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# Mission Applicability and Benefits of Thin-Film Integrated Power Generation and Energy Storage

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# MISSION APPLICABILITY AND BENEFITS OF THIN-FILM INTEGRATED POWER GENERATION AND ENERGY STORAGE

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## ABSTRACT

This paper discusses the space mission applicability and benefits of a thin-film integrated power generation and energy storage device, i.e. an "Integrated Power Source" or IPS. The characteristics of an IPS that combines thin-film photovoltaic power generation with thin-film energy storage are described. Mission concepts for a thin-film IPS as a spacecraft main electrical power system, as a decentralized or distributed power source and as an un-interruptible power supply are discussed. For two specific missions, preliminary sizing of an IPS as a main power system is performed and benefits are assessed. IPS developmental challenges that need to be overcome in order to realize the benefits of an IPS are examined. Based on this preliminary assessment, it is concluded that the most likely and beneficial application of an IPS will be as the main power system on a very small "nanosatellite", or in specialized applications serving as a decentralized or distributed power source or uninterruptible power supply.

## INTRODUCTION

Thin-film photovoltaic (TFPV) power generation has been under development for some time. TFPV sample cells and panels have flown in space, but a full TFPV solar array has not yet been built. The principle benefits of TFPV arrays include very high mass specific power (W/kg), radiation tolerance and good stowability. The mission benefits of TFPV solar arrays have been identified [refs. 1 and 2], and may soon be realized once full scale TFPV arrays are constructed and space qualified.

In comparison to TFPV power generation, thin-film energy storage (TFES) is a relatively recent development. Very small thin-film lithium-ion batteries have been developed and tested in the lab for use in multi-chip modules (MCMs) [ref. 3]. With a typical operating range between 3.0 V and 4.2 V, the useable capacity of these initial TFES batteries is very small, ranging from 0.2 to 10 mAh/cm$^2$.

Because of the similarity in the materials and processes that go into TFPV and TFES devices, it is practical to consider combination of the two. It is feasible to combine a TFPV cell on a Kapton™ substrate with a Li-ion thin-film battery sandwiched in Kapton™. With the further addition of very small power conditioning and control electronics, an Integrated Power Source (IPS) is possible, as depicted in figure 1.

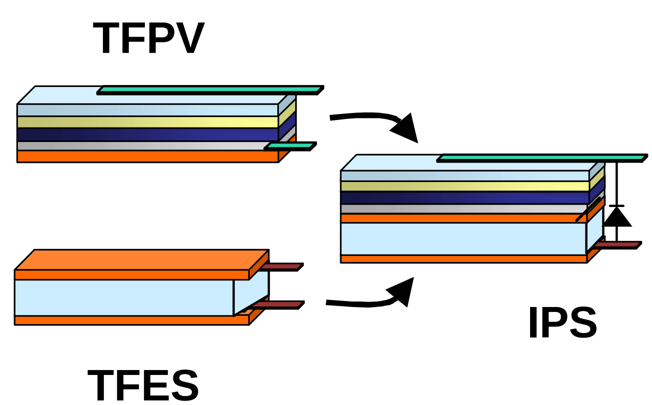


**Figure 1 – Conceptual diagram depicting IPS construction.**

In fact, a number of devices of this type have been built and tested in the laboratory [refs. 4 and 5]. The first in-space demonstration of an IPS, although with a GaAs monolithically integrated module (MIM) solar cell and a Li-ion thick "coin" battery (figure 2), should occur on launch of the Starshine-3 satellite in late summer 2001 [ref. 6].

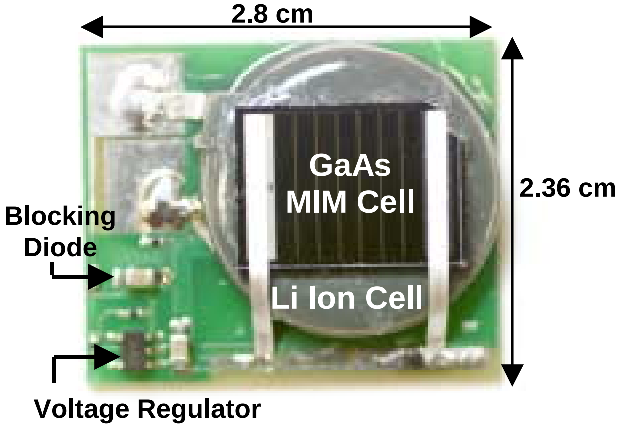


**Figure 2 – Starshine-3 satellite flight unit IPS.**

In the remainder of this paper, the characteristics of a thin-film IPS under development at NASA's Glenn Research Center will be described. The potential benefits and mission applicability of an IPS is assessed. In addition, the challenges that must be overcome in order to realize the benefits are mentioned.

