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14. ABSTRACT

This paper discusses results of a program to develop an innovative solar thermal propulsion system for application to orbit change and mobility for small spacecraft. In this system, solar radiation is collected by the concentrator which transfers the concentrated solar radiation to the optical waveguide cable consisting of low-loss optical fibers. The optical waveguide cable transmits the high intensity solar radiation to the thermal receiver for efficient, high performance thrust generation. Through the course of the preceding program, we have demonstrated the base for the system hardware. This paper discusses results of the program to develop and evaluate an engineering model of the solar thermal propulsion system based on the OW technology.

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Solar Thermal Propulsion for Small Spacecraft

- Engineering System Development and Evaluation -

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This paper discusses results of a program to develop an innovative solar thermal propulsion system for application to orbit change and mobility for small spacecraft. In this system, solar radiation is collected by the concentrator which transfers the concentrated solar radiation to the optical waveguide cable consisting of low-loss optical fibers. The optical waveguide cable transmits the high intensity solar radiation to the thermal receiver for efficient, high performance thrust generation. Through the course of the preceding program, we have demonstrated the base for the system hardware. This paper discusses results of the program to develop and evaluate an engineering model of the solar thermal propulsion system based on the OW technology.

I. Introduction

Solar thermal propulsion has been considered to be an efficient propulsion method for orbit transfer from LEO to GEO. A large technical database has been developed under solar rocket technology programs with funding from the Air Force, NASA and other agencies since the 1970's.[1-10] For solar thermal rocket application, a set of large-scale, lightweight inflatable parabolic mirrors is to be used to focus solar radiation into the thruster's absorber cavity. The propellant gas, heated by the concentrated solar radiation, achieves specific impulse, Isp, on the order of 850 to 1100 s.

During the Air Force sponsored program in 1979, the off-axis inflated concentrator solar thermal propulsion system was developed. In this design configuration as shown in Figure 1, the concentrator and the absorber/thruster are optically coupled with the absorber located at the concentrator focus. Due to its large-size inflated concentrators and non-rigid support structure, the optically coupled concentrator-absorber configuration can be sensitive to structural deformations caused by concentrator subsystem rotation or acceleration. These deformations could create problems of blurred concentrator focus or misalignment. If the thruster can be decoupled from the concentrator, the absorber/thruster can be placed at a site best suited for effective thrust generation, and the high intensity solar beam can be directly transmitted via flexible optical waveguide transmission lines to the solar absorber/thruster. Such a solar thermal thruster can satisfy a wide variety of spacecraft propulsion needs.

Physical Sciences Inc. (PSI), in collaboration with Rocketdyne Propulsion & Power, The Boeing Company, is developing an innovative solar thermal propulsion system for application to small spacecraft with funding support

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by the Air Force Research Laboratory. In this system, as schematically presented in Fig. 2, solar radiation is collected by the concentrator which transfers the concentrated solar radiation to the optical waveguide transmission line consisting of low-loss optical fibers. The optical waveguide cable transmits the high intensity solar radiation to the thermal receiver for efficient, high performance thrust generation. Part of the solar radiation can be switched to attitude control thruster as necessary. The features of the proposed system are:

