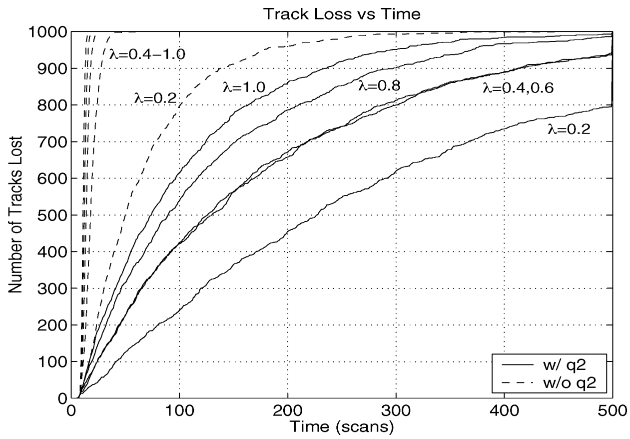

Fig. 4 Combined track loss for both targets with and without the q_2 parameter.

eigenvalue of the desired covariance matrix, for both targets with and without the use of the q_2 parameter (indicated by the solid lines). When q_2 is not used, the minimum eigenvalue of the actual covariance is 40–170 times that of the desired covariance, which means that the covariance goal is not achieved at any clutter level. Furthermore, the minimum eigenvalue of the predicted covariance difference (indicated by the dashed lines) is always positive, which means that the controller "thinks" that it will achieve the desired covariance at every scan. Because the probability of track loss is a function of the number of gated measurements[9,10] and, thus, a function of the prediction covariance and the clutter density, the failure to maintain a stable covariance leads to dramatically reduced track life, as shown in Fig. 4, the cumulative number of tracks lost at each time scan for each clutter density level.

When the q_2 factor is used (Fig. 3), the desired covariance goals are easily achieved, resulting in a positive definite covariance difference, in all but the highest clutter levels. The error between the predicted difference and the actual difference is always negative, which means that the system is consistently underestimating the sensor performance, as predicted in Sec. II. This improved covariance tracking performance also leads to significantly longer track lifetimes (Fig. 4) over that seen when the q_2 parameter is not used.

V. Conclusions

This Note augments a previously proposed multisensor covariance control technique through the addition of a scalar loss of information parameter, which allows the algorithm to evaluate the reduced effectiveness of each sensor when tracking in cluttered environments. Monte Carlo simulations show that without this parameter, the covariance control system is unable to maintain the desired covariance, resulting in a much larger actual covariance level. Use of the loss of information parameter restores this performance, which allows the control system generally to achieve the desired covariance goals.

Acknowledgments

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Mars and Mercury Missions Using Solar Sails and Solar Electric Propulsion

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Introduction

FROM the beginning of space travel, numerous types of propulsion systems, alternative to the original chemical engine, have been proposed. Today the low-thrust engines are a reality and the Deep Space 1 mission, launched by NASA in October 1998, demonstrated this in practice. Different types of missions obtain advantages from this technology. Scheel and Conway[1] and Kechichian[2] applied low thrust for transfer from low earth orbit (LEO) to geostationary orbit, and Kechichian[3] analyzed optimal steering for north–south stationkeeping of geostationary spacecraft. With regard to the lunar mission, Kluever and Pierson[4] and Herman and Conway[5] have studied the transfer from LEO to low lunar orbit in the three-dimensional case. Interplanetary missions have also been studied, particularly the rendezvous trajectories from Earth to Mars.[6,7] Solar pressure gives the possibility of using another propulsion system, the solar sail. The use of a solar sail is a wonderful prospect for interplanetary travel because no propellant mass is required, but only the weight of the sail.[8] Minimum-time for Earth–Mars transfer has been studied by Powers and Coverstone[9] and Otten and McInnes,[10] whereas McInnes et al.[11] have presented an extensive investigation of the use of solar sail propulsion for both Mercury orbiter and Mercury sample return missions.

Both solar sails and solar electric propulsion (SEP) use solar light, the first to push the sail and the second to produce energy for the engine. The present work compares the rendezvous missions to Mars

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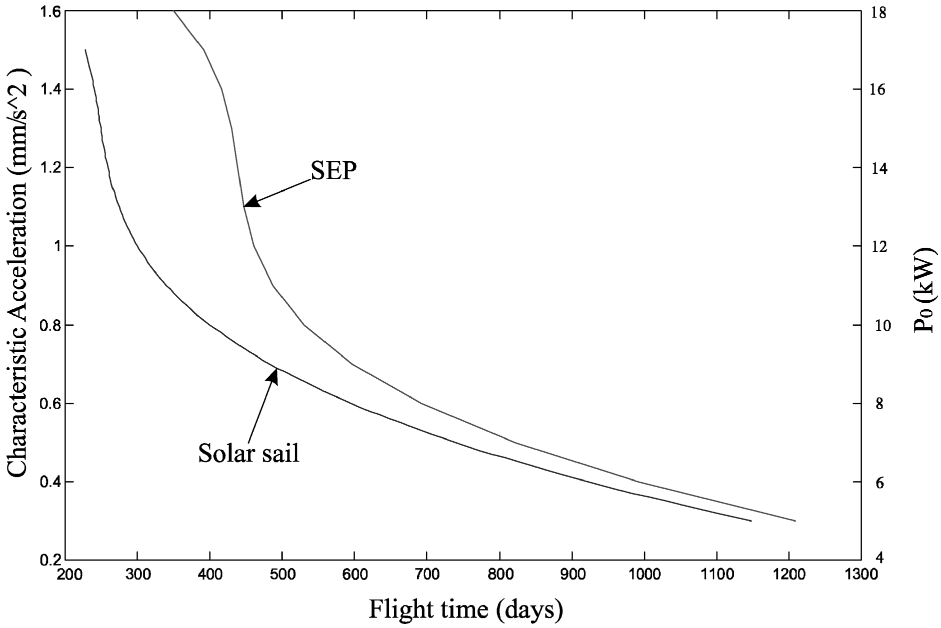

Fig. 1 Earth–Mercury performances for solar sail and SEP.

