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Design Considerations for Space Transfer Vehicles Using Solar Thermal Propulsion

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DESIGN CONSIDERATIONS FOR SPACE TRANSFER VEHICLES USING SOLAR THERMAL PROPULSION

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Abstract

The economical deployment of satellites to high energy earth orbits is crucial to the ultimate success of this nation's commercial space ventures and is highly desirable for deep space planetary missions requiring earth escape trajectories. Upper stage space transfer vehicles needed to accomplish this task should ideally be simple, robust, and highly efficient. In this regard, solar thermal propulsion is particularly well suited to those missions where high thrust is not a requirement. The Marshall Space Flight Center is, therefore, currently engaged in defining a transfer vehicle employing solar thermal propulsion capable of transferring a 450 kg payload from Low Earth Orbit (LEO) to Geostationary Earth Orbit (GEO) using a Lockheed Launch Vehicle (LLV3). The current design uses liquid hydrogen as the propellant and employs two inflatable elliptical off-axis parabolic solar collectors to focus sunlight onto a tungsten/rhenium black body type absorber. The concentration factor on this design is projected to be approximately 1800:1 for the primary collector and 3:1 for the secondary collector for an overall concentration factor of nearly 5400:1. The engine, which is about twice as efficient as the best currently available chemical engines, produces 8.9 N of thrust with a specific impulse (Isp) of 860 sec. Transfer times to GEO are projected to be on the order of one month.

Introduction

The Marshall Space Flight Center (MSFC) has performed an in-house feasibility study to assess the technical and economic feasibility of a Solar Thermal Upper Stage (STUS)[1]. This stage utilizes concentrated solar energy to heat hydrogen propellant to a high temperature after which it is expelled through a nozzle to produce thrust. All aspects of the stage were studied for technical feasibility. The results indicate that the design, development, and operation of a solar thermal upper stage is feasible; however, it was also found that there are several key components which will need advanced technology development.

In particular, tungsten material fabrication methodologies will need to be developed in order to produce an absorber which will be able to withstand the thermal cycling resulting from the numerous startups and shutdowns required of the engine. Tungsten is a notoriously difficult material to work with and the successful production of an STUS absorber will require that innovative fabrication techniques be mastered along with the possible use of various tungsten/rhenium alloys and other types of materials. There is currently an effort underway at the MSFC to construct several prototype absorber assemblies to evaluate various materials and fabrication methodologies.

In the collector area, a great deal of work remains to be performed with regard to the precision manufacturing and testing for controllability of large, inflatable, ultraviolet (UV) rigidized structures. In addition to the absorber and collector components, developments are required in composite tank and fluid management technologies to enable the

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effective control of cryogenic propellants in a microgravity environment.

Solar Thermal Upper Stage Mission

The STUS mission involves the transfer of approximately 450 kg of payload from Low Earth Orbit (LEO) inclined at 28.5°, to a Geostationary (GEO) orbit (inclined at 0°). The STUS payload is first injected into a 740 km circular orbit by the Lockheed Launch Vehicle 3 (LLV3) after which multiple (133) perigee burns are performed to increase the apogee altitude. The perigee burn times, starting at one hour in duration, increase in length throughout the apogee raising portion of the mission. After apogee has been raised to GEO altitude, the STUS begins the first of 22 apogee burns during which time the perigee is raised to the GEO altitude and the inclination is reduced to 0°. Proper phasing is required to align the nodes and apsides to achieve a particular longitude at GEO. This transfer technique involves a compromise between transfer time and propellant consumption. The gravity losses incurred for the large burn-arcs are accepted and weighed against long transfer times and propellant boil-off rates. The baseline transfer time is 34 days.

Configuration

Figures 1 and 2 illustrate the Solar Thermal Upper Stage (STUS) in both its flight and launch configurations. The estimated mass of the stage is 770 kg including a 20% dry mass contingency. The LLV3 (Lockheed Launch Vehicle Three) with three or four strap on Castors and a large shroud was selected as the primary launch vehicle based on preliminary estimates of the required shroud volume and stage lift requirements. At launch, the propellant tank is the largest component of the STUS. The outer diameter of the propellant tank is primarily influenced by the inner diameter of the launch vehicle shroud. The 2.96 m outer diameter of the tank established the length of the tank which was required to contain a volume of 17.8 m^3 of liquid hydrogen. Since the initial assumption was made that the STUS payload would be contained within the two frustum sections of the nosecone, the tank was positioned at the top of the cylindrical section. The Reaction Control System (RCS) tank and the absorber/thruster assembly were then placed between the bottom of the propellant tank and the top of the launch vehicle payload adapter.