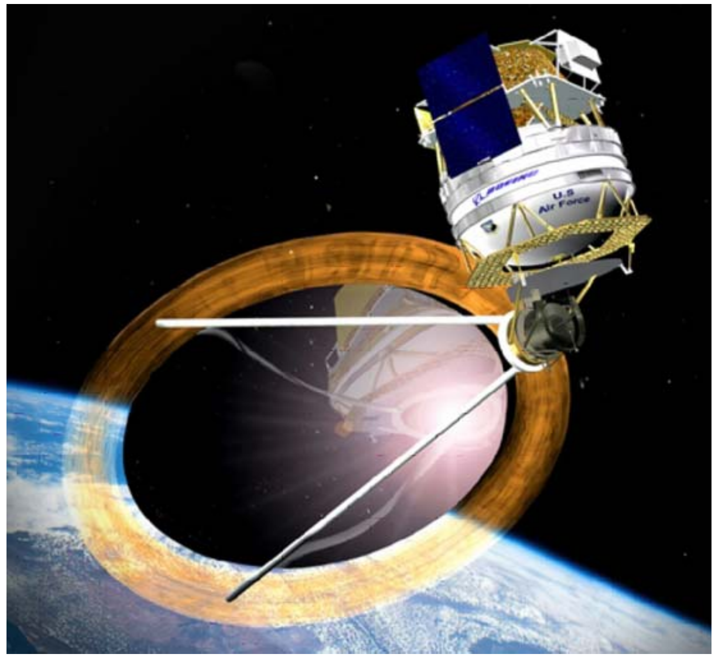

Figure 1. The off-axis inflated concentrator solar thermal propulsion system.

1. Highly concentrated solar radiation (~10^3 suns) can be transmitted via flexible optical waveguide transmission line to the thruster's absorber cavity;
2. The flexible optical waveguide linkage de-couples the thruster from the concentrator to provide freedom from the constraints imposed on previous solar propulsion system designs;
3. The configuration of the solar receiver can be optimized for efficient heat transfer with minimal re-radiation loss;
4. Aiming and tracking for the concentrator become significantly easier by moving the termination of the optical fiber cable to follow the focal point of the primary concentrator; and
5. High intensity solar radiation can be switched to different receivers to deploy several thermal thrusters as necessary.

II. Technology Development Program

In this section we will discuss results of the SBIR Phase I and the ongoing Phase II program to develop and evaluate engineering prototype of the proposed system shown in Fig. 2.

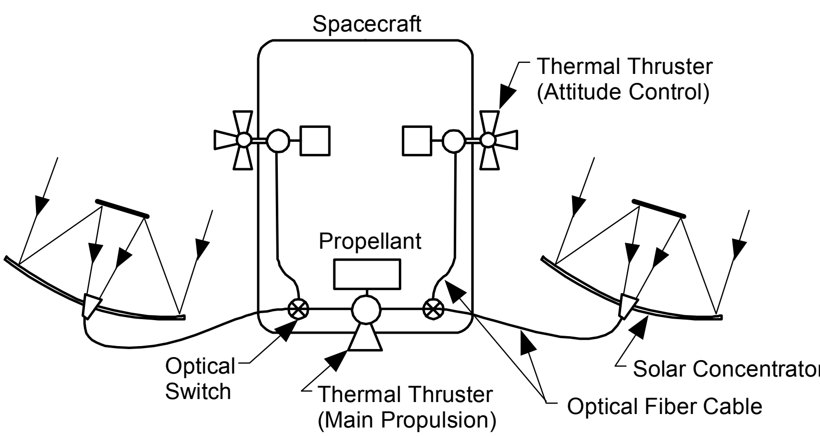

Figure 2. Solar thermal propulsion system for small spacecraft.

Phase I Summary

During the Phase I program we conducted proof of principle experiment to validate the basic concept using the PSI Optical Waveguide (OW) Solar Thermal Facility. Figure 3 shows the Phase I experiment. The experimental facility consists of two solar tracking units each with two 50 cm parabolic concentrators. The two concentrators are mounted on a rotating frame to track the sun. The optical fiber cable placed at the focal point of the concentrator transmits the concentrated solar radiation to the solar receiver located at the center of facility. The optical fiber cable (4 m long) consists of 37 fused silica fibers (1.2-mm dia). The four optical fiber cables deliver about 200 W of solar power into the receiver. The solar receiver is located at the center with four optical fiber cables connecting it to four concentrators. The configuration of this experimental setup simulates the solar thermal propulsion system described in Fig. 2.

Figure 3. Experimental setup for the solar thermal test conducted in Phase I.

