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THE PLACE OF SOLAR THERMAL ROCKETS IN SPACE

1

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ABSTRACT

The harnessing of sunlight for propulsive energy is a recurring theme in space propulsion, particularly for applications requiring large velocity increments, such as planetary exploration or comet rendezvous. Characteristically, it is viewed in terms of the solar sail and the solar cell; but for operations in earth orbit these approaches are less desirable because the very low thrust leads to undesirably long maneuver times. Thrust levels several orders of magnitude higher are available with solar-thermal rockets, while preserving a specific impulse advantage over chemical systems. The performance advantages, penalties, technological problems, and approaches are examined for solar thermal rockets. Its suitability in several earth orbit missions is assessed. The peculiarities of vehicle design, the nature of the thruster and the solar concentrator are presented, and AF plans to implement the development of solar rockets are outlined.

INTRODUCTION

Proposals to use solar energy for propulsion purposes appear in several contexts, the best known and most studied of which is the solar cell in combination with the ion engine (SEPS). This concept competed (successfully) with another solar propulsion system, the solar sail, for the Halley's Comet rendezvous while that was a viable mission. When the mission faltered the SEPS concept retreated to a surviving mission application in earth orbit where its Isp advantages were less important, and its deficiencies in thrust harder to overlook. In today's constrained budget environment, it found no safe harbor there, and was cancelled. The demise of such a distinguished predecessor must be pregnant with lessons for a fledgling propulsion concept with a large developmental effort yet to be completed. Its uses and range of applicability as well as its deficiencies should be well understood in the early planning of its development effort.

As shown in Figure 1, the characteristics of the solar-thermal rocket are the most nearly like those of chemical rockets among the three main solar propulsion approaches. Its Isp is too low relative to electric propulsion to enable a competitive position in long-duration interplanetary missions, but in earth orbit, where military space missions are likely, its higher thrust levels and shorter maneuver times are overriding. It also promises to be lighter and more simple in design. With these collection of potential advantages, it is of interest to inquire as to why the concept was not pursued more energetically in the past.

HISTORY OF SOLAR THERMAL PROPULSION

A solar-thermal rocket was described (Ref 1) by Ehricke in 1956 which contains many of the essential features of the concept. He envisaged the use of inflated plastic spheres as solar concentractors. Sunlight would enter the sphere through a transparent side and be concentrated by reflection from the mirrored opposite side to an absorber internal to the sphere. Liquid hydrogen was proposed as propellant. It would be pumped through the absorber and then back out of the sphere to the thrusters. Ehricke recognized the desirability of high thrust to minimize the so-called gravity losses of low thrust trajectories. He also recognized the hardships in obtaining high thrust with dilute energy sources such as sunlight, as shown by the size of his collectors and choice of a low weight inflatable design. Propellant temperatures up to 3600°F with resulting specific impulse values of 700 seconds were envisioned. It is doubtful, however, that these levels of performance could be achieved with the approach described, due to the limitations of spherical concentractors, the nature of the proposed absorber, and losses in the long lines through which heated propellant would be transported back of the spheres to the thrusters.

Detailed analysis of the solar rocket concept with subscale experimental demonstrations of solar propellant heating were carried out in the early 1960's (Ref 2) by Electro-Optical Systems, Inc. (EOS) under a contract to the Air Force Rocket Propulsion Laboratory. On the experimental side, the contractor accomplished a number of significant things. He made cold flow studies of hydrogen in tiny nozzles to evaluate the boundary layer losses under these conditions, graduating thence to hot flow studies using arc-heated hydrogen. These hot tests included measurements of temperature and total enthalpy achieved in the heated hydrogen, and also thrust measurements using an impulse target mounted in the exhaust flow. These thrust measurements yielded Isp values up to 680 seconds. Although these arc-heating tests were done for better understanding of nozzle phenomena, they had indirect application to solar heating since the arc-heated gases were introduced to the nozzles through a tungsten "calming chamber" that might resemble a solar absorber. Finally EOS measured temperatures in solar-heated hydrogen using a five foot diameter concentrating mirror of search-light quality and solar absorbers made of molybdenum and of rhenium respectively. Although thrust measurements were not taken, temperatures of nearly 2300°K were achieved, corresponding to Isp approaching 700 seconds as judged by the arc-heated nozzle results. The significance of the concentrator employed is that it marked the first use of the electro-forming technique for building light-weight concentrating mirrors needed for practical solar propulsion systems.

On the analytical side, EOS compared solar rockets with chemical rockets and with arc jets and ion engines for various orbit-raising applications and in evasive maneuvering spacecraft. The results are summarized in Figure 2 for an orbit-raising maneuver from low earth orbit to geosynchronous orbit for a 6000 lb gross weight spacecraft. Shown are the payload weights and other weights for a point design chemical system and electric system compared with a thrust optimization chart for a solar rocket. It was found that mirror mass cut into payload at high thrusts, while hydrogen boil-off and additional tank insulation which attended longer mission duration were payload limiters at low thrust. In between was an optimum payload region where solar rockets out-performed the LOX/LH_2 chemical systems. The optimum thrust was between one and two pounds, which resulted in mission times of about twenty days. The ion propulsion system gave payload advantages over the solar rocket, but at a penalty of about an order of magnitude in trip time.

At this juncture, late in 1963, the solar rocket idea fell on evil days. Despite the generally up-beat findings of the EOS study, it received no additional funding and little is found in the literature to say why. But the reasons were both programmatic and technical. Insufficient funds forced a decision between solar rockets and a competing advanced concept, the isotopically heated thruster. The deciding issue was one that was scarcely broached under the EOS study, part technical and part aesthetic. It fell under the term "vehicle integration", which refers to the general awkwardness of a spacecraft possessing the enormous mirrors needed to concentrate the dilute energy fluxes from the sun at earth distance. The awkwardness was visible in the Ehricke design with the long distances over which heated hydrogen was pumped. Figure 3 shows the EOS concept of a solar rocket. The mirror is much smaller, partly by artistic fiat and partly due to a commitment to lower thrusts. Not mentioned are the acrobatics required of the mirror to acquire the sun at various directions of travel while staying out of the way of other spacecraft structures, not to mention the rocket plume. This problem of "vehicle integration" was much feared and considered sufficient grounds for the decision that was made.

Nevertheless the solar rocket concept is once again an active area of consideration, and its history must include some recent footnotes. During FY 78 the AFRPL funded a contract with the Space Systems Group of Rockwell International (Ref 3) to reanalyze the value of solar thermal propulsion. There were two main reasons for the renewed interest. The first was the emergence of the Space Shuttle, which was unknown and unplanned in 1963. The Space Shuttle represents a national commitment to extended operations in space. It invites forward thinking in other areas, such as ambitious missions and applications that are made possible, and more potent rockets for propelling spacecraft beyond the reach of the Shuttle. The second reason was the realization that the "vehicle integration" problem had never been specifically addressed by people with spacecraft design experience, and that it was premature to regard it as a fatal flaw. The Rockwell contract then had as a major objective the task of devising alternative concepts for stowing and deploying large mirrors, and arranging their mountings and movements to eliminate interferences with the rocket plume and other structures, while preventing changes in the vehicle center of gravity from occurring due to motions of these oversized structures.

The design which emerged as the best solution to these problems is shown in Figure 4. The innovation which reconciles so many of these apparently conflicting requirements is the use of "off-axis" parabolic surfaces, which allow the mirrors to remain outboard from the vehicle. The mirrors appear to be canted or angled between the direction to the sun and the direction to the thruster, and this is sufficient for a first description. This is in contrast to the usual arrangement in which the target is on a line between the mirror and the sun, which severely constrains the geometry of the vehicle. Here the mirrors are not required to orbit around the spacecraft as the direction to the sun changes. Rather they rotate around the line between the target and the center of the mirror. This enables them to track the sun in any direction within a plane perpendicular to this line. If the sun lies in some different plane, then the entire spacecraft must be rolled around its long axis to place the sun in this plane. In this way the sun can be acquired in any position, and without having to move the mirror into the plume or some other awkward place.

(IF)

Since there are two equal mirrors on opposite sides of the spacecraft, and since each has its center of mass on its rotation axis there is no change in the vehicle center of gravity during any of these maneuvers. This rather long description needs an additional bit of clarification to satisfy the optical purists. It is well known optical fact that the object and the image must lie within a small angle of the optical axis of a lens or mirror, or the image will be severely distorted. One reading of the above description may suggest that the objective (sun) and image are each 45 degrees away from the optical axis of the mirror, leading to great distortion and loss of concentration ratio. This is not the case. The mirrors are not just tilted symmetrical paraboloids. Rather each is a patch of surface taken from the wall of a deep paraboloid of revolution enveloping the vehicle (Figure5). More particularly the patches are centered around the latus rectum of the parabola. The thruster is exactly at the focal point of this large paraboloid, and it should be evident that the paraboloid can be pointed in any direction like a searchlight without moving the patches from their outboard positions. The "off-axis" notation means that the optical axis of the mirror is not centered on the physical surface. In an optical sense the system is exactly "on-axis". The design seems fully responsive to the "vehicle integration" criticisms levelled at earlier solar thermal models, and invites a continuation.