## NOMENCLATURE

| | |
|---|---|
| A | Amperes |
| mA | milli-amperes |
| Ah | Ampere-hours |
| a-Si | amorphous Silicon; solar cell |
| BOL | Beginning Of Life |
| CIS | Copper-Indium-diSelenide; solar cell |
| CPU | Central Processor Unit |
| DOD | Depth-Of-Discharge |
| DRACO | **D**ynamics, **R**econnection, **A**nd **C**onfiguration **O**bservatory |
| EOL | End-Of-Life |
| GaAs | Gallium Aresenide; solar cell |
| IPS | Integrated Power Source |
| Kg | Kilograms |
| LEO | Low Earth Orbit |
| Li | Lithium |
| MCM | Multi-Chip Modules |
| MEM | Micro-Electrical-Mechanical |
| MIM | Monolithically Integrated Module |
| NASA | National Aeronautics and Space Admin. |
| PV | Photovoltaic |
| TF | Thin-Film |
| TFES | Thin-Film Energy Storage |
| TFPV | Thin-Film Photovoltaic |
| UPS | Uninterruptible Power Supply |
| V | Volts |
| W | Watts |

## IPS CHARACTERISTICS/DESCRIPTION

The physical characteristics of an IPS can differ dramatically, and to a large extent will be governed by the specific application. Regardless of the configuration, every IPS will include devices for power generation, energy storage and power conditioning. So far, TFES and IPS systems created at NASA Glenn have been developed to meet the needs of microelectronic devices in space, such as in Multi-Chip Modules (MCM) like the one in figure 3 [ref. 3].

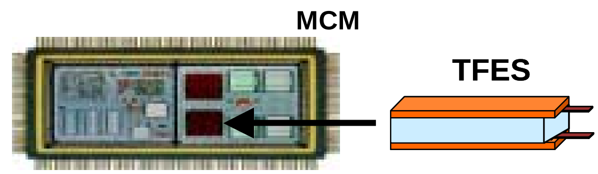


**Figure 3 – Integration of TFES into an MCM.**

NASA Glenn IPS systems have focused on the use of photovoltaic (PV) power generation and Li-ion battery energy storage. However, one can easily envision the use of other energy generation sources (e.g., alpha or beta voltaics, micro fuel cells, etc.) and other storage devices (e.g., super-capacitors, MEM flywheels, etc.) based upon energy needs and mission requirements.

The power requirements placed on an IPS will play a large role in determining the ultimate size of the device. The voltage of the PV portion of the device is determined by the nature of the p-n junction, or, in other words, the materials used. In the case of a GaAs homo-junction device this will be around 1.0 V. For thin-film a-Si or $CuInSe_2$ (CIS) PV, the voltage generated will be somewhat less (0.4-0.8 V). However, through the use of monolithically interconnected modules (MIM), many junctions can be put together in series to increase the voltage. Unfortunately, the available current will always be a function of the active surface area of the device. The current density presently available from a thin-film CIS cell is rather small due to its low photovoltaic conversion efficiency, although the goal of NASA Glenn's in-house TFPV program is >20% efficiency via a dual junction thin-film PV cell like the one illustrated in figure 4.

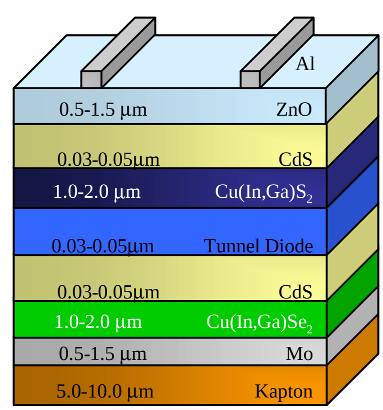


**Figure 4 – Dual junction thin-film cell concept.**

The voltage of a Li-ion battery is based on its chemistry and is primarily determined by the material used in its cathode. A vanadium pentoxide or manganese oxide battery will have and open circuit voltage of 3.0 V, whereas a nickel cobalt cell will be 4.2 V [ref. 7].

