Viking Mars Orbiter 1975 Solar Energy Controller

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The Viking Orbiter system is part of the overall Viking Project managed by the Viking Project Office at NASA Langley Research Center. Two spacecraft were launched on Titan IIIE/Centaur launch vehicles in August and September 1975. Solar energy for temperature-control purposes is introduced into the propulsion module of the Orbiter. The energy enters by way of four individually commandable solar energy controllers (SEC's). This paper summarizes the SEC thermal development, design, test, and flight results.

Introduction

THE idea of using solar energy directly for temperature-control purposes was considered on the Voyager Project, a forerunner program having objectives similar to those of the Viking Project. Generally speaking, as the spacecraft grows in size, so do the energy requirements for temperature-control purposes. The Viking Mars Orbiter 1975 (VO'75) is more than twice the size of the thermally similar Mariner 1971 spacecraft and the first to control the use of solar energy directly for temperature-control purposes. Figure 1 illustrates how the SEC's introduce the solar energy into the propulsion module (P/M).

Basic Requirements

Without heating, the thermal losses from the P/M at Mars are sufficient to take the P/M and bus below the allowable temperature limits.(The bus is that part of the Orbiter where most of the electronics are housed and, as a result, where the majority of the energy is dissipated. It surrounds a portion of the P/M, as shown in Fig. 1, and both the bus and the P/M are thermally very influencing boundaries to each other.) Early studies had indicated that this result would occur because of a low bus power density and uncertain thermal blanket losses. (Because of its size, 125 ft^2, the P/M blanket consists of three sections. Energy losses by way of the resulting seams potentially could be excessive. The other primary blanket for the P/M, referred to as the bus blanket in Fig. 1, is approximately 50 ft^2 but is seamless). For an effective blanket emittance of 0.009, studies indicated that 60 W would be necessary to maintain the P/M and bus at the desired temperature levels.

An oxidizer-to-fuel bulk allowable temperature difference requirement of 10°F and allowance for a single SEC failure made a four-unit configuration desirable. Four units would lessen the amount of energy per unit area impinging on the very thin titanium propellant tanks. Additional diffusing of the energy was necessary to avoid local tank hot spots. To minimize hardware, it was desirable that some of the dispersion of energy take place at the SEC. The surfaces on which the energy first impinged after leaving the SEC would be given a high solar reflectance (painted white) to diffuse the energy further. Because of a possible 3.6-hr solar occultation at Mars, energy losses by way of the SEC's could not be excessive. From these basic requirements, a commandable shuttered greenhouse device having a useful energy capture at Mars of 20 to 25 W was pursued.

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Development of the Design

A greenhouse-type solar device was built to evaluate several candidate window materials.[1] This unit had all of the significant thermal characteristics of the flight unit except that the shutter and reflector were combined. As the Viking configuration became firm, so did the SEC configuration. The prototype unit was evaluated thermally in the Jet Propulsion Laboratory (JPL) 25-ft space simulator separate from, but in a parallel mode to, the Mariner 1971 spacecraft. Shake tests followed and demonstrated the feasibility of using glass (quartz, 0.060 in. thick) as a window material. However, the energy capture requirements had been reduced to 15 to 20 W per SEC because the bus power density was higher than originally assumed. Rather than change the size of an acceptable design, the glass window was removed. Removal of the glass eliminated an outgassing contamination concern (condensing of the outgassing constituents on the glass), provided additional venting area for the spacecraft, and reduced the weight of the SEC by 10% while increasing the infrared energy losses by an acceptable amount.