The hardware components that we developed in this program include: optical waveguide transmission line; interface optical components; and the solar thermal receiver.

The optical waveguide transmission line

The optical waveguide transmission line is the key component to integrate the concentrator system with the solar thermal receiver. The cable inlet interfaces with the concentrator system and the cable outlet interfaces with the solar thermal receiver. The cable inlet design we used in this program is based on our heritage: the quartz secondary concentrator collecting the solar radiation and injecting it to the optical fibers. Figure 4 shows the inlet portion of the four optical fiber cables used for this program. All four cables are 4 m long and each consists of 37 high numerical aperture (NA = 0.48), low-OH polymer clad optical fibers (HWF1200/1235/1600 T48) manufactured by CeramOptec. The fiber has an excellent off-axis transmission up to 25 degrees.

The design of the cable outlet was developed for optimum interface with the high temperature solar receiver. A photo of the fiber cable outlet is given in Fig. 5. The 37 optical fibers transfer the solar radiation to the 10 mm quartz rod. The quartz rod, by the principle of total internal reflection, transfers the solar radiation to the thermal receiver. The tip of the quartz rod is placed close to the receiver high temperature heat exchanger in order to deliver the solar power directly to the receiver heat exchanger elements.

Figure 4. The inlet of the optical fiber cable.

Figure 5. The optical fiber cable outlet made of quartz rod.

Solar Receiver

One of the important objectives of this program was to demonstrate the basic solar receiver heat transfer mechanisms:

- Transport of high intensity solar flux from the concentrator to the solar receiver via optical fiber cable;
- Efficient delivery of high intensity solar flux to the solar receiver heating element;
- Achievement of high temperature via radiative heat transfer; and
- Viability of optical components.

A schematic of the solar thermal receiver is given in Fig. 6. The solar receiver core is made of graphite cylinder (diameter = 1.75 cm; height = 2.54 cm), because of (i) high solar absorptivity (α = 0.7~0.9), (ii) excellent thermal-mechanical stability, and (iii) ease of fabrication. The gas was injected tangentially into the graphite cylinder and flows out through the molybdenum tube. The graphite core is surrounded by the molybdenum radiation shields. Solar power (~200 W) was delivered to the graphite core by four quartz rods (dia. = 1 cm).

The solar receiver housing with four optical fiber cables is shown in Fig. 7. The construction of this housing was similar to the materials processing experiment conducted in the previous NASA Program.[11] The propellant gas flows from the bottom of the housing, flows through the heat exchanger, and flows out of the housing.

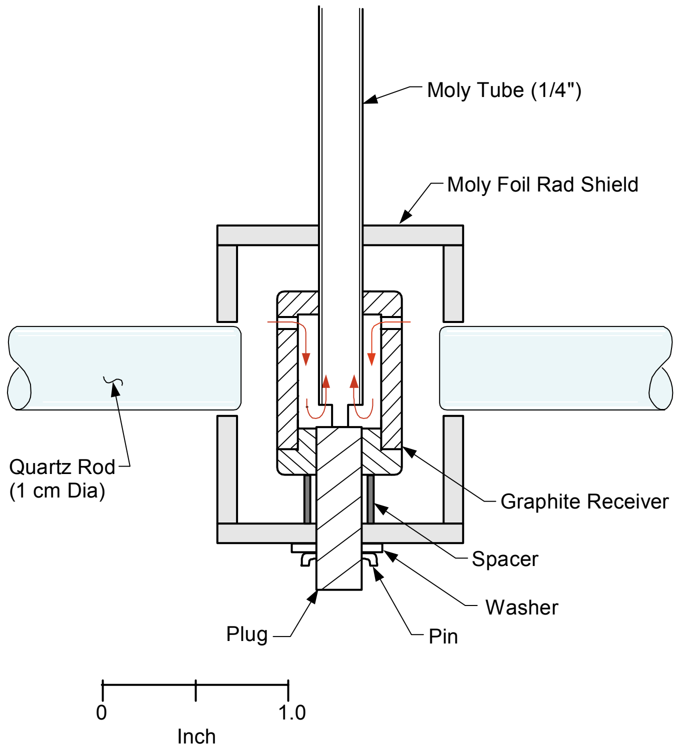

Figure 6. Schematic side view of solar receiver.