The remainder of the Rockwell study strongly resembled the earlier EOS study. It involved determining the masses and scaling functions for the various components of a solar propulsion system, the synthesis of a conceptual vehicle and the optimization of that vehicle for various missions, which were themselves similar to the EOS missions. Comparisons were again made with chemical propulsion and electric propulsion. The motivation for re-doing this task was to catch up with fifteen years of technology. Areas of update included subsequent mirror work, improvements in the competition ($LOX-LH_2$ in particular), and a better knowledge of the micrometeroid hazard. Micrometeoroids were a big concern in 1963 since the fluxes and energies of these particles were not well known, and the designs for shielding against them were not well tested. Solar rockets are particularly vulnerable because their large hydrogen tanks makes such good targets. At the larger tank sizes considered by EOS the weight of the required micrometeoroid bumper exceeded the weight of the rest of the LH_2 storage tank and insulation--weight terms that came directly out of the payload. The hazard was better defined in the late sixties and early seventies by the Pegasus series of satellites and determined to be much less than earlier estimates. This was a favorable result for solar rockets, since they suffered disproportionately compared with their competition. On the other hand, the LOX/LH_2 chemical system gained a large increase in Isp from 400 to 475 seconds due to higher chamber pressures and area ratio. As discussed later there was also motivation to change the construction of the concentrating mirror from metal to inflatable.

The net result of these updates and others was to shift the balance in favor of chemical systems, and in fact to eliminate any payload advantage of the solar rocket. When this disappointing result was obtained it became necessary to make an important compromise in order to salvage a worthwhile competitive edge for solar rockets. That compromise was in longer trip time for the mission. Such a compromise, of course, sharpens the issues between solar rockets and ion rockets, which form the other boundary of the competitive domain of solar rockets.

The mission analyzed was the same as studied by EOS--low earth orbit (LEO) to geosynchronous orbit (GEO)--except that the vehicle size was changed. The common denominator was the Shuttle cargo bay, and the game was to maximize the payload to geosynchronous orbit when the payload and its propulsion were forced to fit into those weight and volume constraints. On any orbit-raising mission there is a difference in the so-called gravity losses between the nearly impulsive, perigee-apogee maneuvers of a high thrust system and the outward spiralling mode of a low thrust, continuously thrusting system. For LEO to GEO maneuvers the difference amounts to 5000 feet/sec, which is a large penalty levied against solar and ion propulsion. The very high Isp of ion engines allows them to overcome this added burden. Not so for solar rockets.

The advantages of perigee propulsion in reducing ΔV requirements are accessible to a low thrust system, but only by shutting down the thruster during the greatest part of the orbit and building up the total impulse (which appears as increased apogee) over many passes at perigee, ergo--a trade-off between payload and trip time.

Table I, reproduced from the Rockwell final report, gives the latest comparison between solar-thermal propulsion, chemical propulsion, and ion propulsion. As indicated earlier a 14-day trip time yielded no gain in payload over a chemical system, while relaxing the trip time to 40 days gives about a 40% payload advantage to solar. The ion engine has the greatest payload, but a 4-5 times greater trip time.

TABLE I PROPULSION SYSTEM COMPARISONS

	LO_2-LH_2	ION	SOLAR 1	SOLAR 2
ΔV, ft/sec	14,000	19,200	19,200	15,750
Trip Time	5 hrs	180 days	14 days	40 days
Isp, Sec	475	2,940	872	872
Mass Fraction	0.90	0.68	0.85	0.85
Payload, lbs	20,400	44,000	20,500	29,000

A long range solar propulsion planner must hang his case on a two part argument. The first is that a mission application will develop at some future date which needs the payload advantage of solar rockets; and the second is that the mission will tolerate a 40 day trip, but not a six month maneuver for which an ion approach is indicated. Providing the mission or missions are ambitious enough, then a potential cost savings over an extended period will pay for the large developmental costs of a new system. In general such large projects are difficult to foresee many years in advance, and the technologist with money to spend must be content with shorter range indicators, such as existing missions or missions already on the planning boards, for which a propulsion approach has already been selected. He can then appeal to retrofit arguments or second generation missions of like kind. There is little to record about this process in the solar context except to say that the objective analysis of the Rockwell program has successfully passed through the more subjective assessment and decision making; and that we can now append a present and a planned future to the history that has been presented.

CURRENT AFRPL SOLAR ROCKET EFFORTS

The on-going AFRPL solar rocket program is connected with developing and demonstrating the technology of solar powered thrusters. Basically we are buying the engineering and manufacturing of a subscale thruster and assembling in-house the capability to test it.

Under Contract F04611-80-C-0039, Rocketdyne is designing and building a nominal ½ lb thruster for delivery. We gave them a number of engineering tasks, including the analysis of several thruster concepts. From this analysis a best candidate was selected for detailed design and fabrication. This analysis took into account estimated Isp performance, endurance, and the level of available supporting technology.

The Isp that can be achieved using hydrogen as a working fluid is shown in Figure 6 as a function of chamber temperature. Corrections have been made for the rate-limited recombination of H atoms in the flow and for the boundary layer nozzle losses, so the graph is particularized for the indicated thrust level of 20 pounds. The Isp levels assumed by Rockwell in their applications study (872 sec) imply a temperature of about 2800°K or 5000°R. Since hydrogen is transparent to solar radiation, some solid surface must survive this temperature or greater in order to absorb the solar flux and pass it on to the working fluid. The success of solar rockets is thus vested in the refractory capabilities of just a few metals or possibly metal carbides. Among the metals, only tungsten and rhenium are serious candidates, while hafnium carbide or tantalum carbide may be useful in some specialized designs that do not require the mechanical and fabricability characteristics of metals. Carbon would be acceptable only in structures that do not allow it to contact the hydrogen, since it reacts rapidly to form acetylene and C_2H radical at elevated temperature.

In the current contract, Rocketdyne was given five solar thruster concepts to analyze before selecting their recommended design. These are represented schematically in Figures 7-11. Four of the designs involve the use of physical windows to admit the solar flux to the interior of the thruster. The advantage of this is that whatever solid surface is selected to absorb the radiation is not simultaneously required to form the walls of a pressure vessel, as is the case with the windowless heat exchanger cavity. All other things being equal windowed designs should be capable of higher temperatures or increased lifetimes over windowless designs. To be sure, all other things are not equal, so the choice is complex.

The most interesting windowed approaches are those with refractory particulate matter to absorb the sunlight and act as a high surface area heat exchanger. In one version (Figure 9), the particles are injected into the flow and exhausted from the thruster. The main concerns here are window and concentrator contamination, although there is obviously an Isp penalty due to the increase in molecular weight associated with the particulates. In Figure 9 a vortex concept is illustrated, in which particulates are again injected into the flow, but are retained in the chamber by swirling them away from the centerline and withdrawing the gases from the clean middle flow. This potentially solves the contamination problem and restores Isp; but the approach involves a delicate aerodynamic balancing act, in that the particles must be suspended by the inward blowing of hydrogen acting against the centrifugal force of the vortex. Finally in Figure 10 a rotating bed concept is illustrated which is similar to the vortex version, but offers a mechanical solution to the aerodynamic problem. It consists of a rotating porous cylinder overlaid with a bed of refractory particles, held down by centrifugal force. The particles and the countercurrent in-flow of H_2 form a dynamic insulation for the cylinder and its bearings and seals. Since there is virtually no mechanical load on the particles they can be heated to near their melting temperature.

Each of these concepts has been considered in detail under the concept analysis phase of the Rocketdyne contract, and the results have been presented to AFRPL. For present purposes I will confine my remarks to the bottom line. Rocketdyne found that the simple windowless design was capable of satisfying the Isp and durability requirements of the contract, and involved the least technological risks. It is the design that has been selected for detailed design and fabrication.

THE AFRPL SOLAR TEST FACILITY

The thruster is scheduled to be delivered in about a year. In the interim AFRPL will be developing a facility for measuring its Isp and efficiency and demonstrating its durability under test conditions. The key facility item is a six meter solar concentrating mirror, manufactured by the Ominium-G company in Anaheim. It has already been installed for use by another AFRPL project, and its use will be shared whenthe solar thruster project comes on line. The rest of the facility remains in the planning stage, but will consist of a hydrogen storage and flow system, and the instrumentation for measuring and recording propellant flow rate, chamber pressure, pyrometric measurement of chamber temperature, solar flux, and thrust.

There are a number of complications associated with the job of measuring the performance of subscale solar thrusters, particularly when it is desired to make thrust measurements. The Omnium-G mirror is a sun-tracking device with limited weight carrying ability at the focal plane. Thrust stands are typically stiff and massive. Beyond that they work best when planted firmly to the earth, and not on a platform that moves with a solar image. The commitment to measure thrust, thus was a commitment to lock the concentrator in place and add a sun-tracking heliostat to the system. So we are presently adding this capability.