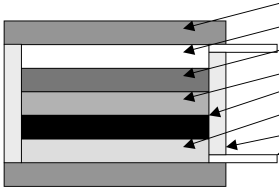


**Figure 5 – Li-ion thin-film energy storage (TFES)**

In a way similar to PV cells, Li battery cells can be connected in series configurations to produce different voltages. However, the amount of energy that can be stored in a cell, its capacity, is determined primarily by its volume. Thus for a thin-film Li-ion battery, the capacity will be determined in the same way the current capability of the PV cell is determined - by the area of the device. The size also impacts the rate at which a battery can be charged and discharged (i.e., the smaller the battery the smaller the charging and discharging currents it can handle).

Ideally, in order to minimize the control electronics associated with an IPS, the photovoltaic array is designed such that its output voltage matches the voltage needs of the battery and its current output is sufficient to charge the battery while simultaneously providing power to the load. The precise sizing of the array and battery will also be dependent on the anticipated illumination scheme. For example, in a typical 90-minute low-earth orbit (LEO) period, the battery will have to support the electrical load for 35 minutes of eclipse. During the 55 minute insolation (daylight) period, the solar array has to provide load power while fully re-charging the battery.

The matching of the solar array and batteries for these small power systems is essential as the parasitic power loss in a conventional charge controller normally used in a larger power system actually exceeds the output of a small IPS. Once the PV and battery are matched, the only additional components required are a blocking diode to prevent the battery from discharging through the PV array during eclipse.

The Li-ion batteries play a large role in determining the temperature regime in which these systems are suitable. Li-ion cells will deliver a sizeable fraction (i.e. 80%) of their capacity at temperatures as low as –20 $^{o}$C [ref 8]. Below such a temperature they do not perform well. However, they do not exhibit permanent damage if they are cycled between larger temperatures regimes (i.e., plus or minus 80 $^{o}$C) [ref. 4]. The high temperature performance is much less of an issue with thin-film Li-ion batteries as they have been shown to operate well at temperatures up to 60 $^{o}$C [ref. 9]. Thermal control issues associated with IPS applications are discussed later.

## POTENTIAL BENEFITS

As one might anticipate, the primary benefit resulting from the combination of two extremely light weight devices providing distinct functions is a less complex, reduced volume, light weight system providing an integrated function. A thin-film IPS could serve as the main power system on a spacecraft or satellite. Scaling up the manufacturing methods should allow an IPS to deliver the highest specific power and energy for the lowest cost. Reducing power system mass, which is typically 20% to 30% spacecraft dry mass, will help reduce launch mass, perhaps enough to enable a mission concept previously too heavy to fly, or allow the use of a smaller, cheaper launch vehicle. Incorporating energy storage with power generation reduces volume formerly required by traditionally separately located chemical batteries, freeing up valuable space for other systems or an increased payload.

Almost completely opposite in approach to a fully integrated single power source is a decentralized or distributed power bus. In this instance, numerous IPSs are used to provide continuous power to loads, either spacecraft bus components or payload instruments, *in situ*, wherever the component is located. Of course this would require components to be located such that they have view of the sun for at least some portion of the orbit. The main benefit of this approach is a reduction in spacecraft complexity, especially with respect to power distribution wiring, simplifying spacecraft integration.

Related to this concept is the use of IPSs as power sources for MCM sensors that may be placed wherever they are needed in a “postage stamp” fashion.

Finally, a hybrid concept would be to use IPSs as un-interruptible power sources, providing “stay-alive” power for select spacecraft components. The main power system may or may not be an IPS in this case.

## CANDIDATE MISSION CONCEPTS

In keeping with the potential benefits outlined in the previous section, specific mission/application concepts for an IPS as a main power system, a decentralized power bus and an uninterruptible power source will now be examined. For two concepts employing an IPS as the main power system, a novel nanosatellite mission and a more traditional LEO spacecraft mission, specific requirements are identified enabling a preliminary IPS applicability and benefits assessment to be performed.

### Main Power System

The requirements for use of an IPS as the main power system will be considered for novel micro/nano satellite missions and a typical LEO earth or space science mission.