Figure 7. The solar receiver housing with four fiber cables connected.

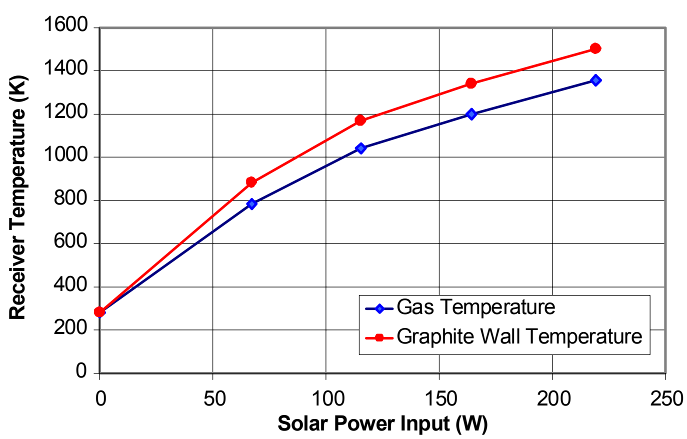

Figure 8. Receiver temperature vs solar power input for argon gas (flow rate: 6.34×10^{-3} mole/s; asymmetric heating; 12/8/03).

Results

A total of 9 hours of tests was conducted with argon and helium as working gases. In the experiment with argon at the atmospheric pressure, the maximum temperature reached for the graphite receiver was 1502 K, and that for the argon gas was 1357 K. Figure 8 shows the temperature plot for the graphite receiver vs. solar power for argon gas at the atmospheric pressure.

We conducted the experiment with argon under reduced pressure to achieve the highest temperature within the limited solar power available. For the low pressure experiments, the solar receiver housing was connected to the vacuum pump. The configuration of the solar receiver and the gas flow pattern were the same as those for the atmospheric pressure tests. The results showed that the receiver temperatures for both cases are nearly identical. We conclude from the results that, at least for argon, the dominant factor in the receiver is the radiation flux intensity and the convective heat loss is relatively small.

During the experiment all components of the system functioned properly. After the series of experiments, the apparatus was disassembled and each component inspected for possible damage. No deterioration was observed. It is noteworthy that the Quartz rods in the receiver have been placed only 5 mm away from the receiver wall which was heated to 1500 K (see Fig. 6). These positive findings underscore an important attribute of the system: that the optical waveguide can deliver the concentrated solar radiation to the location best suited for efficient heating of the propellant.

Phase II Program

The purpose of the Phase II program is to develop and evaluate an engineering model of the solar thermal receiver based on the optical waveguide (OW) technology. The specific tasks to be conducted in this proposed Phase II program are:

1. Develop the key components of the optical waveguide solar thermal propulsion system.
2. Integrate the components to establish the engineering model of the optical waveguide solar thermal propulsion system and evaluate the performance.
3. Develop the conceptual design for a proto-flight model of the solar thermal propulsion system for small spacecraft and evaluate its performance.
4. Perform a system level comparison of the optical waveguide solar thermal propulsion system with a conventional solar thermal propulsion system.

At the writing of this paper, development of key components is proceeding. Integration of the components and testing of the system will start as key elements of the engineering model are integrated. In parallel with hardware development effort, system level comparison of the optical waveguide solar thermal propulsion system with conventional system has been made. In the following section we will present a brief analysis of such comparison.