A drawing of the proposed test article and thrust stand is shown in Figure 12. The thruster is represented as a coil of refractory tubing wrapped into the shape of a cavity through which H_2 is pumped. This is generally the plan of the thruster selected under the Rocketdyne contract. This design obliges us to provide a high quality vacuum around the thruster to eliminate large convection losses, not to mention the possible rapid oxidization of the coils in air. A set of radiation shields will also be needed between the thruster and the vacuum canister. It goes without saying that the canister will need a window to admit sunlight, so part of the complexity that was avoided with the choice of thruster has been shifted to the facility. Since the thruster is sealed to the canister at the nozzle exit it is necessary to mount the canister on the force balance and measure the thrust of the whole unit. Small thrusts always present a measurement problem, and to leave the test article open to our desert winds would make it impossible to make a meaningful measurement at the ½ pound level. So another layer must be added--a wind enclosure with yet another window. The drawing shows a perforated wall to screen out the wind while letting out exhaust gases. More recent planning has the wind enclosure being evacuated (perhaps by the thruster exhaust) so that a high area ratio nozzle can be employed for altitude studies. The thrust balance must fit inside the wind enclosure, and the dimensions of the entire thrust facility must be minimized to avoid excessive blockage of sunlight to the concentrator. A design goal is to maintain the envelope within a diameter of three feet, since the concentrator has a hole in the middle of that size.

LONG RANGE PLANS

Approximately two years of development and testing will be occupied in completing the solar thruster feasibility studies described above as on-going. Longer range plans include a mix of scale-up and alternate technology development. A three-to-four-fold scale-up in thrust level can be achieved with high concentration ratio (> 10,000 = 1) mirrors already available in the US, and a forty-fold increase in thrust might be obtained at the CNRS one megawatt facility that has been in operation in France since the early 1970s. Although a commitment to such an ambitious scale has not been made, it is significant that this would test thrusters to near the maximum levels found useful in the Rockwell applications study.

Technology development will emphasize alternate thruster concepts, propellants, and lightweight mirrors.

Thruster Technology

The windowless design selected under the concept phase of the Rocketdyne contract is the most straight-forward concept within existing technology. Higher performance will require higher temperatures than this design can endure. Beyond this, a back-up thruster approach is desirable in case the windowless design falls short of expectations. Alternate designs will be studied as a part of our in-house effort. These studies will emphasize the rotating bed thruster shown in Figure 11. Technology areas include windows for passing the solar flux, refractory materials for the bed material, heat transfer models for fixed and fluidized beds, hardware cooling strategies, high temperature bearings and seals, rotation techniques, and start-up and shut-down methods.

The design places very little structural demand upon the particles of the bed, where the greatest temperatures are achieved. This allows greater latitude in materials selection. The most attractive materials for the bed are hafnium carbide and tantalum carbide, which melt well over 400°C higher than tungsten. An issue is whether the large temperature difference necessary between the hot interior of the bed and the porous liner and its seals and bearings can be sustained by the dynamic insulation qualities of the bed. Another issue is whether the carbon in the metal carbide will be reacted away to form gaseous carbon species, leaving behind lower melting metal particles. The higher temperature accessible with this concept will place greater thermal loads upon the rest of the hardware, in particular the windows. The optimum window material appears to be quartz. But there are a number of strategies which might be used to upgrade its capabilities. Specially purified quartz may be helpful in reducing its absorptions in the visible. Coatings which selectively reflect the infrared radiations coming from the chamber may reduce the thermal load while increasing the thruster efficiency. Finally it may be necessary to incorporate cooling passages in the window and pass a transparent and high refraction index coolant through them.

Alternate Propellants

From a strict performance standpoint, hydrogen is greatly superior to any other. Indeed only a handful of working fluids can offer an advantage over LOX/LH_2 when heated by solar energy. The interest in alternate propellants is motivated by the same arguments that are advanced to promote storable propellants over the more energetic cryogenics. And the baseline for comparison is more appropriately some storable bipropellant, such as N_2O_4/MMH, rather than LOX/LH_2. Applications may occur in which propellant storage over extended periods of time is needed, and in which the boil-off losses associated with H_2 are not acceptable.

Even with a lower baseline the storable solar rockets will need to operate at higher temperatures than the H_2 storable in order to be competitive. This is because the molecular weight advantage of storable working fluids over conventional propellants is not nearly so great as pure H_2 gives over LOX/H_2. So the higher temperatures offered by advanced thruster concepts such as the rotating bed device may be relatively more important in the storable context. The equilibrium Isp of various fluids vs temperature is shown in Figure 13, up to the melting point of hafnium carbide.

Although several hydrides are higher in performance, ammonia is the highest performing storable that lends itself to convenient operation. The remaining candidates produce condensed or condensible species, and some are normal solids and difficult to deliver to the chamber. The Isp of ammonia varies from about 500 sec to about 700 sec in the potentially accessible temperature range.

Another potential application of solar energy is through the enhancement of conventional bipropellants. The Isp gains are more modest, but the engineering problems are greatly reduced as well. This consists of pre-heating both a fuel and an oxidizer prior to injecting them into the chamber. The heating process need not be limited to simple increases in sensible heat, but may embrace endothermic chemical change, and in some cases to metastable conditions. The heating of N_2O_4, for example will yield a mixture of O_2 and NO. NO is a gas with a very endothermic heat of formation and a strong kinetic resistance to further disproportionation to less endothermic N_2 and O_2. For a fuel, it may be noted that NH_3 disproportionates to N_2 and H_2 at relatively modest temperatures, eliminating the negative heat of formation of NH_3. These endothermic features would add about 100 seconds of Isp to the N_2O_4/NH_3 bipropellant system, and would involve relatively low temperature absorbers and less demanding solar concentrator optics.

A final alternate propellant area would be tailoring of H_2 for better compatibility with carbon or metal carbides. A drawback of hafnium carbides or tantalum carbide in the rotating bed concept for example is the probable chemical scavenging of carbon by formation of high temperature gaseous carbon species such as acetylene

$$2\,HfC\,(c) + H_2 \longrightarrow Hf\,(metal) + C_2H_2$$

The reaction is reversible and can be prevented by providing the incoming propellant with a small amount of acetylene, or something that forms acetylene, during the heating process. Methane is the most promising additive for this purpose.

Mirror Technology

The progression of analytical studies from Ehricke, through EOS to Rockwell has had a profound impact upon our appreciation of the mirror problem. In 1956 it was sufficient to show an Isp of 450 seconds in a solar rocket in order to show a worthwhile advantage. A simple spherical mirror was adequate, even though it had to be very large. In 1963 an Isp of 800 seconds was needed and a parabolic shape was required. But the mirror size was more tractable. In 1979 the worst of both requirements came together; and the analysis showed that the mirror required was both large and accurate, and beyond this was required to be fabricated to a peculiar off-axis symmetry.

Partly these changes are progressions imposed by the need to keep ahead of an advancing competition from chemical systems. Partly they are oscillations caused by alternate choices between the two trajectories for orbit raising. Ehricke chose an efficient trajectory requiring high thrust. EOS chose an inefficient trajectory requiring high Isp. Rockwell showed that this degree of freedom does not exist. To be competitive the solar rocket must fly an efficient trajectory (high thrust) and still deliver high Isp.

These requirements are conflicting when viewed from the standpoint of the mirror. EOS demonstrated that a sufficiently accurate, lightweight mirror could be built at a five foot diameter, but they estimated that an order of magnitude loss in concentration ratio would occur in scaling up to collectors of 50 foot diameter.

The high-water mark of lightweight, large, high-concentration ratio mirrors occurred in the early and mid-1960s as a result of AF and NASA programs for providing high levels of electrical power in satellites. The AF ASTEC program (Reference 4) for a sun-powered turbo-alternator spawned accessory mirror studies, aimed at placing a 52 foot mirror in orbit for tests of storage, deployment and heat generation. The project was cancelled before this happened, but ground tests of the mirror produced by Goodyear were carried out by the Sundstrand Corporation on a 10 foot and 44.5 foot model, yielding maximum concentration ratios of 3900:1 and 3200:1 respectively. The mirrors were fabricated by laying up gores of one mil mylar on a paraboliodal tool, seaming the radial butt joints with tape, inflating to shape, and spraying the backside with a lacquer coat and polyurethane foam for rigidization. A satisfactory transposition of the foam-rigidization process from earth to space was an unsolved and highly respected problem - and one that might be unnecessary. It was noted that the optical quality of the mirror surface was far superior before the inflated shape was rigidized.

The Rockwell conclusion that large thrusts were necessary places great importance upon inflatable mirror technology. Their study assumed the use of inflatable mirrors and it is likely that their estimated mirror weights would have to be multiplied by a factor of 5 to 10 if honeycomb or foam rigidized designs became necessary. If the solar rocket concept passes the preliminary thruster feasibility demonstration a large mirror technology program will be required.