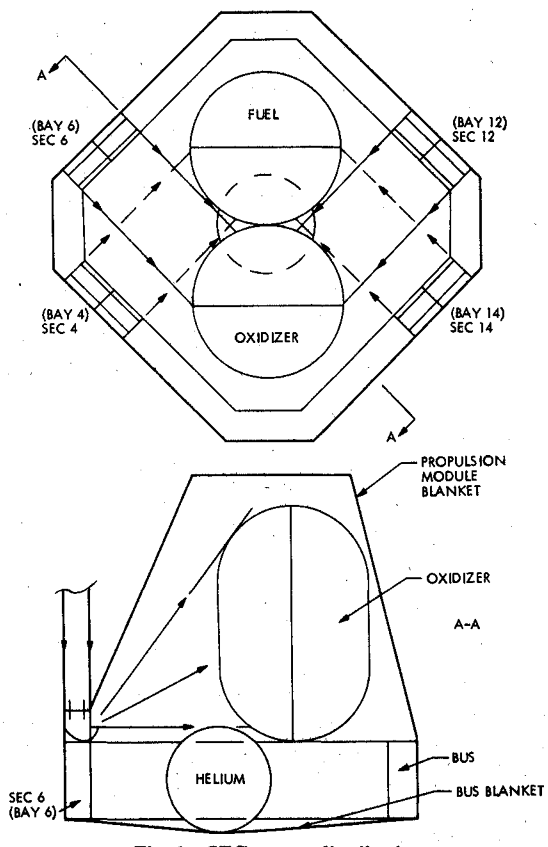

Fig. 1 SEC energy distribution.

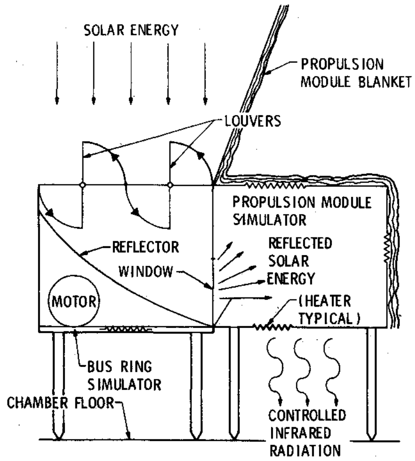

Fig. 2 SEC calibration unit thermal model.

An additional change, which appears to be an extension of the reflector in Fig. 2 also was incorporated into the design at the time of the removal of the quartz window. One of the late major configuration changes (relative to the SEC's) was moving the propellant tanks from a partially submerged position within the bus to a point outside the bus. With the tanks raised, some of the downward-reflecting SEC energy impinged directly on the bus bays. This was undesirable, and the curvature of the reflector was changed slightly so that the emerging rays were horizontal and/or upward. The extension covers that portion of the opening previously utilized by the downward-directed energy and significantly reduces the infrared energy losses by way of the SEC.

The SEC weighs 2.76 lb and has exterior dimensions of $6.4 \times 6.6 \times 10.3$ in. Except for the louvers, the portions of the SEC directly and indirectly exposed to solar energy have the optical properties of polished 6061 T6 aluminum alloy. The sun side of the louvers is anodized to lower the louver temperatures when closed (normal to the sun rays) near Earth. Also, the reflector receives additional polishing, since it is required to have a specular solar reflectance of 0.79 or better.

Discussion

SEC Thermal Model

A better understanding of the SEC calibration test results is obtained if a discussion of the energy balance on the P/M simulator (Fig. 2) is presented first.

This energy balance is written symbolically as

$$Q_{\text{solar},\phi} + P_\phi - Q_{W,\phi} - Q_{R,\phi} - Q_{C,\phi} = 0 \tag{1}$$

where

ϕ = opening angle of the SEC louvers (0° = closed, 90° = full open)
$Q_{\text{solar},\phi}$ = total solar energy captured
P_ϕ = heater power supplied to maintain the P/M simulator at a specified temperature T
Q_W = infrared energy radiated from the P/M by way of the SEC
Q_R = infrared energy purposely radiated to the surroundings
Q_C = energy conducted across the interface between the SEC and module

In the full-closed position ($\phi = 0$)

$$Q_{\text{solar},0} + P_0 - Q_{W,0} - Q_{R,0} - Q_{C,0} = 0 \tag{2}$$

Subtracting Eq. (2) from (1) results in

$$Q_{\text{solar},\phi} - Q_{W,\phi} = Q_{\text{solar},0} - Q_{W,0} + (P_0 - P_\phi) + (Q_{R,\phi} - Q_{R,0}) + (Q_{C,\phi} - Q_{C,0}) \tag{3}$$

Purposely, the P/M simulator is maintained at the same temperature throughout all measurements for a specific solar intensity. Therefore

$$Q_{R,\phi} - Q_{R,0} = 0$$

A gap of a few thousandths of an inch exists between the P/M simulator and the SEC. There are no penetrations of the gap (screws, etc.), and

$$Q_{C,\phi} - Q_{C,0} = 0$$

Equation (3) reduces to

$$Q_{\text{solar},\phi} - Q_{W,\phi} = Q_{\text{solar},0} - Q_{W,0} + (P_0 - P_\phi)$$

The net energy captured by the SEC is

$$Q_{(\text{capt})\phi} = Q_{\text{solar},\phi} - Q_{W,\phi}$$

Therefore

$$Q_{(\text{capt})\phi} = Q_{(\text{capt})0} + (P_0 - P_\phi)$$

Because of the cracks along the edges of the ends of each louver (for clearance purposes), some solar energy (direct and propulsion-blanket-reflected) is captured at the full-closed portion ($Q_{\text{solar},0} \neq 0$). There is also an infrared energy interchange between the SEC and P/M which varies with the solar intensity.