An IPS will most likely find its first application on one of the many emerging micro/nano spacecraft mission concepts [ref. 10]. The bus voltage for many of these missions will be 3.3 V, which is ideally suited to a thin-film Li-ion battery. Some of the more interesting micro/nanosat mission concepts entail the use of many satellites in constellations, working together to make simultaneous measurements from various, widely separated locations. NASA's Magnetotail Constellation **D**ynamics, **R**econnection, **A**nd **C**onfiguration **O**bservatory (DRACO) Mission is one proposal among the Solar Terrestrial Probe mission class of NASA's Office of Space Science Sun-Earth Connections theme. The DRACO mission calls for 50 to 100 nanosatellites, defined here as having a mass equal to or less than 10 kg, in nested, near-equatorial orbits sharing the same perigee, with varying apogees out to 40 earth radii. The nominal two-year mission is proposed for launch in 2010/2011. The total load power requirement is 4.5 W [ref. 11].

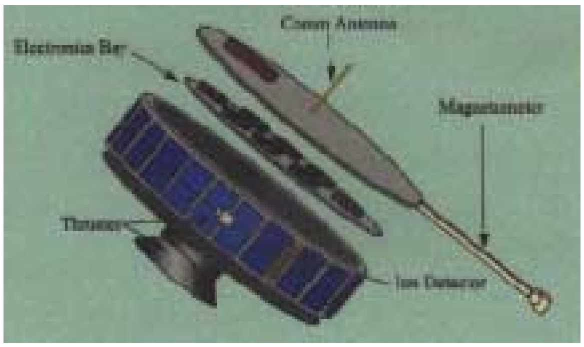


**Figure 6 – DRACO Nanosatellite Concept.**

Similar to the DRACO nanosatellite application, IPSs can be also used to make even smaller, extremely tiny satellite systems consisting of little more than a single chip and a power system. These IPSs would be useful in applications where a constellation of many remote sensors monitors a single, low bit-rate variable. A good example is swarms of nanosatellite weather monitors on Venus or Mars, such as the "Pascal" microprobe network proposed by Haberle and Catling [ref. 12].

However, the low Mars nighttime temperature can be an issue. In this case, the IPS could benefit from the use of an alpha-voltaic cell with an integrated lithium polymer battery; the alpha-voltaic cell would trickle charge the lithium battery, allowing short weather measurement sessions (e.g., two seconds once per hour). In the Venus application, thousands of micro-balloon weather stations, each one with a mass under one gram, can be sprinkled in the atmosphere to compile a global climate map.

On a more traditional earth science mission application, a typical low earth orbit (LEO) Earth or space science spacecraft, similar to the one depicted in figure 7, requires about 500 to 1000 W at 28 V. Mission durations are on the order of two to five years. Accounting for system losses and including a 60% depth-of-discharge (DOD) limit, about 22 Ah of energy storage is required (based on a 500 W load and a 35 minute eclipse).

**Figure 7 – NASA's SeaWinds QuickSCAT LEO spacecraft.**

## Decentralized/Distributed Power Bus

An IPS is unique in that it combines three formerly separate functions of an electrical power system into an integrated package. Taking advantage of this feature, and applying it locally, so to speak, results in a decentralized or distributed power bus.

Use of a local IPS could allow distant portions of a satellite or space probe to be operated without a physical wire to provide power. Wiring can be 10% of the mass of a spacecraft in some cases. Removal of the physical wires would considerably simplify design. The command and control could use infrared or microwave remote control (similar to "wireless internet" control). Potential applications include:

- Actuators for deformable mirrors for large (15-25 meter) space telescopes
- Interferometric sensors
- Wireless remote actuators for spacecraft attitude control
- Gossamer spacecraft controls
- GPS attitude sensors
- Dipole array antenna elements

## Uninterruptible Power Source

A decentralized power bus concept applied to discrete components, leads to the notion of an IPS as an uninterruptible power source, or UPS, to increase the reliability of essential spacecraft functions. Two specific functions that could benefit from this are computer memory and spacecraft communications.

CMOS ("volatile") memory is faster than non-volatile memory and has higher density and lower mass. However, if power is not maintained on the memory, it is erased. The amount of power required for this is extremely small, and a tiny IPS could be incorporated to make certain that even in a low-power condition, the memory remains charged.

Loss of attitude control on many satellites is a fatal error. This can occur when solar arrays lose pointing and batteries discharge. When battery voltage drops so low that the spacecraft central processing unit (CPU) and radio receiver lose power, there is no way to regain control of the satellite. An IPS could be used as a back-up power system, designed to provide enough power to run a low bit-rate omni-directional receiver and the spacecraft CPU only when the main power system failed.