CONCLUSION

Analysis of the solar-thermal rocket concept continues to indicate high promise as a primary propulsion system, assuming trip times of 40 days or so are tolerable for orbit-raising applications. based on these analyses AFRPL has initiated a combined contractual and in-house effort to develop and evaluate a sub-scale solar thruster delivering a thrust of about ½ pound and an Isp of 800 seconds. A formidable technology must be developed in support of the long range practicality of the concept. This includes high temperature, high endurance thruster designs and very large, high concentration ratio, lightweight mirrors.

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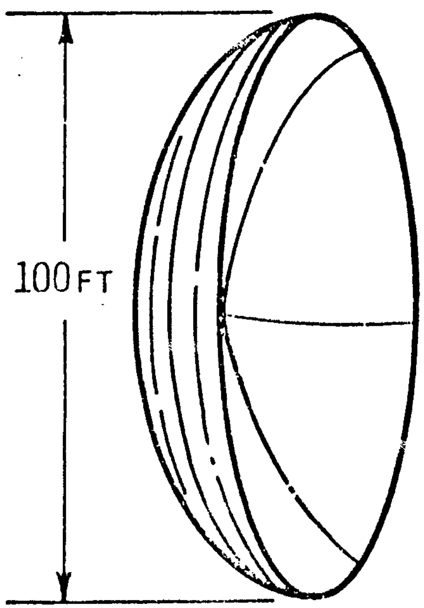

	THRUST (LB)	ISP (SEC)
SOLAR SAIL	.0005	00
SOLAR-ELECTRIC	1	3000
SOLAR-THERMAL	10	900

Figure 1. Comparison of Solar Propulsion Approaches

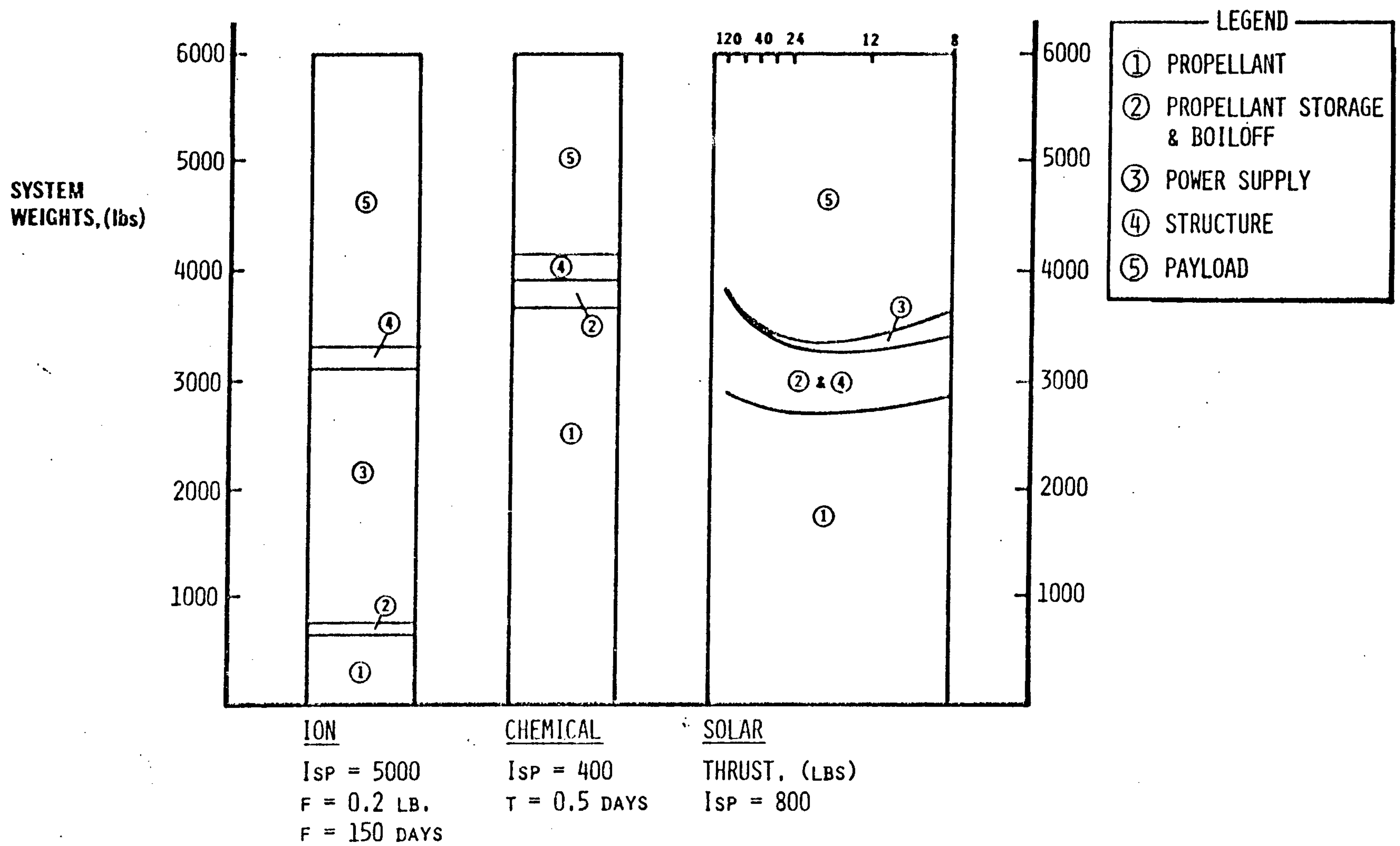

Figure 2. Propulsion System Performance Comparison for Orbital Transfer From 300 NM to 24 Hour Orbit for 6,000 lb Vehicle