With most of the major components and interfaces defined, the support structure was defined. The major elements of the support structure are the forward tank skirt, aft tank skirt, upper adapter and lower adapter. These structural elements are of the skin and stringer type with the skin radially inward. This inward surface allows for the mounting of avionics boxes, etc. The reaction control system (RCS) tank is mounted with struts which are attached to the interface flange between the aft tank skirt and upper adapter. The absorber/thruster assembly is mounted with struts which are attached to the interface flange between the upper adapter and lower adapter. A two degree of freedom

Figure 1
SOLAR THERMAL UPPER STAGE FLIGHT CONFIGURATION

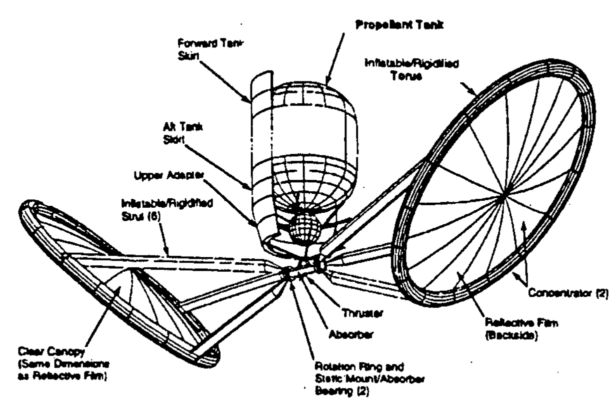
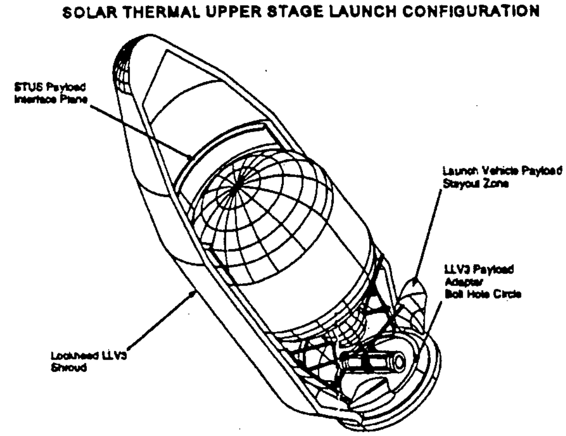


Figure 2
SOLAR THERMAL UPPER STAGE LAUNCH CONFIGURATION



gimbal is located between the struts and the absorber/thruster assembly for the purpose of thrust vectoring. The two inflatable concentrators and their associated inflatable/rigidified tori and struts, are the largest STUS flight configuration components. Since the two concentrators systems are inflatable, however; they can be packaged in small deployment canisters at launch which are a small fraction of their deployed volume.

STUS Thruster Assembly

The purpose of the STUS thruster assembly is to absorb the sunlight incident upon it from the solar concentrators and transmit that energy to the hydrogen working fluid after which is then expelled through a nozzle to produce thrust. The thruster assembly is gimbaled on the STUS for pitch and yaw control during each main burn.

Sunlight enters the thruster through a secondary concentrator which concentrates the incident sunlight by a factor of 3. The light then enters the absorber cavity whose walls absorb the incident sunlight. The cavity walls have spiral internal channels through which the hydrogen working fluid flows, absorbing the solar energy as it goes along. As the hydrogen travels along the cavity walls, it is heated to a high temperature and injected into a collection plenum after which expelled through a nozzle to produce thrust. The thruster assembly is constructed of a tungsten/rhenium alloy and is surrounded by a carbon foam insulation. Figure 3 illustrates the design of the thruster and Figure 4 shows the engine performance as a function of propellant temperature. At the operating I_{sp} of 860 sec, the corresponding propellant temperature is 2540 K. Figure 5 illustrates the light distribution within the absorber for an axisymmetric normal light distribution with an entrance half angle of 30°.