# IPS APPLICABILITY AND BENEFITS

The preliminary results of sizing an IPS for the DRACO nanosatellite and a representative LEO spacecraft are given below. A first-order benefits assessment is also performed.

## DRACO Nanosatellite

Given the 3.3 V bus voltage and low power requirement of only 4.5 W, the applicability of an IPS for this mission as a main power system should be quite promising.

Preliminary studies indicate the DRACO nanosatellites will be 10 kg cylindrical disks 30 cm in diameter and 10 cm in height. The DRACO reference power system has triple-junction GaAs-based crystalline solar cells mounted on eight 11.5 cm x 10 cm panels with a beginning-of-life (BOL) capability of 7 W. Energy storage is to be provided by two lithium-ion batteries sized to support the load during a 1.17 hour eclipse at 60% depth-of-discharge. (All requirements and preliminary design options are from reference 11.)

Since only half of the DRACO nanosatellite cylindrical disk will see the sun at any time, and accounting for the incidence angle on the illuminated panels, the equivalent normally illuminated panel area will be about 0.03 $m^2$. Using a yearly average AM0 solar flux of 1350 $W/m^2$ and a solar cell packing factor of 0.85 while accounting for array integration, power system losses, and including power margins, the minimum required end-of-life (EOL) bare-cell efficiency at operating temperature will be about 15%. Allowing for minor degradation, a TFPV solar cell BOL efficiency will need to be about 18% to 20% at 28 °C. While this TPFV cell efficiency is challenging, it should be attainable before the 2010/2011 timeframe of this mission given development efforts at NASA Glenn and elsewhere. Alternatively, an IPS using a crystalline solar cell would suffice, although with a mass penalty.

As for the energy storage portion of the IPS, using a 60% maximum DOD and accounting for conversion system losses, 3.3 Ah total battery capacity will be required. Since there is 920 $cm^2$ available on the eight 115 $cm^2$ panels, the required specific capacity is 3.6 $mAh/cm^2$. Given that thin-film Li-ion specific capacity is presently at about 2.4 $mAh/cm^2$, improvements are necessary. As alternatives, either the size of each panel could be increased 50% to provide the area required by thin-film Li-ion TFES or alternate Li-ion battery technology could be used. In any case, since the TFES portion of the IPS prefers a warm environment, its location on the inside of the eight cylindrical panels is ideal from a thermal perspective.

The discussion thus far has indicated that a thin-film IPS is feasible for this mission. In terms of benefits, there are both mass and systems integration advantages of using an IPS in this case. First, the TFPV cells will be about two to three times lighter than the baseline crystalline high-efficiency cells. Also, the lithium-ion TFES is lighter than alternative lithium-ion batteries. While these relative benefits are significant, the absolute benefit will be small given the original mass of the baseline components are small. However, with a total mass goal of only 10 kg, any mass reduction can have a significant effect.

While mass is typically a prime discriminator in space missions, the primary benefit of an IPS in this case will be in the spacecraft layout and system integration. With the power generation and energy storage functions integrated and included on the cylindrical structural walls of the spacecraft, three typically separate spacecraft subsystems/functions are combined into one. Also, since the energy storage system is removed from the interior of the spacecraft, there is more room for other system's components. So, not only are there synergistic mass savings, but also the integration and assembly of the spacecraft is simplified. Given TFPV and TFES advancements, or a slightly larger spacecraft bus, nanosatellite applications such as DRACO could be ideally suited to an IPS.

## LEO Spacecraft

Because of the higher voltage and power requirements and the thermal environment experienced by a deployed, sun-tracking solar array wing (as opposed to the small, body-mounted IPS in the previous section), this will be a challenging application for an IPS.

Sized to provide 500 W to the spacecraft loads at EOL and accounting for solar array integration and degradation losses, battery charging during insolation, and other electrical system losses, the total TFPV array area with 20% efficient cells will need to be about 8.5 $m^2$. In comparison, the total array area using 30% efficient multi-junction GaAs-based cells is estimated at less than half of this, or about 4 $m^2$. However, the mass of the flexible TFPV array should be about one-third of the typically rigid crystalline cell array.