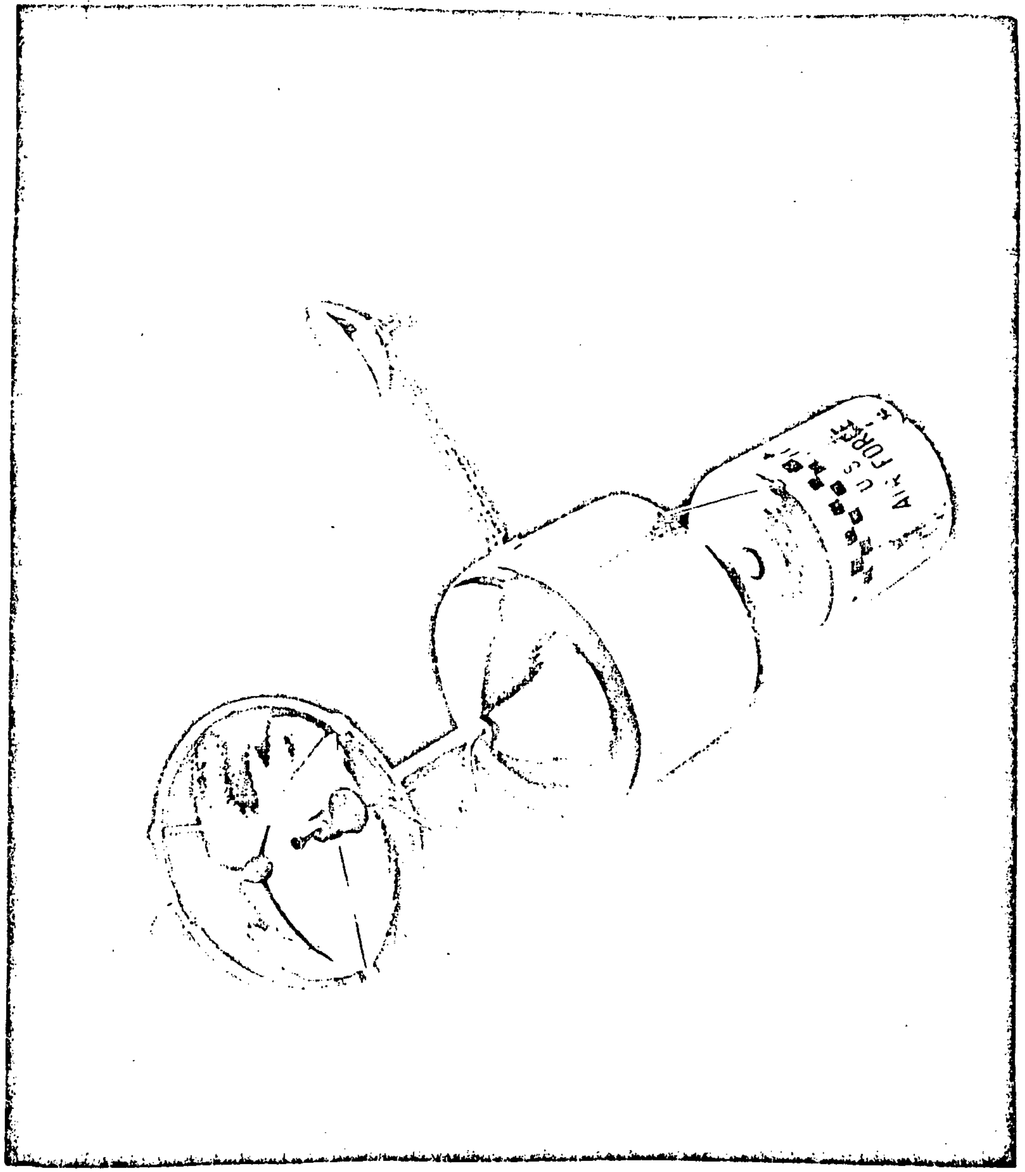

Figure 3. Artist's Concept of Sohr Propelled Spacecraft

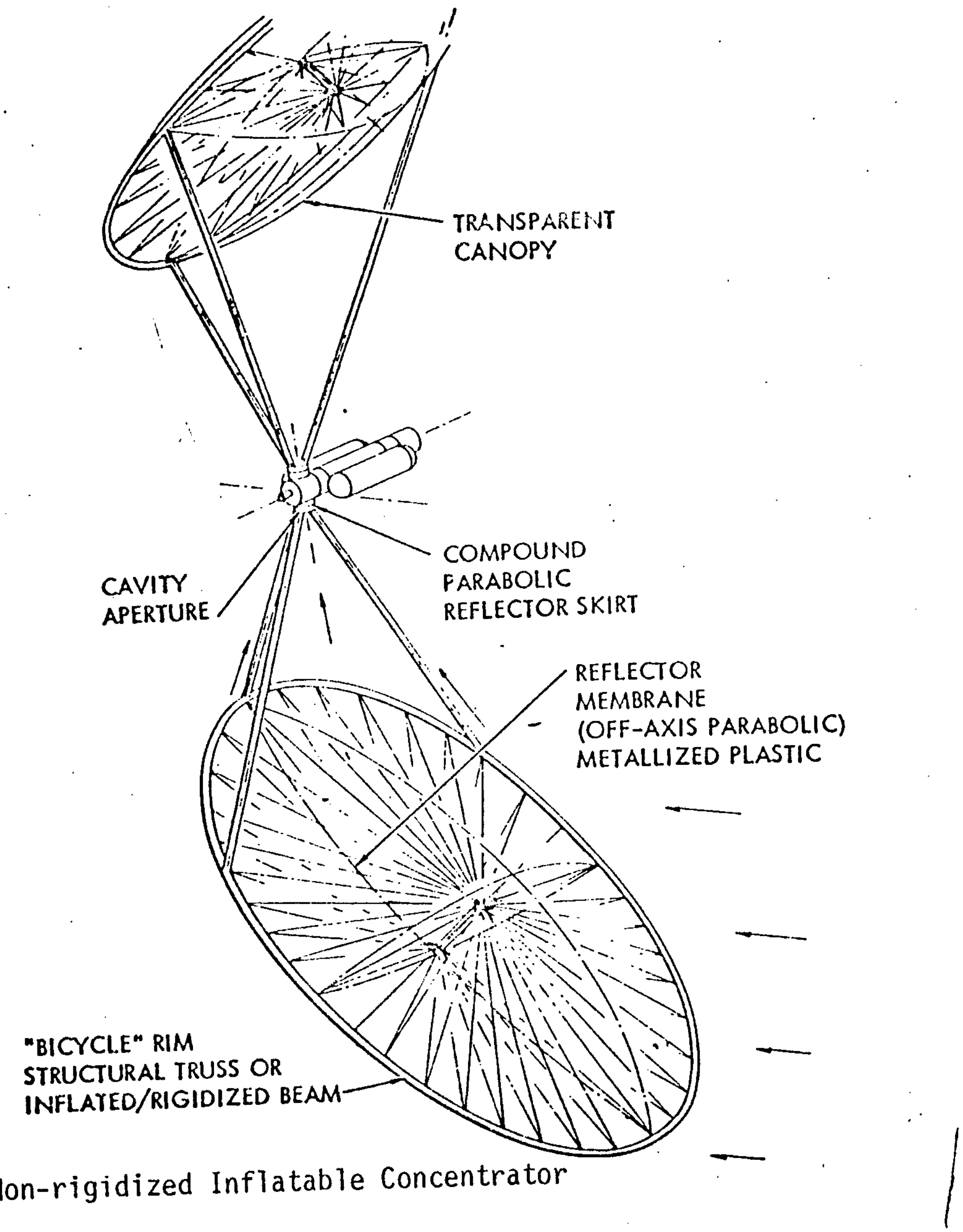

Figure 4. Non-rigidized Inflatable Concentrator

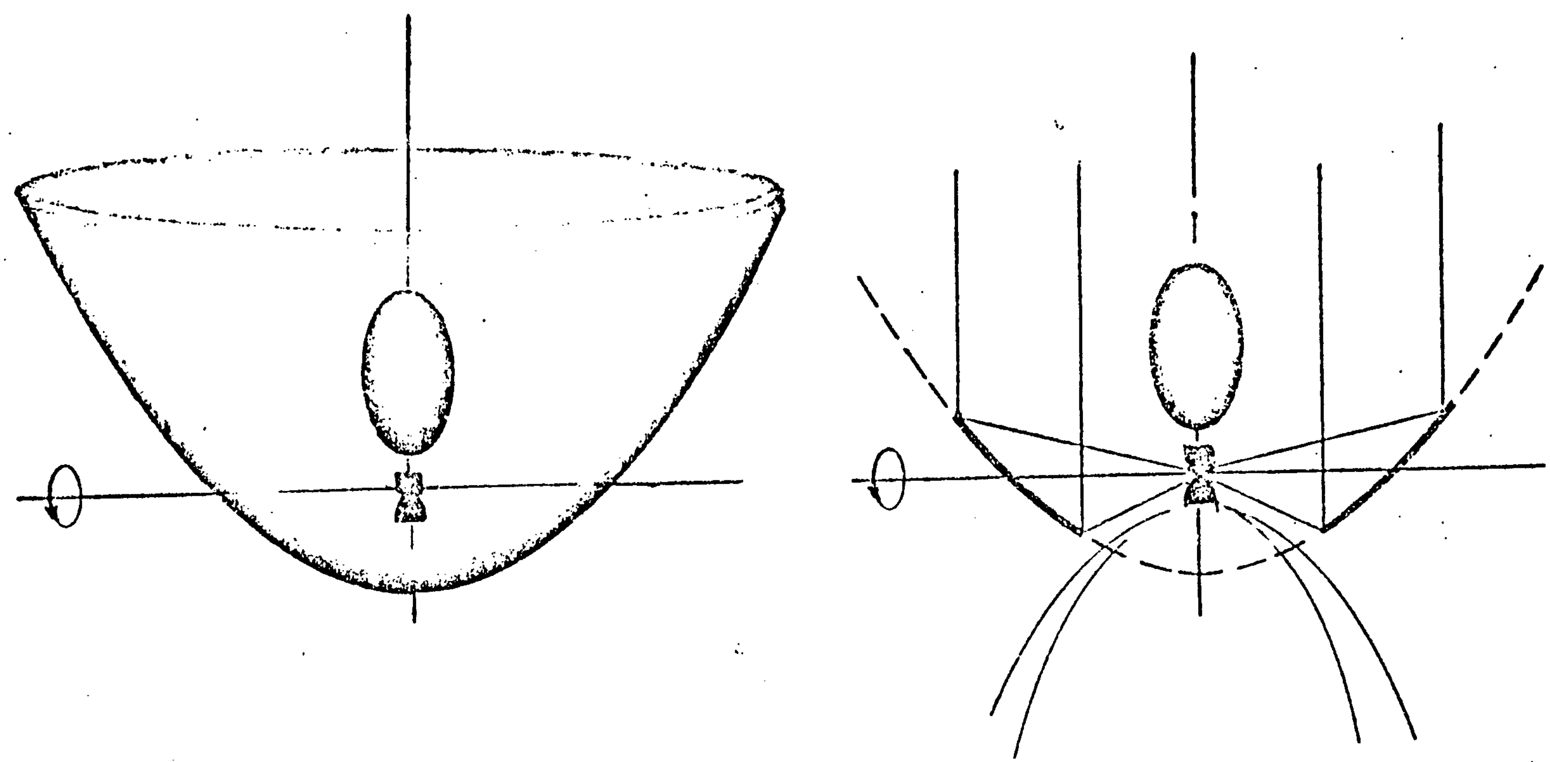