Prototype Unit (SEC I)

The prototype unit was constructed purposely so that the solar energy captured was zero at the closed position. An infrared energy gain at Mars intensity was calculated to be 0.8 and 1.7 W for the closed position of the glass and no-glass configurations, respectively. Because of a low solar reflectance for the polished aluminum reflector (verified by measurements), the performance of SEC I shown in Fig. 3 was lower than expected. The comparison of the test results with the computer model was very good. Using the computer model with the better reflector properties, the performance of several window materials was evaluated. These results are presented in Table 1.

The window material optical properties are given in Table 2 and Fig. 4. The favored lightweight gold-coated Teflon and

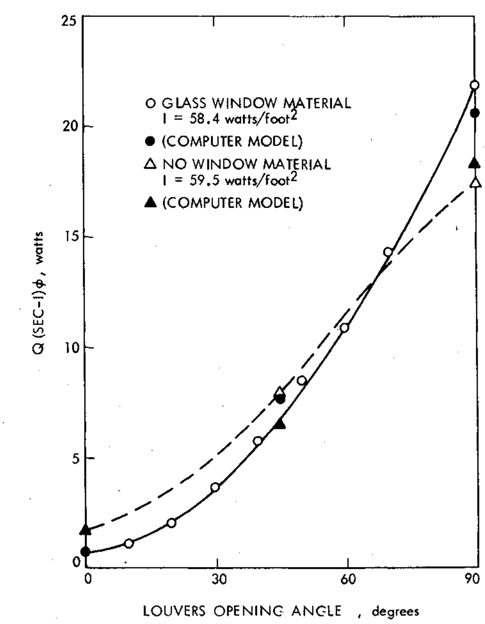

Fig. 3 Measured useful energy capture for SEC-I with and without a glass window.

Table 1 Maximum SEC performance at Mars for the various window materials

Window material	Net total energy captured, W	Solar energy captured, W	Net infrared energy captured, W	Window temp., °F
Glass with index coating	25.1	25.2	−0.1	40
Gold-coated glass	24.6	20.4	4.2	87
None	23.3	26.6	−3.3	N/A
Gold-coated Teflon	22.8	15.3	7.5	126
Gold-coated Kapton	22.4	9.9	12.5	178

Table 2 Thermal properties for the various window materials

Property	Window material				
	Index glass	Gold glass	None	Gold Teflon	Gold Kapton
Solar radiation					
Transmissivity	Fig. 4	0.75	1.00	0.56	0.36
Absorptivity	Fig. 4	0.05	0.00	0.22	0.39
Reflectivity	Fig. 4	0.20	0.00	0.22	0.25
Infrared radiation, sun side					
Transmissivity	0.05	0.00	1.00	0.00	0.00
Emissivity	0.85	0.10	0.00	0.19	0.12
Reflectivity	0.10	0.90	0.00	0.81	0.88
Module side					
Emissivity	0.85	0.86	1.00	0.74	0.78

Kapton films were evaluated earlier with the greenhouse feasibility model.[1] However, calculations indicated that these two films performed less favorably (Table 1) than the glass or no-window material configurations and, therefore, were dropped as candidate window materials. Also, it was questionable whether the films could survive the acoustic environment. Based on the computer model, the no-window material configuration performed so well that it was difficult to justify the glass configurations. However, as illustrated in Fig. 5, the calculated energy lost per SEC when passing through the Mars shadow with the louvers full open was substantial for both the index-coated glass and the no-window material configurations. Also shown in Fig. 5 is the effect of doing the occultation with the louvers closed and without any window material. Commanding the louvers to actuate twice each day (twice each orbit) was undesirable. Also, vibrational tests had demonstrated that the glass would be only 0.06 in. thick and contribute only about 0.3 lb (10%) to the weight of the SEC. As a result, the glasses, especially the gold-coated glass, continued to be the favored configuration.

