photovoltaic Solar Array Program (APSA) gave us an alternative to the amorphous silicon arrays.* Standard silicon cells are bonded to a blanket of 0.05mm Kapton and interconnected in the usual manner. A flat flexible wiring harness, very similar to that used on computer printers and plotters, is used to route the power from the cell strings to the satellite bus. To allow a large panel to fit into a small volume requires the blanket be folded accordion style, perpendicular to the deployment direction. And while the compound fold used in Phase I allows packaging into a narrower canister, it was abandoned in this design in favor of the simpler unidirectional fold. As with other test panels, a full population was deemed unnecessary; only a 10% population was used.

This return to a tried and true crystalline silicon cell technology solved the problem of a continuously degrading cell such as amorphous silicon; the problem then became one of incorporating a semiflexible blanket into a configuration originally designed for completely flexible arrays.

Overall Design

The new design is shown pictorially in Fig. 3. The comer joints are not used in this design by simply letting the canister and its lid perform as two of the sides of the torus. In fact, the word torus as it was originally defined in the first phase, may now be more aptly described as two parallel booms. The blanket is attached to the booms in several places along its length. by cables going from the hinge pins of the blanket to straps that wrap around the booms. Note that the orientation of the fold lines in the tubes are perpendicular to the fold lines in the blanket. This method limits out-of-plane motion of the inflating boom.

Dry nitrogen inflatant fills the booms to a pressure that forms their original cylindrical shape. After inflation, it does not matter if the inflatant leaks out or not since the extremely thin, but strong. monocoque cylinder is in place. This permanent rigidization eliminates the need for makeup gas. Although the design pressure is 117 kPa, tests on the full length tubes indicate that the inflation pressure can be as low as 76 kPa and still produce a tube capable of taking the expected flight loads.

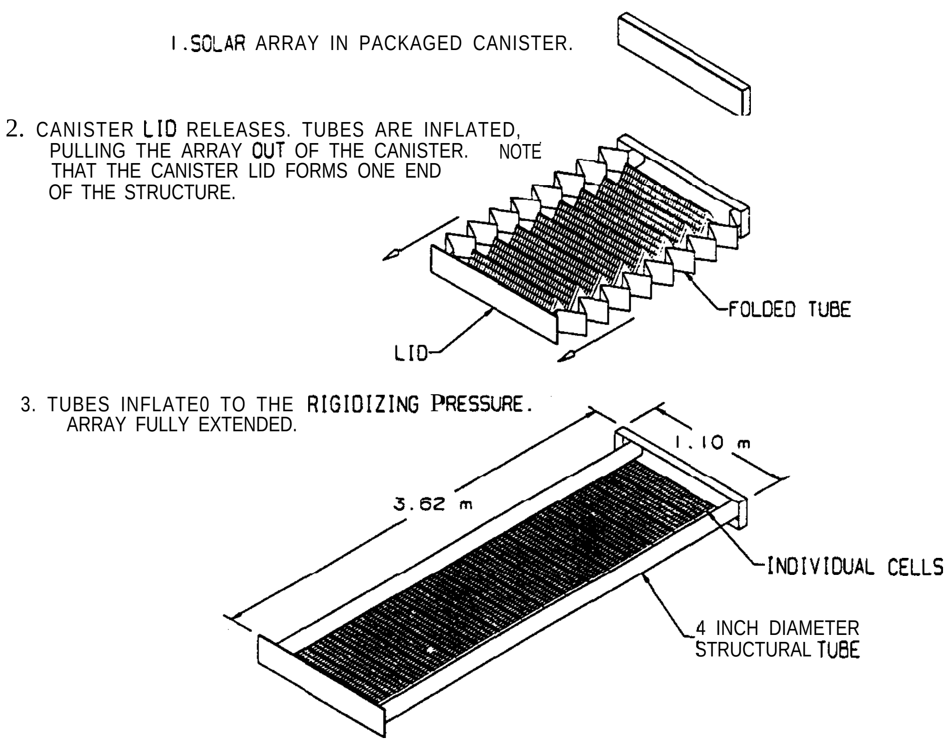

Fig. 3 ITSAT Phase II design.