Figure 5. "Off-Axis" Parabola Concept

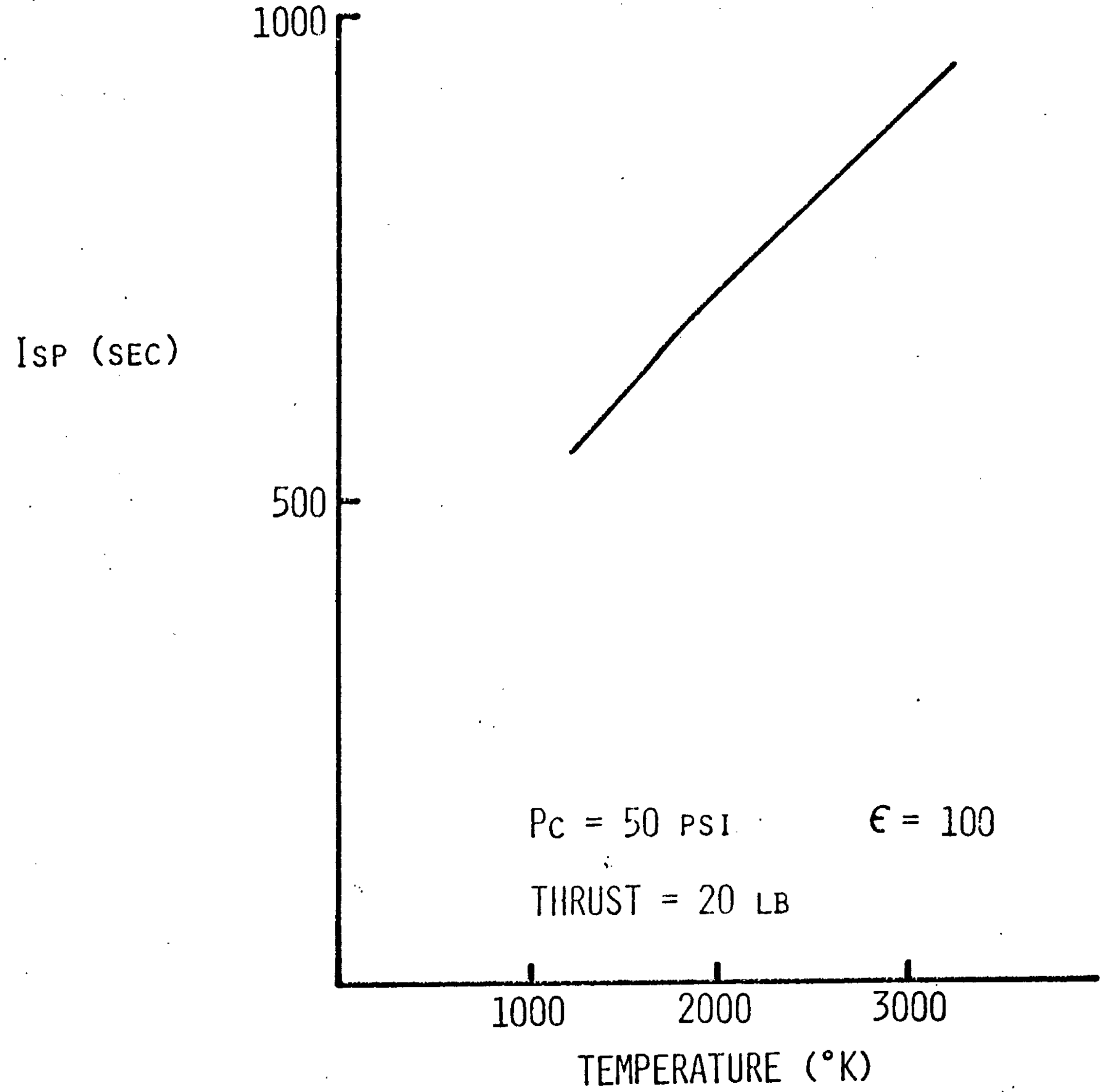

Figure 6. Estimated Delivered Isp for Heated Hydrogen

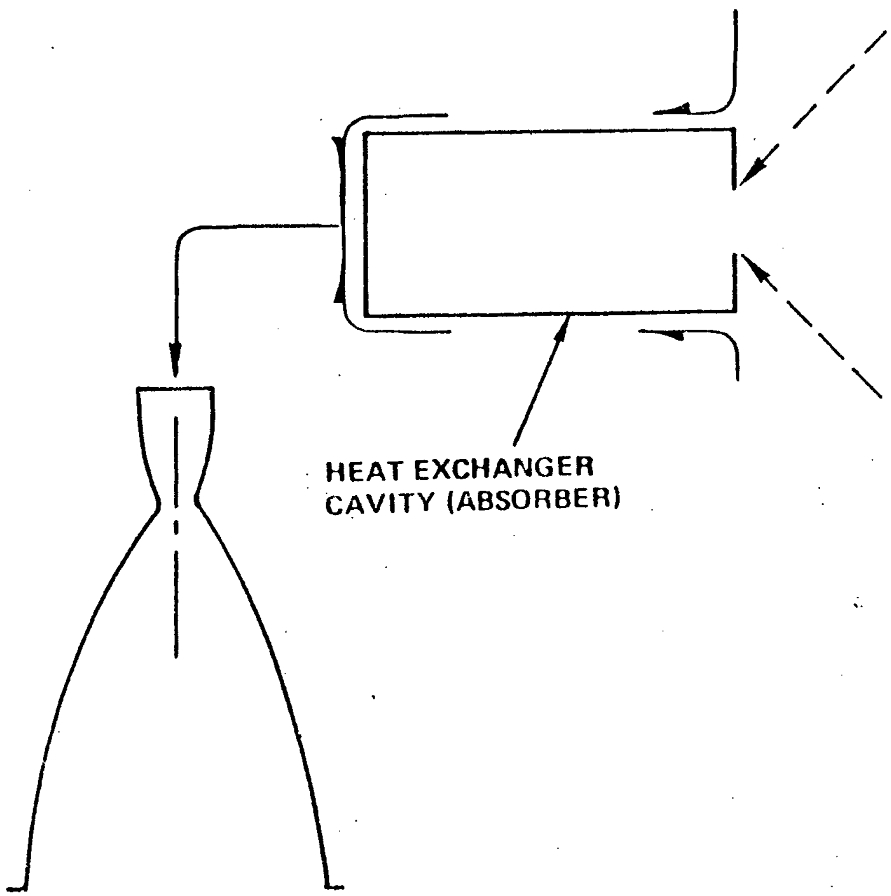

Figure 7. Heat Exchanger Cavity (Windowless)