A simplified thruster design is currently in the process of being constructed to evaluate different materials and manufacturing techniques. A vacuum plasma spray (VPS) process is being used initially to construct axisymmetric thruster assemblies out of tungsten, rhenium, and hafnium carbide. The simplified design consists of a simple light pipe with no secondary concentrator. The hydrogen working fluid flows through a spiral channel around the outside of the light pipe and is expelled through a conical nozzle at the end of the absorber. Figures 6 and 7 illustrate the light and temperature distribution expected from this simplified design. Figure 8 illustrates the efficiency variation of the simplified absorber as the absorptivity of the light collecting surface is changed. The figure shows that there is a broad maximum in efficiency at an absorptivity of approximately 0.5.

Mirror Concentrator Assembly

Due to its inherent light weight and perceived relative simplicity, an inflatable concentrator was chosen as the means by which solar energy is collected and directed into the thruster assembly. The surface shape of the reflector is parabolic, but because the reflector is off axis, the projected circular area of sunlight on the parabola forms an ellipse. The two inflated collectors are pointed at the sun by rolling the vehicle and rotating the collector. The size of the elliptical collectors are 5.03 m x 7.16 m.

The concentrator torus is made of epoxy impregnated kevlar cloth with the reflector film being constructed of aluminized mylar covered with