Flight SEC

A flight SEC calibration unit graphically illustrated in Fig. 2 was tested separately, but in parallel, from the VO'75 temperature-control model. The purpose of the flight calibration unit was to determine the useful energy captured as a function of solar intensity and louver opening angle. The equations developed previously are applicable to the calibration unit.

Unlike SEC I, the flight unit does capture solar energy when fully closed. (Energy enters through the clearance cracks at the ends of the louvers.) A summary of the calculated solar and infrared energy for the fully-closed position and at the test intensities is shown in Table 3. Although not negligible, the solar energy captured at the closed position is very small, and, for most opening angles, the net energy captured can be approximated by the heater power difference $(P_0 - P_\phi)$. The heater power difference is plotted for each measurement in Fig. 6.

The net energy captured, $Q_{(\text{capt})0} + (P_0 - P_\phi)$, was determined for a solar intensity of 39.5 W/ft^2. Using the calculated infrared energy loss presented in Fig. 7, the solar energy captured, $Q_{(\text{capt})\phi} - Q_{W,\phi}$, also was determined for this

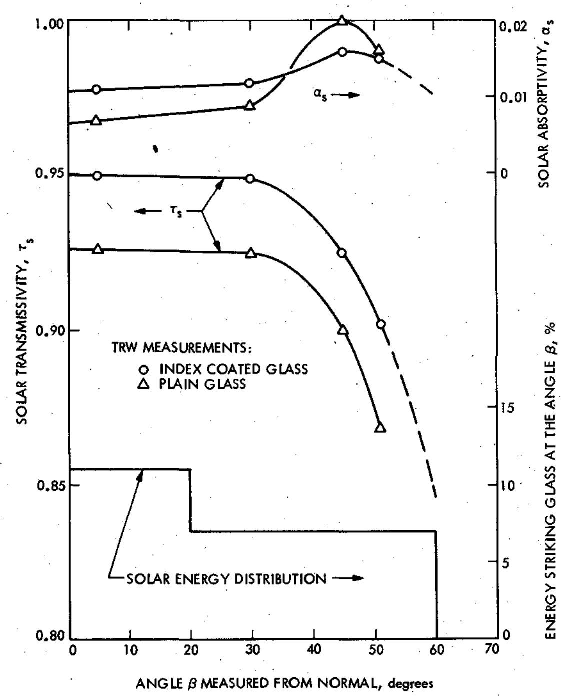

Fig. 4 SEC-I window properties.

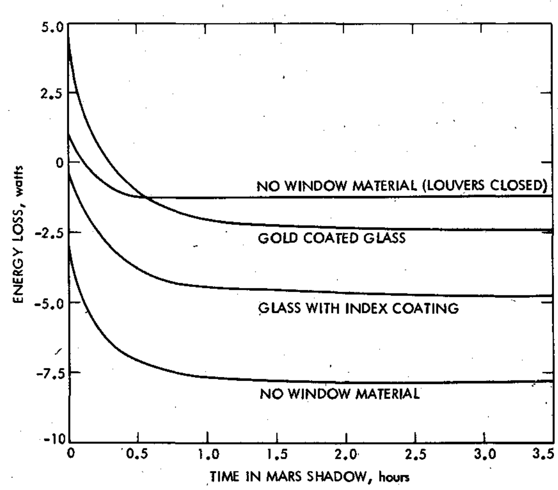

Fig. 5 Energy loss per SEC during solar occultation at Mars.

Table 3 Calculated net energy captured for louvers fully closed, flight SEC

Solar intensity, W/ft^2	$Q_{\text{solar},0}$ W	$Q_{W,0}$ W	$Q_{(\text{capt}),0}$ W
39.5	0.41	−0.55	−0.14
59.9	0.62	−0.08	0.54
117.2	1.21	1.65	2.85

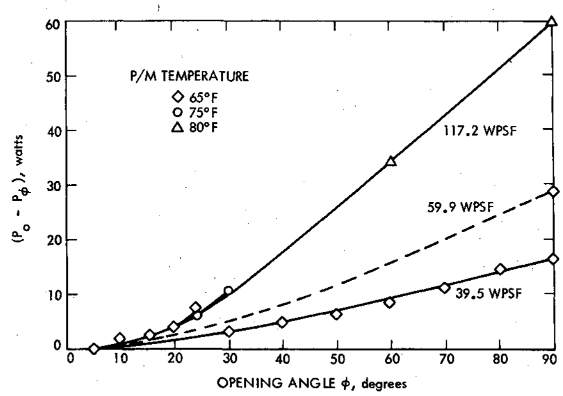

Fig. 6 SEC calibration unit TCM-1 results.