Table 1 Blanket population summary

Type	Quantity	Thickness, mm	Description
Cells	31	0.05	0.05 mm cells with 0.07-mm covers
Cells	186	0.20	0.20 mm cells with 0.07-mm covers
Simulator	1550	0.15	Aluminum simulators. black anodized
Simulator	421	0.15	Glass
Simulator	44	0.20	Two 0.07-mm layers of glass bonded w/DC93500 (simulating a working solar cell assembly)

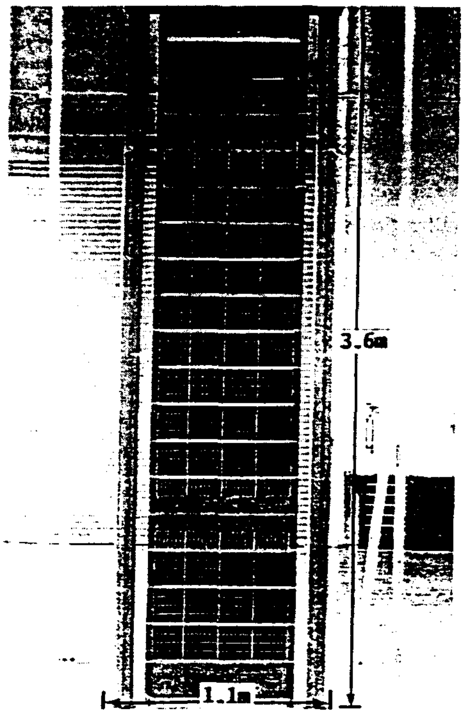

Fig. 6 Qualification test unit.

10-cm-diam tube struts. and previously flown ordnance initiators), a 10% population of 0.2-mm cells and a single string of 0.05-mm cells, with the balance of the cells made up of glass on glass and aluminum cell simulators. The specific power is 59 W/kg at the beginning-of-life (BOL) at an array output of 274 BOL W.

By further optimizing the tubes, enclosure, inflation system, other components, and AR, it is easily possible to increase the specific power to 93 W/kg, still using crystalline silicon solar cells. Specific powers above this level can be realized if nontraditional inflation system components such as sodium-azide are used. Additionally, advanced technology solar cells like cleft GaAs will increase the specific power further by reducing the required cell area. Another advantage of the ITSAT design is its adaptability to a variety of cell types.

The ITSAT surpasses even the Bi-system at power levels above 600 W.

Potential Cost Savings

The ITSAT design has significant cost savings over similar systems. The cost savings for an ITSAT design in the power range of 200-1000 W is 34–48%/W, respectively, over similarly sized designs. Additionally, the ITSAT design can be stowed in a much smaller volume for launch. A large portion of the stowed volume of a satellite or experiment is solar power subsystems. Using the ITSAT design allows for smaller less costly launch vehicles to be used. (This can be an order of magnitude or more saving alone.) The flexibility and simplicity provided by the ITSAT design allows other latent costs such as system engineering, integration functions, and field support to also be reduced as well.

Qualification Testing

After assembling the ITSAT protoflight unit for qualification testing, it was subjected to the following environmental/functional tests: 1) random vibration in all three axes; 2) testing at NRL; and 3) electrical I-V testing of the solar blanket, before and after deployment. These are described in detail in this section.

Random Vibration

The packaged ITSAT qualification unit was subjected to random vibration testing to qualification levels in all three axes (8.8 g rms, 20-2000 Hz). The tests indicated no visual damage to the structure and no degradation in the cell/RTD continuity per the electrical functional test procedure.

Thermal Vacuum Deployment

This and the next two sections describe a set of tests performed on the ITSAT qualification test article at the NRL during October of 1993.

The deployment test was performed in simulated eclipse conditions (vacuum with shrouds at −90°C). The ITSAT was deployed on a slight slant using a test support frame. The slant was present to eliminate the effects of sliding friction. We deployed horizontally in the l-g environment. All equipment and sensors worked perfectly.

The tube pressures during the first 20 s of deployment are shown *in* Fig. 8. Also shown is the pressure profile *from* our flow model for comparison. At time = 0 the lid was released: note that there is no pressure increase until the primary inflation circuit was opened at time = 5 s. Here we see an almost constant tube pressure as the tube is extending. At about time = 13 s, the tubes are fully extended, and the pressure rises slightly because of the constant volume situation. At time = 30 s, the secondary circuit opens, allowing a faster flow of gas into the tubes. The tubes reached their maximum pressures of 123 and 124 kPa at around 90 s after initiation.