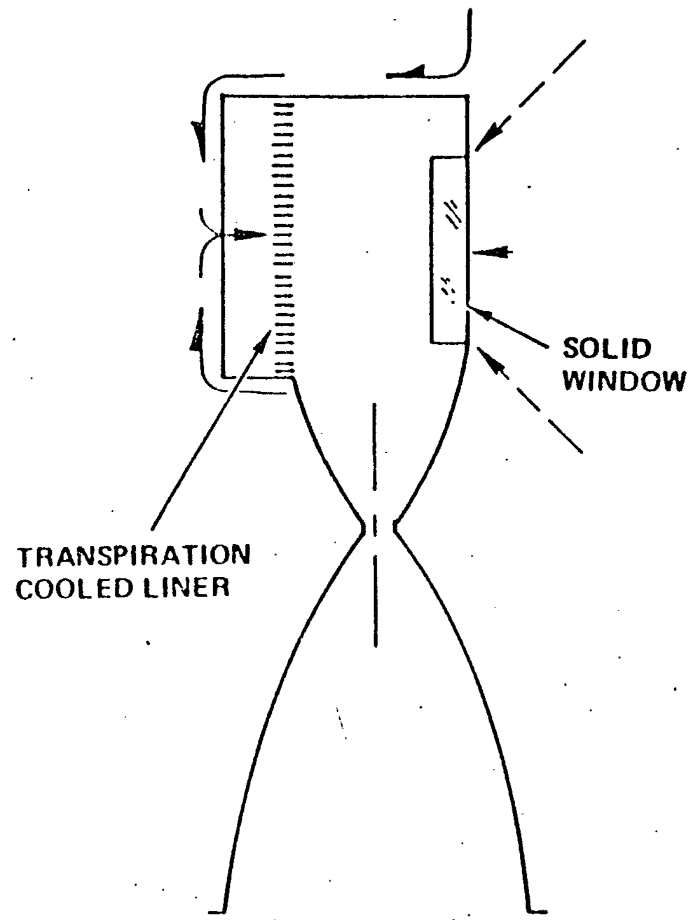

Figure 8. Windowed Heat Exchanger Cavity

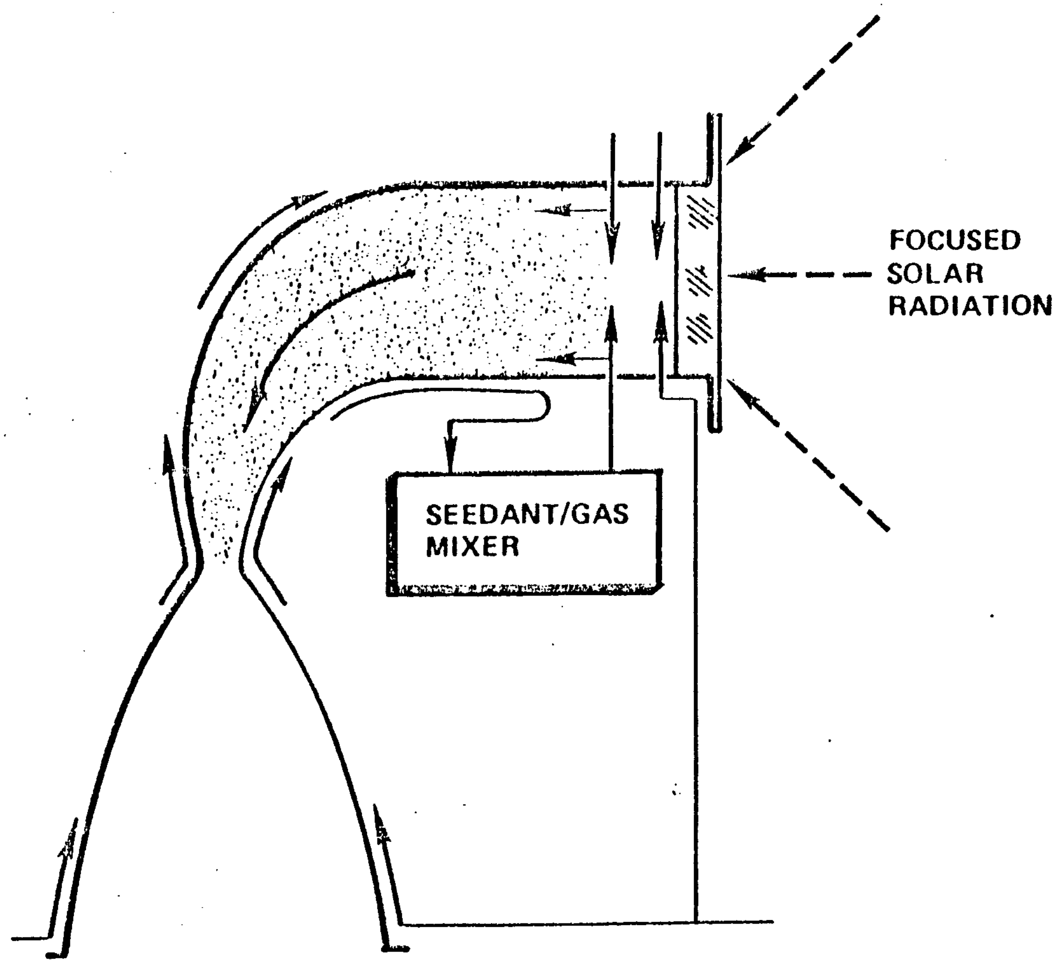

Figure 9. Windowed Molecular or Particulate Concept (Discharged Seed)

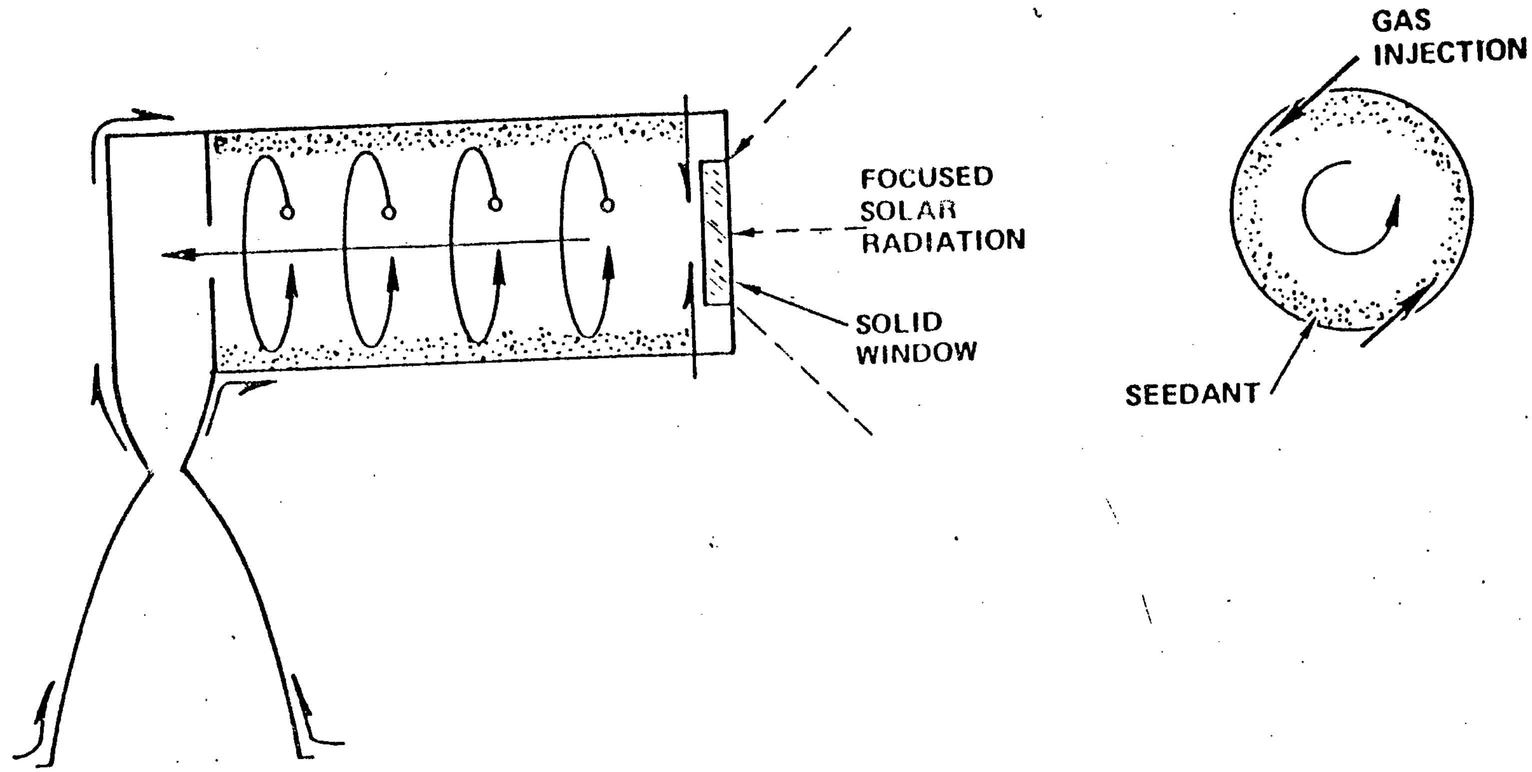

Figure 10. Windowed Vortex Flow Concept (Retained Seed)

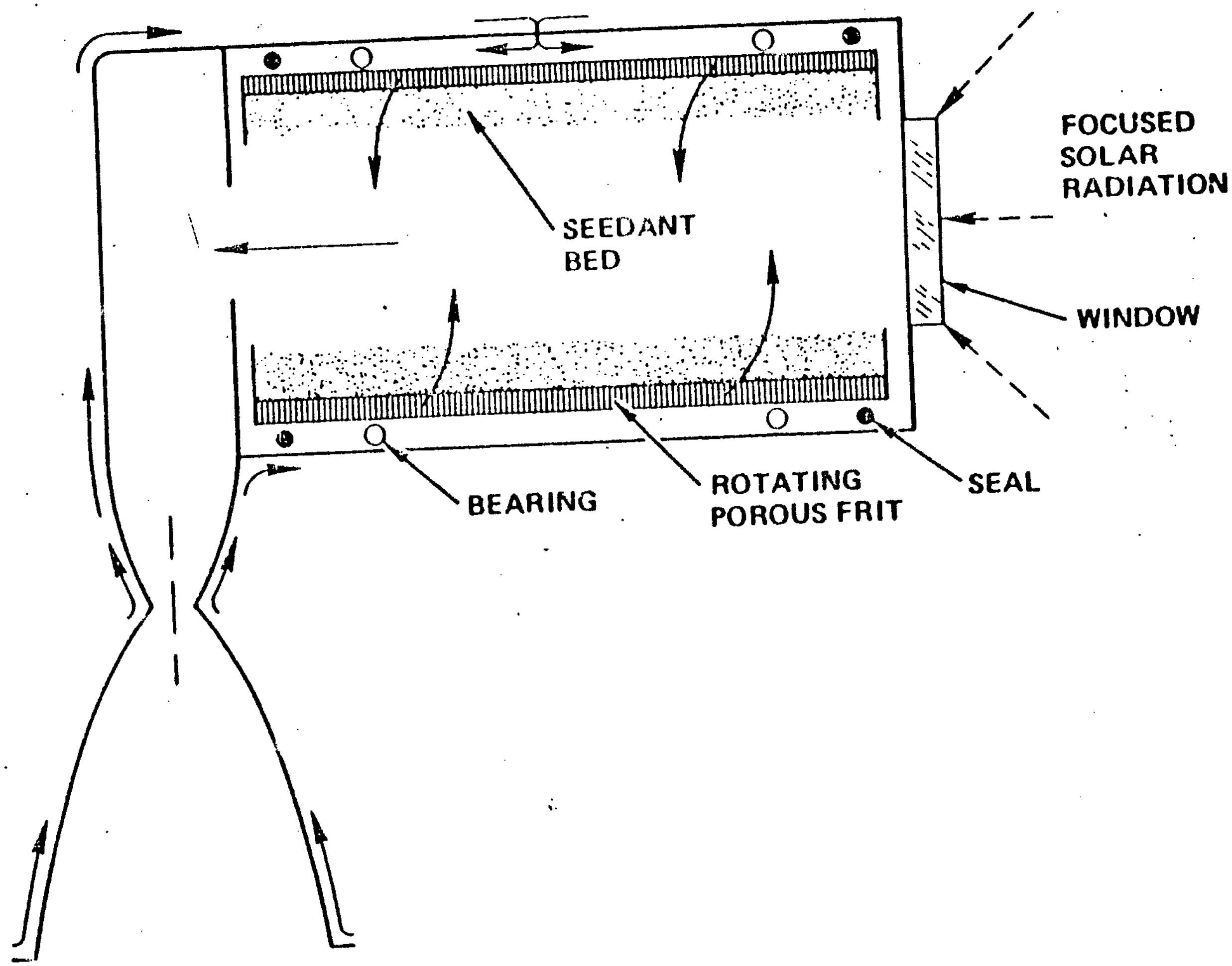

Figure 11. Rotating Bed Concept (Retained Seed)