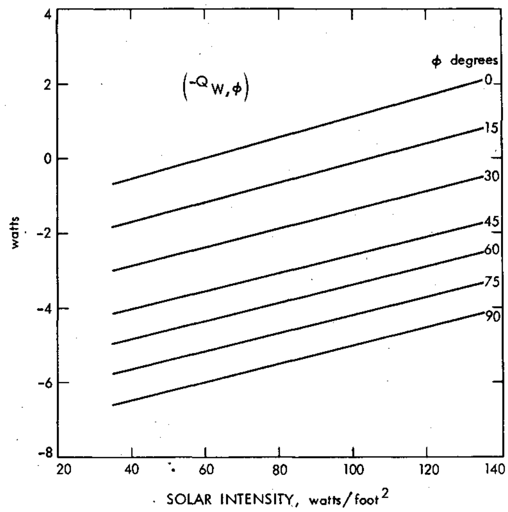

Fig. 7 Infrared energy loss by way of SEC.

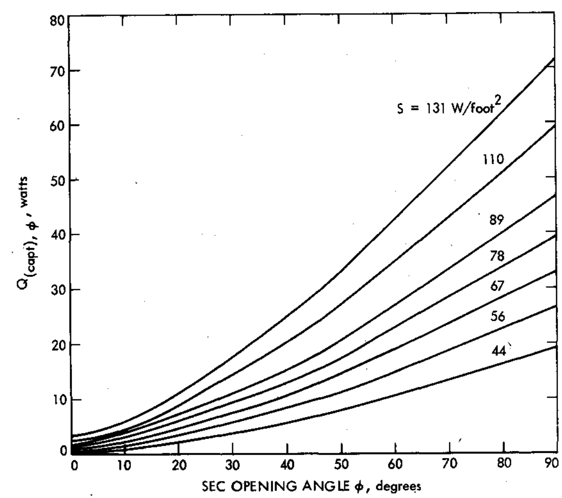

Fig. 8 SEC net energy captured.

solar intensity. The solar energy captured is directly proportional to the solar intensity, and, with the infrared energy loss (Fig. 7), the net energy captured was determined for several solar intensities that the Viking spacecraft will experience between Earth and Mars (Fig. 8).

The amount of solar energy reflected off the P/M thermal blanket is significant. Local measurements indicate that, at a Mars intensity of 44 W/ft^2, the intensity at the louvers is as much as 52 W/ft^2. Because of differences in the local blanket contour, the amount of reflected energy varied from one SEC position to another. It is estimated that the reflected energy is contributing about 10% to the net energy captured.

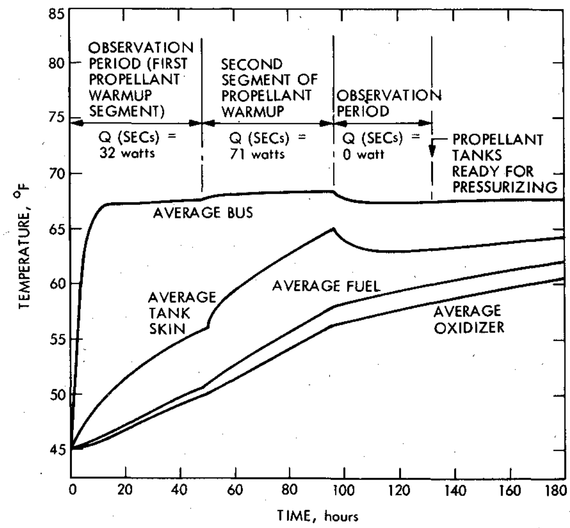

Fig. 9 Time required for propellant warmup.

Of the solar energy entering the SEC (not counting that reflected from the P/M blanket), 51% and 64% are captured as useful energy at the full-open position and at Mars and Earth, respectively. With better reflector properties and a quartz window, an 8% increase in efficiency is feasible with the VO'75 SEC configuration. An equivalent electrical heating system would have an efficiency of about 7 to 8%.