There was no damage to the solar array from deployment. All of the 217 working solar cells were intact. A subsequent section gives the results of the electrical I-V tests conducted before and after the NRL test. Of the 465 glass simulator cells, five minor cracks were noted after the test. There is no clear way to tell if these cracks occurred during array packaging, vibration testing, shipment, or deployment. The fact that cracks occurred only in the simulated cells and not in the actual solar cells may be because of their higher fragility.

Dynamics Testing

After backfilling the chamber, the array was hung vertically (Fig. 9), and an exciter motor was set in position to oscillate the array about the z axis. The chamber was pumped down. and the temperature of the test article was allowed to reach the equilibrium temperature of the chamber, 52°C.

The data are given in Fig. 10 and show a natural frequency of 1.04 Hz. The amplitude of oscillation vs the cycle number was plotted and a curve fit made. It consisted of two exponential terms:

$$A = 0.573e^{-0.429x} + 0.519e^{-0.118x} \qquad (1)$$

Where x is the cycle number and A is the amplitude. A description of the two terms is given in Table 3.

Structural Design

The booms running along each side of the array became a development effort in itself. The laminate is aluminum foil sandwiched between two layers of thin plastic. The plastic film is necessary to hold the pressure when inflating by increasing the tear resistance; otherwise the soft foldable aluminum would tear very easily, allowing large leak paths.

The full length tube is shown in Fig. 4. All of the adhesives used have previous space qualification. The tubes are coated with a proprietary atomic oxygen (AO) resistant layer that is also sufficiently conductive to prevent static discharge.

Inflation System Design

The inflation system is shown in Fig. 5. A custom-made tank was required to fit in the low-profile canister. Upon command from the controller unit, a pyrotechnic puncture cutter pierces an aluminum diaphragm to let the gas flow from the tank. There it flows through a primary restrictor that allows only a very slow flow rate. This restriction is necessary to assure that the blanket deploys slowly and safely. The last component of the inflation valve assembly is a vent valve, which allows any trapped air in the packaged booms to vent during ascent in the launch vehicle; thereby avoiding premature inflation. Prior to releasing the gas, the vent is open to ambient. Upon release the vent closes because of the pressure of the gas.

If the inflation system used the primary circuit only, inflation would take much longer than necessary. To avoid the tubes remaining in the unrigidized state for a long period, a secondary circuit is used. Like the primary, it has a puncture cutter to start the flow through a parallel path. However, its restrictor is sized to allow a much faster flow rate, which is safe now that the tubes are extended. After the restrictor. the gas flows through the vent valve, which has already been opened during the primary inflation sequence.

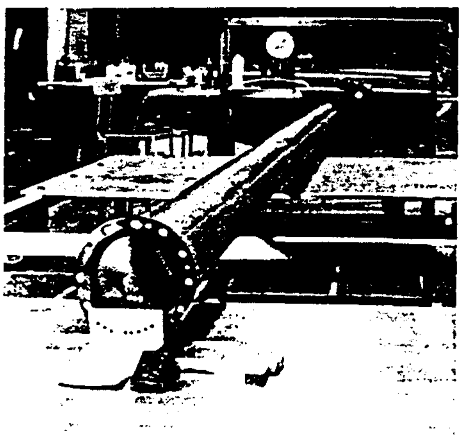

Fig. 4 Flight tube shown **during natural frequency and bending tests.**

Enclosure Design

A composite housing and lid was used to minimize weight and maximize stiffness. The housing consists of 3.2- and 12.7-mm-thick Nomex honeycomb with 0.12-mm-thick graphite–epoxy face sheets bonded to both sides. Reinforcements are used on all comer joints. The lid is constructed similarly, with a small flange overhanging all four edges. The latch mechanism for the lid consists of a pyrotechnic cable cutter, cutting a small cable that holds down the lid. Ejector springs are unnecessary for the lid since the packaged pressure of the booms and solar blanket causes initial separation.