III. System Application

The optical waveguide (OW) system can be applied to any system in which solar radiation is collected, concentrated, and transmitted to a single location, or multiple locations, for use as a high temperature thermal energy source. Currently the potential uses are: lunar material processing using solar energy and high energy space solar thermal propulsion (interplanetary injection and orbit transfer propulsion). This paper focuses on a Low Earth Orbit (LEO) to Geosynchronous Equatorial Orbit (GEO) transportation system using an OW solar thermal propulsion system.

The potential benefits of the OW solar thermal propulsion system compared to a conventional solar thermal propulsion system for an orbit transfer vehicle include reduced total launch mass as a result of: (1) a reduced primary propulsion system mass due to an increase in optical train efficiency (this tends to be offset by the added mass of the OW subsystem) and (2) a lower attitude control system (ACS) mass (inert system and propellant mass) through the integration of an OW subsystem into the ACS using "hot" hydrogen or ammonia (compared with monopropellant hydrazine ACS). In addition, the OW solar thermal propulsion system offers an innovative operational flexibility in which the concentrator, receiver/thruster, and spacecraft can be independently oriented.

LEO-to-GEO Transfer: System Trade Guidelines and Assumptions

The mission chosen for this system evaluation was an orbit transfer from LEO (400 km, 28.5 degree inclination) to GEO with an assumed payload and spacecraft bus mass of 151.5 kg. The primary orbit transfer engine thrust was assumed to be 1.0 N from a single solar thermal engine (2,300 K propellant design temperature) with four facets on a single boom structure as shown in Fig. 9. The propulsion configuration and design conditions were similar to those used in other Air Force programs. The OW solar thermal propulsion system was compared to a conventional direct gain solar thermal propulsion system with the above design conditions. Both thrusters had a delivered specific impulse of 795 sec.

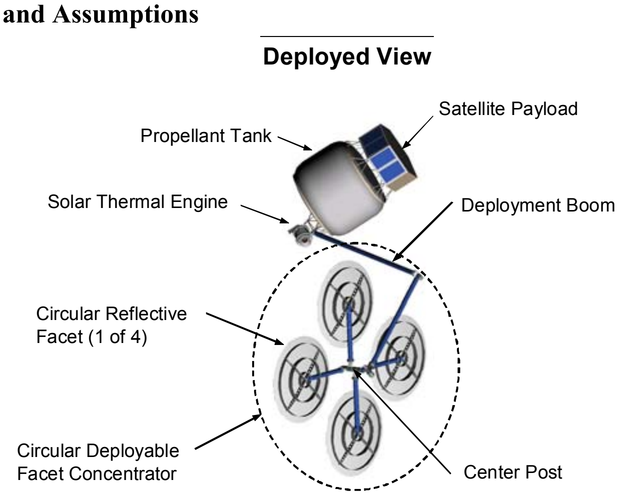

Figure 9. Solar Thermal Propulsion System Configuration.

Using projected optical element efficiencies, the OW system optical train efficiency was 0.575 compared to the conventional solar thermal propulsion system value of 0.531.

The reference attitude control system (ACS) was a fixed (ten monopropellant hydrazine thrusters) thruster system (specific impulse of 220 sec.). The OW ACS was an integrated system with either "hot" hydrogen or ammonia thrusters (two thrusters) with individual receivers and separate ACS propellant tanks as shown in Fig. 10.