SEC's Viking Role

The SEC's will be utilized throughout the Viking mission. A partially opened position is used for launch to increase the Orbiter venting area. Shortly after acquiring the sun, the energy input is increased to as much as 25 W/SEC to warm the propellants from a possible low value of 45°F to an average bulk temperature greater than 58°F. Because of the large propellant load and lack of convection, propellant warmup is expected to take about five days. Following midcourse correction burns, the SEC's are reopened to heat the propellant lines between the biopropellant valve and solenoid latch valves prior to closing the latter valves as an added protection against the loss of propellants during the nine-month cruise to Mars. The additional heating assures that the lines will not overheat as the spacecraft approaches perihelion.

During approximately the next 90 days, the SEC's remain closed, and the average bulk propellant temperature warms to a maximum expected value of 76°F and then cools with decreasing solar intensity to 63°F. The SEC's opening angle then is increased to maintain the average bulk propellant temperature at 63° ± 2°F. Just prior to the Mars Orbit Insertion burn, the SEC's will be closed and remain closed during the resulting thermal soakback period. Following return of the P/M to normal temperatures, the SEC's will be actuated as required to maintain these temperatures during orbit operations.

Results

Test

Significant results relative to the performance of an individual SEC were just presented. To complete the SEC picture, the following discussion presents test results that illustrate the SEC's performance in temperature-controlling the VO'75 P/M and bus.

As previously indicated the SEC's first are utilized to warm the propellants from the cold launch temperature. (Cooling of the lander radioactive thermoelectric generators [RTG's] may require a pad air-conditioning temperature as low as 45°F.) During TCM, 100 W of SEC energy was introduced to verify that the components in the bus and P/M did not exceed allowable limits. However, the calculated temperatures (Fig.

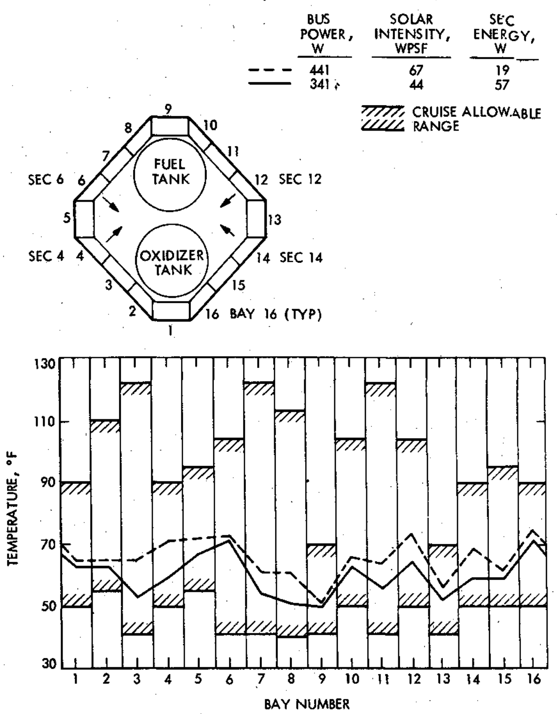

Fig. 10 Bus temperatures near Mars.

9) during propellant warm-up best illustrate the heating function performed by the SEC's. Without the SEC energy or an equivalent source, weeks of natural warming would be necessary before the propellant system could be pressurized and midcourse corrections made.

Temperatures at Mars for the bus are presented in Fig. 10. Despite an increase in bus power (100 W) and solar intensity (23 W/ft^2), temperatures in the P/M (except for the engine assembly, which is more strongly solar dependent) were maintained relatively constant by reducing the SEC energy from 57 to 19 W. Test data indicate that the solar energy is diffused through the P/M very well. A single full-open SEC at an Earth intensity created negligible temperature difference.

Without the SEC's (all SEC's fully closed), tests indicate that the propellant bulk and pressurant tank temperatures are 10°F below the allowable limits. The effect on the bus was more dramatic, with six bays exceeding the lower allowable limits. Although not the intent of the test, it did demonstrate the need for the SEC's or an equivalent source of energy.

During normal operations, testing indicates that it is not necessary to close the SEC's for the 3.6-h solar occultation. However, during a survival power mode (single battery failure), the SEC's would be closed for a long occultation to help minimize the loss in energy from the Orbiter.