Figure 3

SOLAR THERMAL UPPER STAGE THRUSTER ASSEMBLY

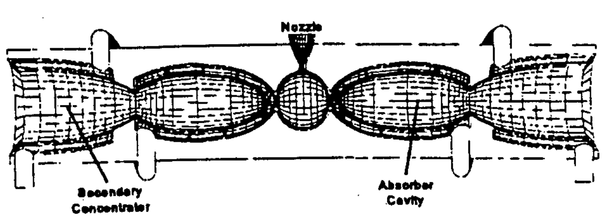


Figure 4

STUS ENGINE PERFORMANCE

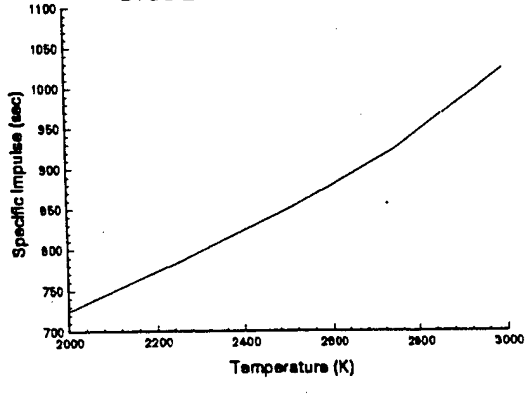


Figure 5

SOLAR ABSORBER LIGHT DISTRIBUTION

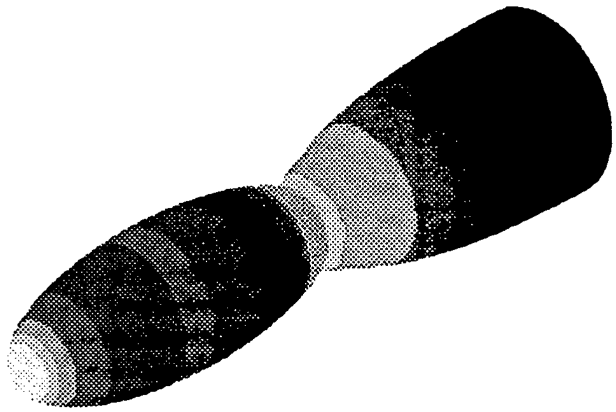


Figure 6

SIMPLIFIED SOLAR ABSORBER LIGHT DISTRIBUTION

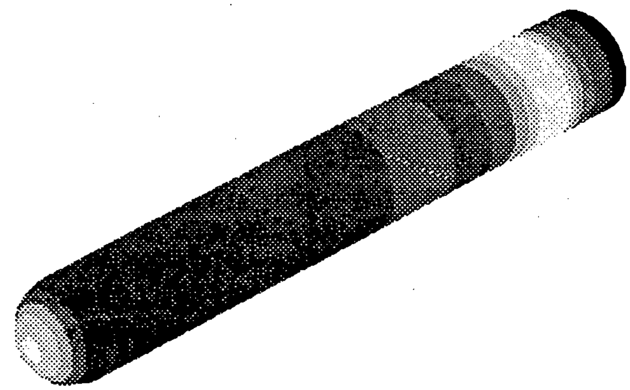


Figure 7

SOLAR ABSORBER WALL TEMPERATURE DISTRIBUTION

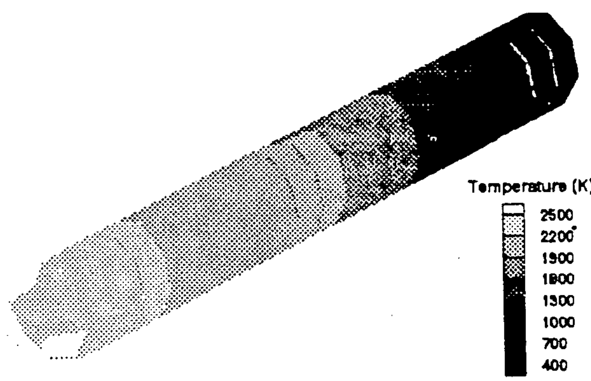
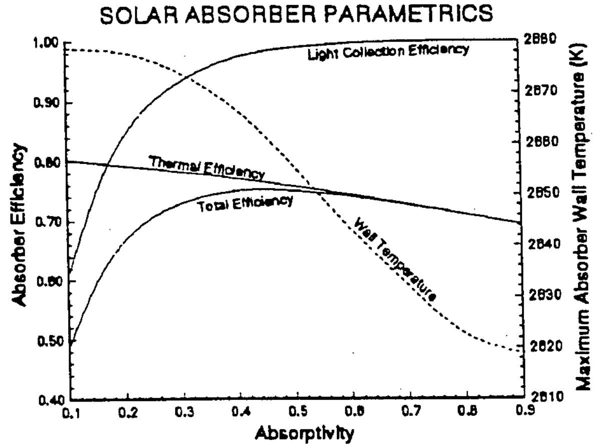


Figure 8

SOLAR ABSORBER PARAMETRICS



a clear canopy. Deployment is initiated when a latch is released and the case containing the concentrator opens in clam-shell fashion whereupon a pneumatic system is activated. The collector support struts are inflated first, followed by the torus, with the collector being inflated last. Upon deployment, collector structural stability requires that the torus be inflated to an internal pressure of 34.5 kPa and the reflector canopy to a considerably smaller internal pressure of 0.021 kPa. The struts and torus rigidify in space as the gelatin-impregnated fabric slowly drys. Because the collector struts and torus rigidify once they are deployed, makeup gas is not required to maintain their structural stability once the curing process is completed. Flexible solar cells are located on the rigidified inflated struts. Figure 9 illustrates the current design of the concentrator.

The efficiency of a solar rocket engine is directly relatable to the concentrating ability of the solar collectors. Figure 10 illustrates how mirror tolerance affects the concentration factor. As the tolerance on the mirror system is relaxed, the spot size of the light at the focal point increases, reducing the effective concentration factor. Here concentration factor is defined as the ratio of the area of the primary mirror to the area of the light spot. The present system is required to have a primary mirror concentration factor of 1800 which corresponds to a mirror system tolerance of 0.4 degrees.