The qualification test unit is shown in Fig. 6. The inflation system is mounted on the back of the housing. The ITSAT assembly mounts to the host spacecraft with four bolts at the center of the housing.

Array Blanket

The lightweight blanket uses 0.05-mm Kapton as the substrate. The blanket is populated by a number of devices, as listed in Table 2. The cell-interconnect-cover (CIC) assemblies are interconnected to each other, with 31 cells per string. The interconnects are soldered cell-to-cell and allow for differential thermal expansion. The cell strings are bonded to the blanket with CVl-1142 adhesive, made by McGhan Nusil.

In addition, resistance-temperature devices (RTDs) are mounted on the back of the working cells to monitor cell temperature. Cell temperature was measured and kept constant at 28°C during electrical current-voltage (I-V) testing. It will also be required for on-orbit performance measurements. Bypass diodes are incorporated to allow for damaged cells and partial shading.

Thermal cycling of the blanket was a concern. To test the strength of the cell-to-substrate bond and the coverglass bonding, a sample of the solar blanket was sent to NASA Lewis Research Center for rapid thermal cycling. No physical damage or degradation occurred during the test.[3] This coupon was cycled from − 100 to +80°C for a total of 2000 cycles.

Current Performance

In summary, the performance of the ITSAT design is shown in Fig. 7. Our current engineering protoflight unit uses proven inflation components (i.e., cold gas inflation, previously tested

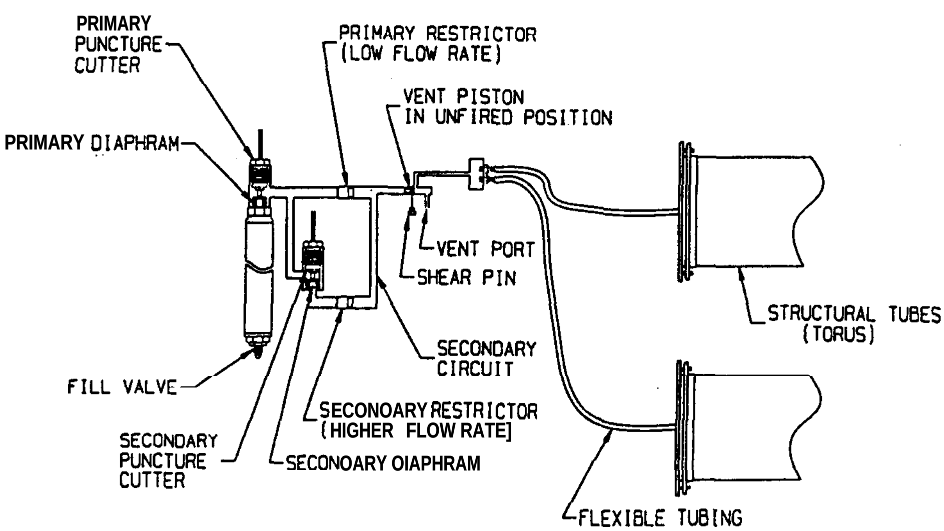

Fig. 5 Inflation system schematic.