To provide the required 22 Ah energy storage, a total TFES area of about 1 $m^2$ will be needed based on an area capacity of 2.4 $mAh/cm^2$. Assuming that the TFES operates near 4.0 V for this application, eight TFES layers will need to be placed in series to meet the 28.5 V bus requirement.

Comparing the required areas, it is seen that the TFES area is only a small fraction of the TFPV. The IPS in this case would consist of a flexible 8.5 $m^2$ thin-film solar array (or pair of 4.25 $m^2$ arrays) with the TFES located on the lower portion of the array along with its associated power condition and charge control electronics.

The benefits of an IPS in this LEO mission application include even better spacecraft volume advantages along with the mass and systems layout/integration advantages identified in the nanosatellite application. The use of flexible TFPV arrays for power generation could reduce array mass by about 67% compared with rigid arrays with crystalline solar cells, although at the expense of increased array deployed area and consequent spacecraft-level impacts (possible field-of-view obstructions and increased moments-of-inertia). In addition, the volume associated with more traditional chemical battery energy storage, especially $NiH_2$ batteries at about 10 Wh/liter, is significantly reduced with placement of the energy storage on the array wing within an IPS. If replacing $NiH_2$ batteries, about 64000 $cm^3$ would be available for other systems and payloads.

However, these benefits may not be fully realizable unless challenges associated with the thermal environment on the solar array wing and series connectivity of the TFES are overcome. A flexible solar array in LEO typically experiences a temperature variation from +80 °C during insolation to –100 °C during eclipse when the energy storage system is active. Passive thermal control techniques such as coatings, or baffles covering the back surface of the IPS where the TFES is located, may be sufficient to maintain the minimum required TFES operating temperature. If not, resistance heaters powered by additional TFES will be necessary.

Based on this preliminary assessment, an IPS as a LEO spacecraft main power system seems feasible, although maybe not the most directly applicable or beneficial application of this device.

## CHALLENGES

There are several developmental challenges that need to be overcome in order to expand the mission applicability of the current IPS technology. With respect to TFPV, attaining 20% cell efficiencies using low temperature processes required for polyimide substrates is a significant challenge. With respect to TFES, as previously mentioned, a major drawback of using current Li-ion batteries is their inability to function at low temperatures as would be experienced on a solar array wing during eclipse, or on the surface of Mars. Another challenge is the limited cycle lifetime associated with the larger capacity polymer batteries. Although, the solid-state thin film batteries do show good cyclability, it is unclear as to whether this can be maintained when they are scaled-up to provide larger capacities. Also, the efficient use of parallel and series combination of Li thin-film batteries has yet to be demonstrated. Finally, one of the largest challenges today is in finding the appropriate sealing technologies for the Li batteries that will inhibit their degradation while also being able to withstand the rigors of space flight.

## CONCLUSION

Combining three traditionally separate power system functions into a single, integrated device is a unique concept made possible by advances in thin-film technology. Assuming further improvement in both thin-film power generation and energy storage performance, applications most likely to first use and benefit from a thin-film IPS as a main power system will be those with low power and low voltage requirements. Upcoming missions with these characteristics are those using constellations of very small spacecraft, or nanosatellites. IPSs may also enjoy nearer-term applicability in specialized instances where they can serve as de-centralized or distributed power sources or un-interruptible power supplies for discrete components.

Given the early stage of development and their inherent performance limits, it remains unclear as to whether or not thin-film IPSs will find widespread applicability. However, since thin-film power generation and energy storage will be developed independently for other reasons, the techniques for combining each of these functions into an IPS, along with any required power conditioning, will be further refined. As better performing devices are built, any applications and associated benefits that can possibly be imagined will undoubtedly be further explored.

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This paper discusses the space mission applicability and benefits of a thin-film integrated power generation and energy storage device, i.e., an "Integrated Power Source" or IPS. The characteristics of an IPS that combines thin-film photovoltaic power generation with thin-film energy storage are described. Mission concepts for a thin-film IPS as a spacecraft main electrical power system, as a decentralized or distributed power source and as an uninterruptible power supply are discussed. For two specific missions, preliminary sizing of an IPS as a main power system is performed and benefits are assessed. IPS developmental challenges that need to be overcome in order to realize the benefits of an IPS are examined. Based on this preliminary assessment, it is concluded that the most likely and beneficial application of an IPS will be as the main power system on a very small "nanosatellite," or in specialized applications serving as a decentralized or distributed power source or uninterruptible power supply.

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