Flight

Both Viking spacecraft were launched successfully on Aug. 20 and Sept. 9, 1975. Generally, SEC operations have been much like expected and as previously described. For launch, each SEC was open to 20°. For propellant warmup, 103 W was introduced by the SEC's continuously for four days on Spacecraft 1 (VO-1), and 86 W was applied continuously for seven days on Spacecraft 2 (VO-2). Following completion of the near-Earth midcourse correction maneuvers, the propellant lines were heated continuously for 24 hr with the same amount of energy used for propellant warmup. The lines were locked up successfully and the SEC's closed.

The SEC's remained closed for the next 154 and 141 days for VO-1 and VO-2, respectively. As the spacecraft approached Mars, the SEC opening angles were increased periodically to maintain the P/M and bus temperatures. Adjustments are commanded when convenient and compatible with the flight sequence. Typically, the opening angles are updated every three to four weeks. The SEC's were closed immediately after the Mars Orbit Insertion (MOI) maneuver to minimize the resulting thermal soakback.

At the time of MOI for VO-1, Mars was at its aphelion point, and the SEC's were reopened after the MOI to 60° (46 W total), which is approximately the maximum opening for normal operation at Mars. Currently, Mars is approaching its perihelion point, and the SEC opening is being reduced to compensate for the increasing solar intensity. The minimum opening at Mars is expected to be 20° (20 W total). With periodic adjustment, the SEC's will continue to maintain the P/M average temperature at 65°±2°F during the extended mission phase of Viking, which can last up to 2 yr.

To date, each SEC has been activated approximately 30 times. There has been nothing abnormal in their operation or performance. Flight bus and P/M temperatures have been very similar to the test temperatures when the conditions are comparable. Propellant bulk temperatures for all of the propulsive maneuvers since the Earth midcourse correction have been within 2°F of the desired 65°F. Thermally, the two orbiters are essentially identical, and there has been a combined total of 20 propulsive maneuvers to date.

Conclusions

Viking has shown the SEC to be a very useful and flexible temperature-control device. Tradeoff studies during the course of development indicated that the SEC's were lighter (by a factor of 2 to 3) and would cost less than an equivalent electrical heating system. The SEC system is six to eight times more efficient thermally than an equivalent electrical system.

Solar panel size is established early in the design, and additional power for thermal purposes, if later required, would not necessarily have been available with an electrical system. In contrast, the SEC net energy captured could be increased late in the design. This could have been accomplished by improving the reflector specularity (aluminized or silvered Teflon bonded to the polished reflector), increasing the length, or adding additional SEC's. Since losses by way of the thermal blankets (seams, penetrations, and attachment points) cannot be visualized properly until the blankets actually are fitted, an underestimate is very possible. Therefore, the energy increase possible with the SEC system late in the spacecraft design is a very attractive feature.

The only undesirable feature of the SEC's expressed during the course of development is that the energy is useful only to the temperature-control system. Energy provided electrically could, on a short-term basis, be used for other purposes.

SEC Automation

For Viking, a commandable SEC system is required. Without the propellant line lockup requirement, calculations indicate that the system probably could have been automated either by using bimetallic actuators like those for the Viking bus louvers or by using the freon-filled bellow actuators employed on the Nimbus bus louvers. The latter actuators probably would best fit a configuration like VO '75, since it would allow the temperature-sensing element to be slightly removed from the SEC.

SEC Replacement Heater

A potential application visualized for a modification of the SEC is as a bus bay replacement heater. Using bimetallic actuators, the SEC would mount to the bus bay shearplate, sharing it with the louvers. When the bay electronics are turned off, the SEC would introduce energy automatically to maintain the bay temperature. Diffusing the energy would not be required, and the reflector could be a straight solar reflector/absorber conductively and radiatively transferring energy to the shearplate. The unit also would replace the shade for the louvers.

Acknowledgment

This paper presents the work of one phase of the research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under NASA Contract NAS7-100. Several people at JPL have contributed to the design of the solar energy controller. Much of the configuration is due to the ingenuity of S.M. Shah and M.B. Gram of the Mechanical Devices Group. W.F. Carroll (materials), K.C. Curry (mechanical devices), and J.A. Plamondon (temperature control) also have influenced the design of this new temperature-control device.

References

[1] "Flight Projects," NASA Space Programs Summary 37-62, Vol. 1, Jet Propulsion Lab., Pasadena, Calif., March 31, 1970.