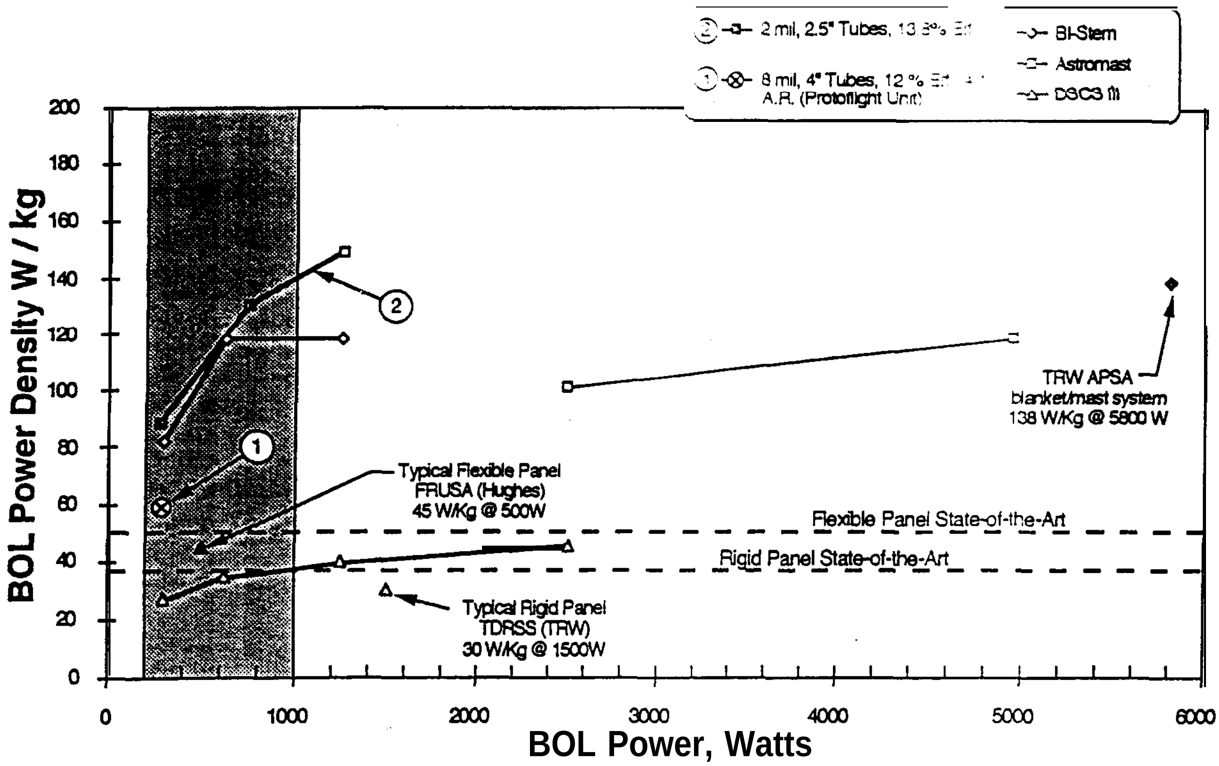

Fig. 7 ITSAT design performance summary.

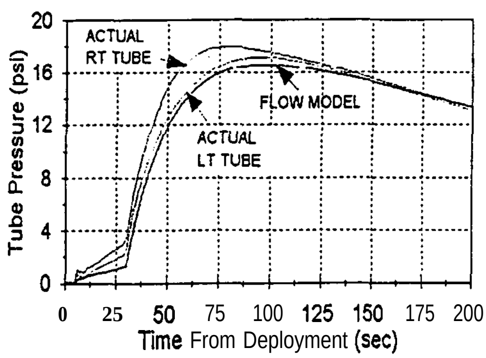

Fig. 8 Tube pressure vs time.

Thermal Cycling

For a 56-h period, the deployed ITSAT was thermal cycled nominally from -85 to +70°C in vacuum conditions. During the cold spikes (zero soak time), the front of the tube was exposed to the shroud temperature (−80°C) while the rear was exposed to cold plates (−180°C). The difference between the front and rear of the tube varied by only 9°C even though they were exposed to a temperature difference of 100°C. The radiative heat transfer between the front and back sides of the tube is large enough to overcome the large difference in exposure temperatures.

I-V Tests Before and After Deployment

Electrical I-V tests were conducted before and after deployment of the blanket at NRL. Each string was tested three times before deployment and three times after deployment. The temperature of the strings were 28 ± 2°C for all tests. No difference could be found in the before and after I-V curves. The strings did not noticeably degrade in performance because of vacuum deployment.

Qualification Test Summary

The qualification testing was successful. The following conclusions were made:.