Figure 9

INFLATABLE TORUS REFLECTOR

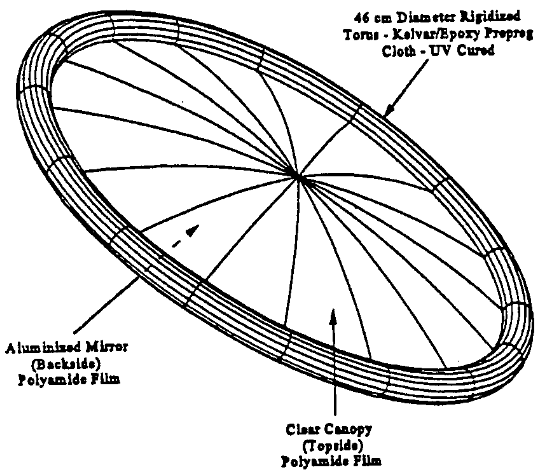


Figure 10

MIRROR PARAMETRICS

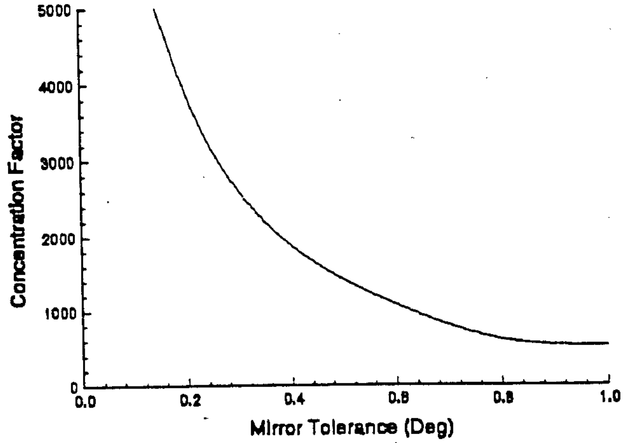
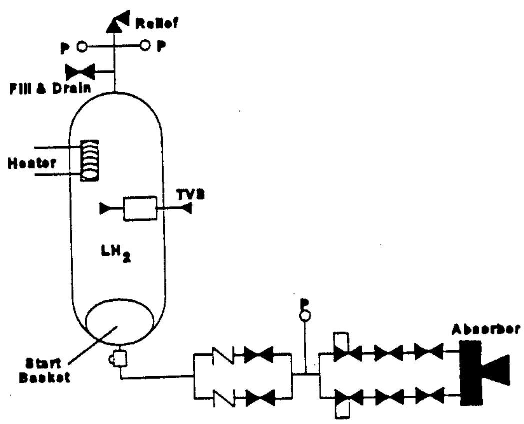


Figure 11

SOLAR THERMAL PROPULSION SYSTEM SCHEMATIC



Thermal Control, Fluid Management and Propulsion Feed System

The main propulsion feed system is based on a single fault tolerant design for the critical components so that the failure of a single critical component will not cause mission failure. The system is pressure fed from a cryogenic hydrogen tank using components which are off the shelf thus requiring no advanced development effort.

The hydrogen tank is required to store liquid hydrogen for a minimum of 30 days. This long term hydrogen storage requirement along with the large number of engine burns and the weight sensitive nature of the entire vehicle have major implications for the fluid management system and its interaction with the thermal control and feed systems. Tank pressure, in particular, is a crucial parameter in determining the structural weight of the tank and to overall system performance. Currently, it is assumed that heaters inside the hydrogen tank will be used to provide the pressurization required to force the liquid hydrogen through the feed system. It is estimated that approximately 20 watts of thermal energy must be supplied to the tank to expel the liquid hydrogen propellant at the required 8 pounds per hour, assuming a tank saturation pressure of 30 psia and a submerged heater with perfect mixing. Since the tank heating rate from the space environment averages about 10 watts, a 10 watt heater in addition will be required to provide the energy necessary to expel the liquid at the desired flowrate. Based on the engine burn frequency and burn durations which ranged from one to four hours, the heater thermal energy requirements each day are estimated to be about 140 watt-hours at the beginning of the mission, decreasing to between 40 and 80 watt-hours at the end of the mission. The total energy added to the tank over the course of the mission will be that required to vaporize approximately 32 pounds of liquid hydrogen.

Figure 11 illustrates a schematic of the overall solar thermal propulsion system.

Conclusions

Solar thermal propulsion provides a cheaper more efficient alternative to chemical propulsion for transporting heavy payloads from low earth orbit to geosynchronous orbit. The main drawbacks to using this propulsion concept lie in the fact that because of the low thrusts involved, satellite transfer times will typically be in the order of a month; and due to the precise pointing required and the large number of burns, the engine system will be fairly complicated to operate. From a technical viewpoint, there does not appear to be any insurmountable problems to developing highly efficient solar thermal engines with specific impulses of over 860 s. If the proposed design goals are met, it should be possible to launch satellites to geosynchronous orbits using smaller launch vehicles than are currently being used with today's chemical engines.

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(1) MSFC Technical Study Team: "Solar Thermal Upper Stage (STUS), In-House Feasibility Study," (1994).