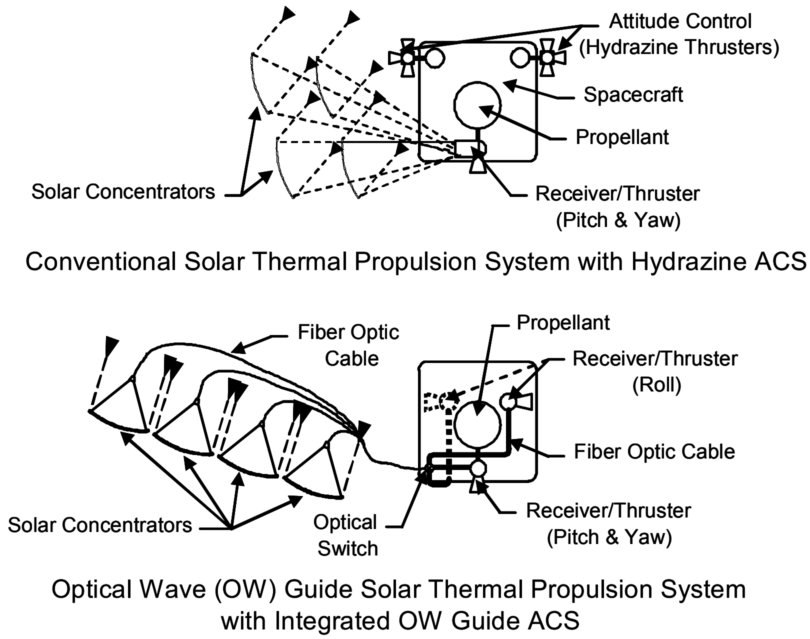

Figure 10. Conventional Monopropellant ACS and Integrated OW Guide ACS.

Mission Analysis Results

Mission analyses were performed for: (1) incorporating the OW system for the primary LEO-to-GEO propulsion system, and (2) a complete integrated OW solar thermal propulsion for both the primary propulsion and ACS. ACS capability from approximately 5,000 to 45,000 kg-sec were considered. This ACS capability would enable the spacecraft to conduct missions in the GEO vicinity.

Bottoms up subsystem mass estimates were determined for the various spacecraft elements and a 30% inert mass margin was assumed.

Primary Propulsion Comparison

Considering the OW system for only the primary propulsion system, the system did not provide a net mass savings with off-axis concentrators, but did realize a net launch mass savings (initial mass in LEO, IMLEO) with on-axis concentrators. The OW system resulted in approximately a 36 % mass reduction in the concentrator mass, but the added mass of the OW guide system offset this savings and a net mass savings was only achieved if the concentrator areal density was greater than 3.1 kg/m^2 as shown in Fig. 11. As discussed previously, the OW solar thermal propulsion system enables a complete independent movement of the concentrator, receiver/thruster, and spacecraft. This provides the operational flexibility of the spacecraft to orient its antenna or other instruments independent of the orientation required by the propulsion system (sun tracking and thrust orientation). The OW system increases the system complexity due to the addition of the OW components.

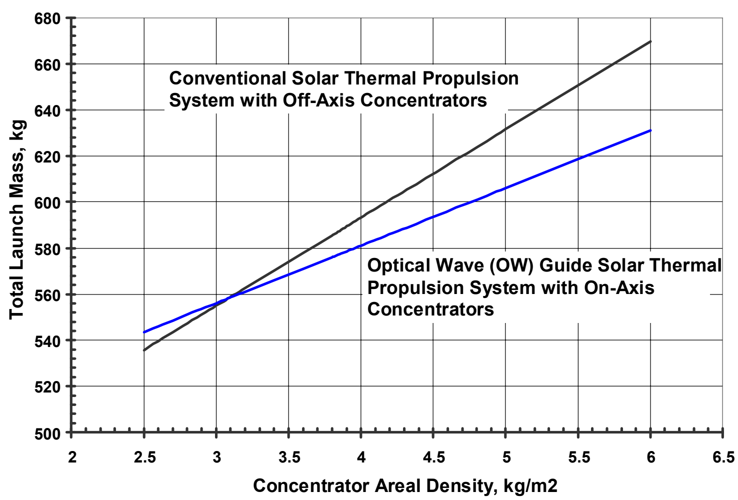

Figure 11. Representative Conventional and OW Guide Solar Thermal Propulsion System Launch Mass Comparison.

Primary and Integrated ACS Propulsion Comparison

The results of an OW solar thermal propulsion system with an integrated OW ACS are presented in Figure 12 and 13 for hydrogen and ammonia ACS, respectively.