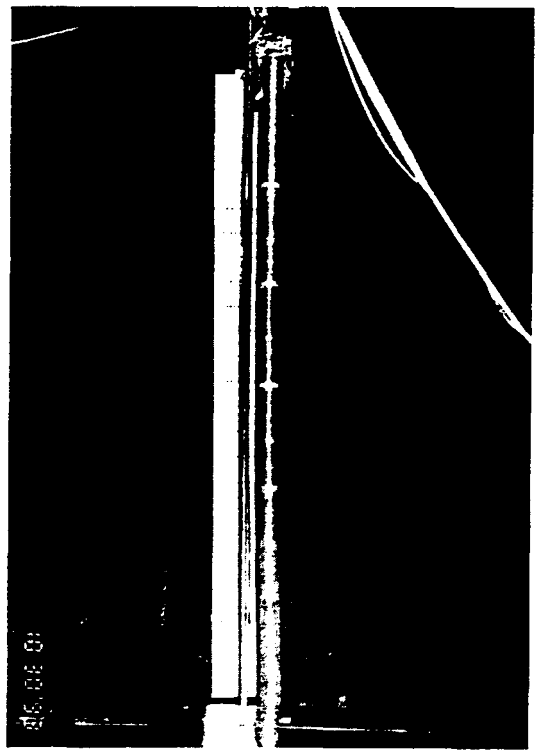

Fig. 9 Dynamics test setup (ITSAT hung vertically from top of NRL chamber).

1) Random vibration had no effect on the electrical or mechanical properties of the array.
2) The ITSAT can deploy successfully in a vacuum and in a simulated eclipse situation.
3) The ITSAT has a natural frequency of 1.04 Hz.
4) The ITSAT was thermal cycled through five complete thermal cycles with extremes of -85 to 70°C, with no physical damage noted.

Table 3 Two exponential terms

$0.573e^{-0.429x}$	$0.519e^{-0.118x}$
35% of amplitude reduced each cycle	11% of amplitude reduced each cycle
57% of energy reduced each cycle	21% of energy reduced each cycle
Probably because. of tubes	Probably because of housing

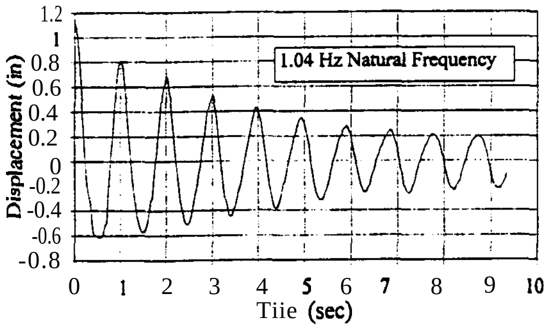

Fig. 10 Oscillation of ITSAT (dynamics test).

5) The electrical performance of the strings did not change because of vacuum deployment.

Flight Test

The plan has been to test a flight-qualified array system in space for a minimum of one year, and to continue the test past the one-year period up to three years. A ride is being sought on a free-flying spacecraft.

The deployment in space will be observed with a small video camera mounted on the host spacecraft. Periodic I-V data will be gathered, evaluated, and compared to solar array output and temperature from the host spacecraft.

Summary and Conclusions

The ITSAT system advances the state-of-the-art by providing power densities greater than many current technologies by using traditional inflation methods. Figure 7 is a comparison of many recent solar array systems in comparison to the ITSAT technologies.

In conclusion, the ITSAT system has shown its feasibility during the Phase I effort and has been ground-test demonstrated during Phase II. It also represents a substantial cost savings for the satellite designer the cost per watt is about one-half that of competing systems.

Acknowledgments

This work was funded by the Advanced Projects Research Agency and monitored by U.S. Air Force Phillips Laboratory. The authors wish to express their thanks to Cost Cassapakis, L'Garde, Inc. and Frank Jankowski, USAF/PL, retired, for their many helpful suggestions. Additionally, we wish to thank George Vendura Jr., of SUMM Associates for his contributions in the areas of solar cell technology and solar cell performance. L'Garde would especially like to thank Joe Hauser and his team at the Naval Research Laboratory for their long hours in support of the ITSAT qualification testing effort.

References

[1]Veal, G.. '*Inflatable Torus Solar Array Technology Program— Final Report," L'Garde, Inc., LTR-91-GV-022, Tustin, CA. Dec. 1991.

[2]Kurland, R., "Advanced Photovoltaic Solar Array Prototype Development & Testing," TRW, Inc., 51760-6005-UT-00, Redondo Beach, CA. Aug. 1991.

[3]Williams, G., and Davey, K., "Thermal Cycling of NASA Coupon," L'Garde, Inc., LTR-93-GTW-036, Tustin, CA, June 1993.

L• Garde Inc
15181 Woodlawn Avenue
Tustin, CA 92780-6487
www.LGarde.com