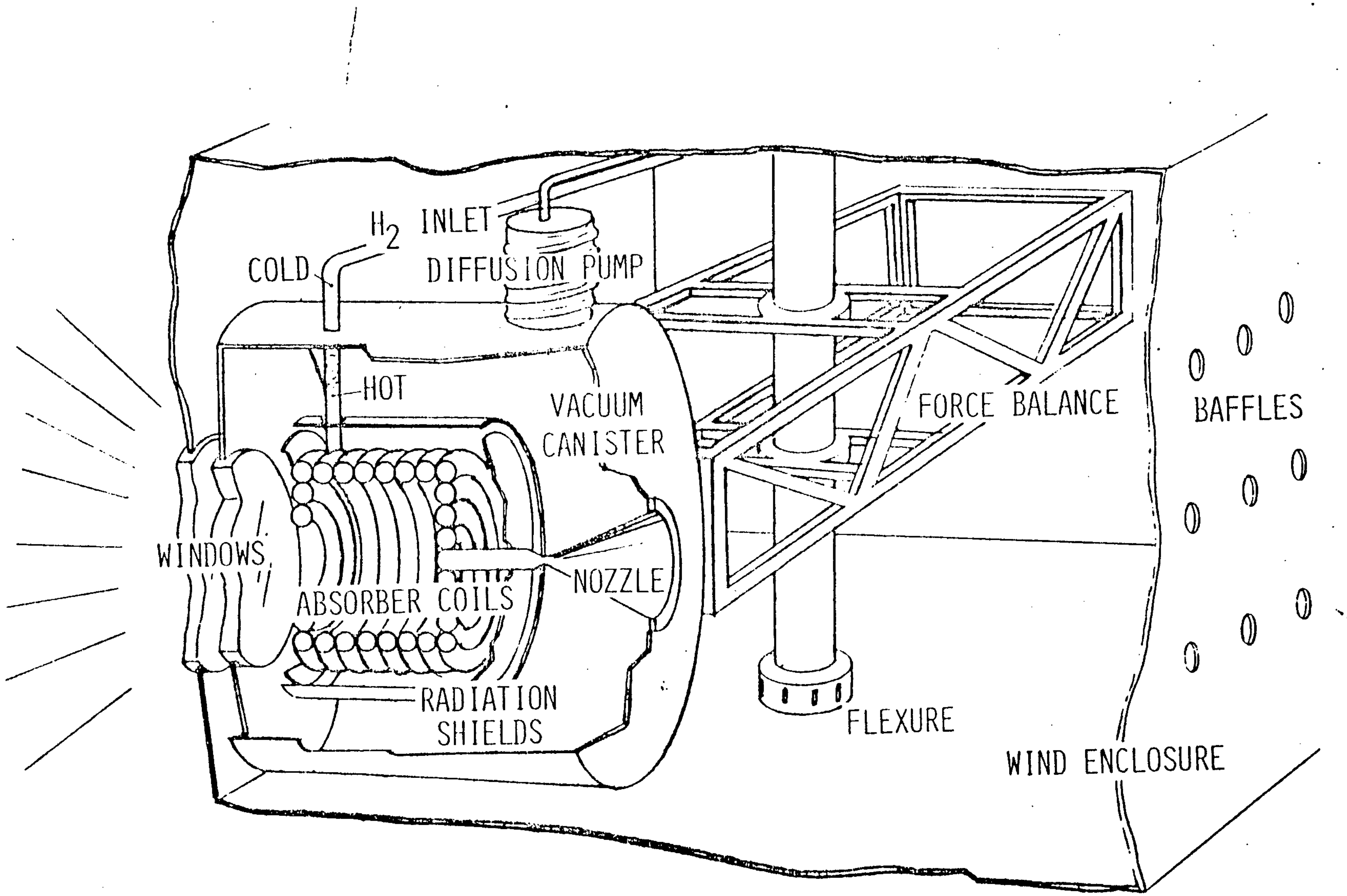

Figure 12. Solar Thrust Stand

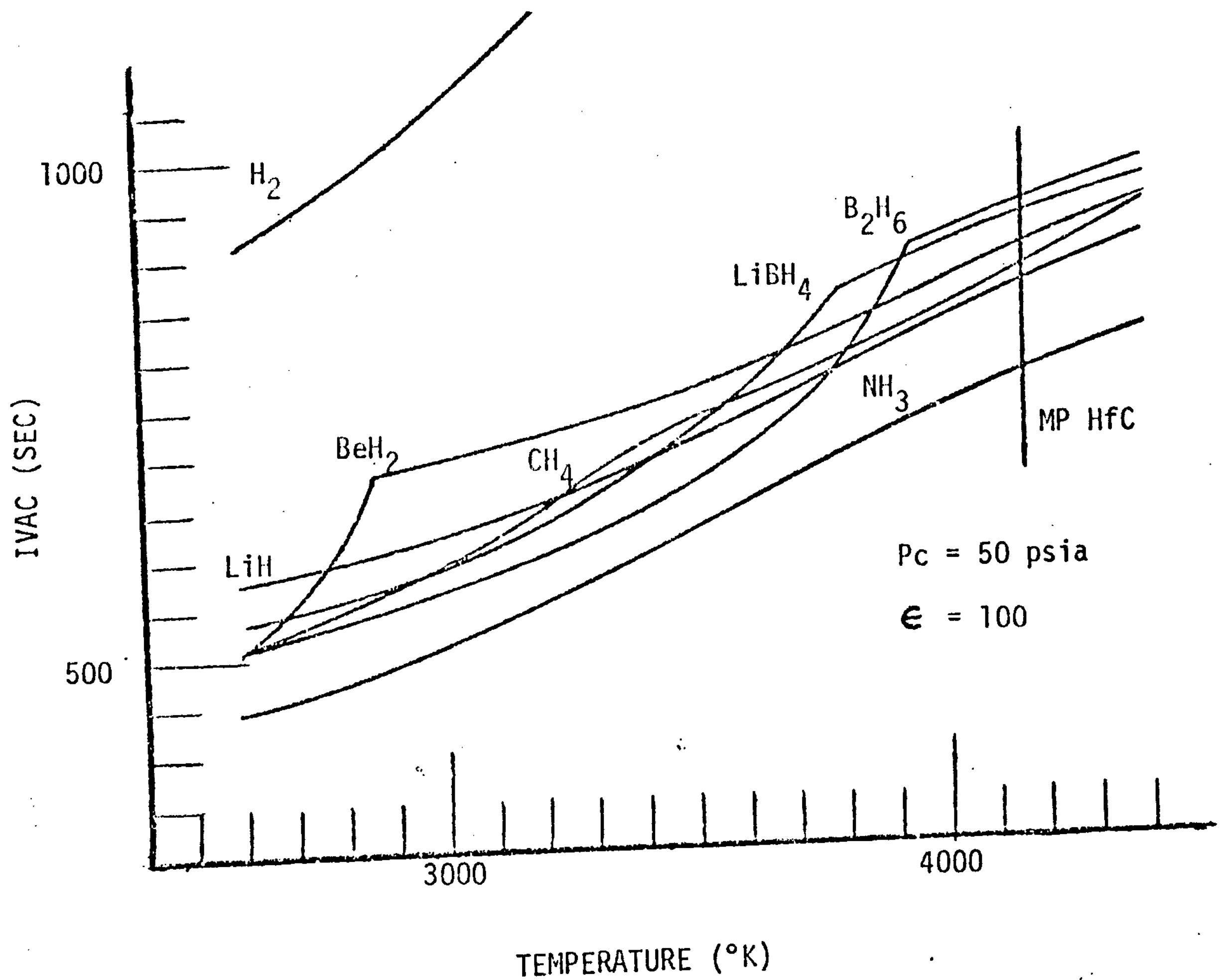

Figure 13. Equilibrium Vacuum Isp