For this portion of the study, concentrator areal density of 3.002 kg/m2 was assumed for the conventional solar thermal propulsion system. As shown in Fig. 12, an integrated OW ACS system with hydrogen resulted in a launch mass savings of 8 to 35 % which increased with total ACS impulse. With ammonia as the ACS propellant (Fig. 13), the launch mass savings was 5 to 23 % also increasing with increase in total ACS impulse. Of course, the mass savings

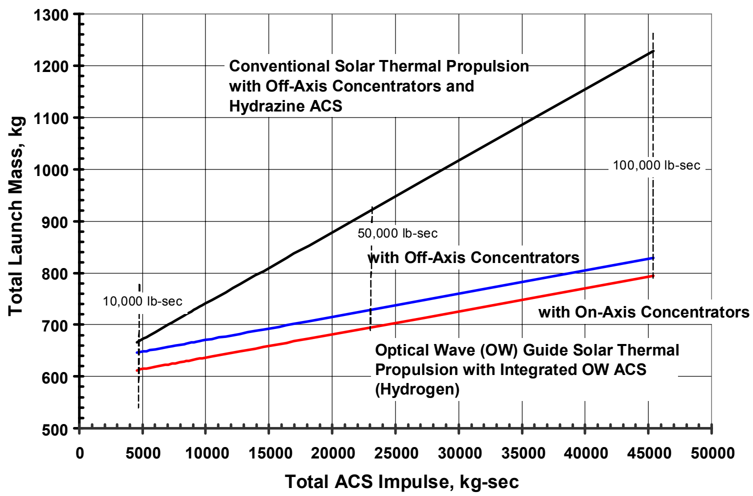

Figure 12. Launch Mass Comparison of OW Guide Solar Thermal Propulsion System with Integrated OW Guide ACS (Hydrogen ACS) with Conventional Solar Thermal Propulsion System with Monopropellant ACS.

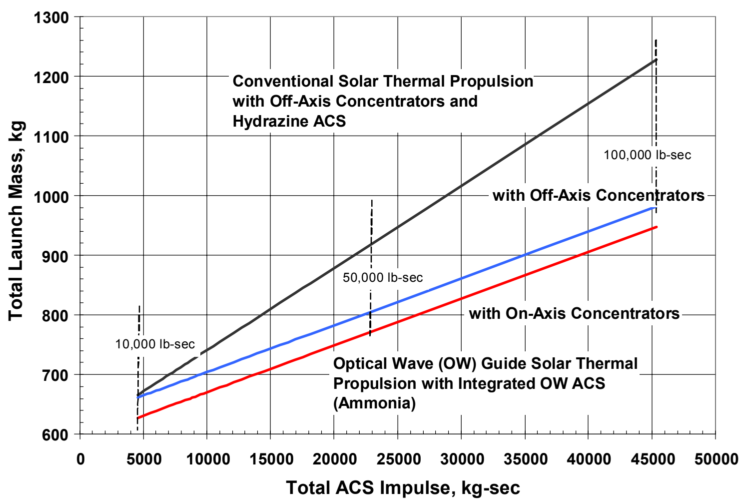

Figure 13. Launch Mass Comparison of OW Guide Solar Thermal Propulsion System with Integrated OW Guide ACS (Ammonia ACS) with Conventional Solar Thermal Propulsion System with Monopropellant ACS.

with ammonia were less due to the lower thruster specific impulse (386 sec compared with 795 sec). The OW ACS is limited to "on-sun" operation which should not be a significant impact at high altitudes where the eclipse time is a small percentage of the orbit time. For hydrogen ACS, long term propellant storage will be an issue.

IV. Summary

In light of the experimental results and analyses of the system in Phase I study, we believe that we have an opportunity to provide the key enabling technology for solar thermal propulsion for application to small spacecraft. System comparison study conducted to date pertaining to LEO to GEO orbit transfer indicates potential savings in launch mass. More experimental and analytical studies pertaining to the engineering aspects of the system are being prepared.

Acknowledgment

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