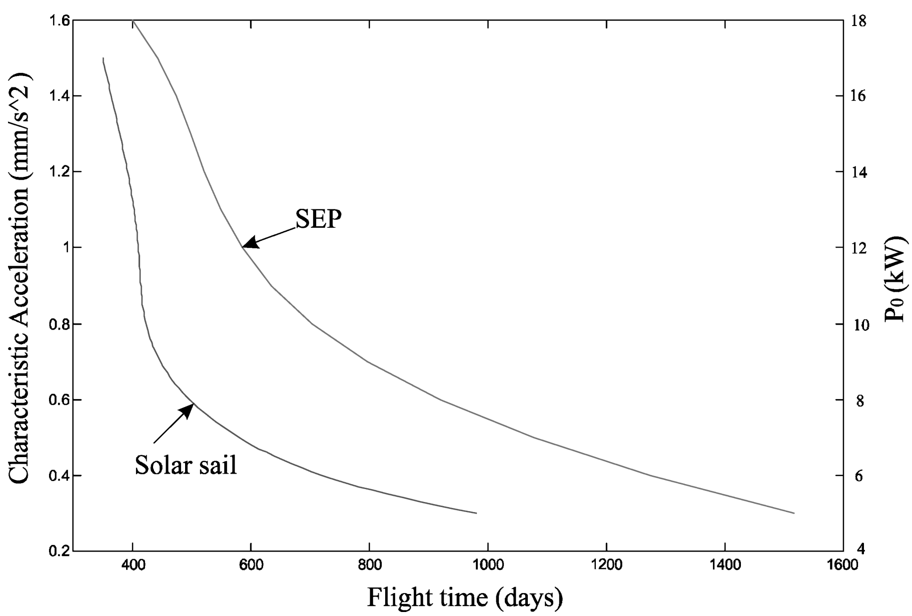

Fig. 2 Earth–Mars performances for solar sail and SEP.

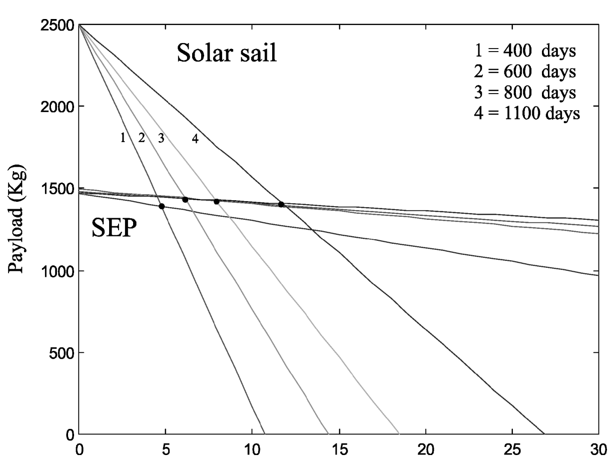

Fig. 3 Performances for Mercury transfer (2500 kg case).

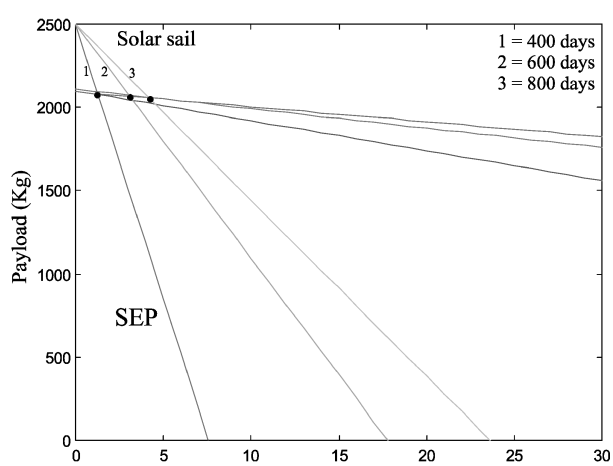

Fig. 4 Performances for Mars transfer (2500 kg case).

and Mercury using solar sail and SEP. The optimal control problem, minimum time for the solar sail and minimum-fuel expenditure for SEP, is solved by using a hybrid direct/indirect method. In fact, a direct method that minimizes the performance index by making appropriate changes to the control history and an indirect method to satisfy the necessary conditions are combined to form the hybrid nonlinear programming method. The solar sail performances are studied as a function of characteristic acceleration a_c, transfer time, and different areal densities. For SEP, the key parameters considered are the input power, transfer time, and specific mass. This Note is based on the Jet Propulsion Laboratory ephemeris data that provide the state vector for the planets. The range of launch dates considered is 2001–2013. A comparison between the performances obtained by the two different propulsion systems gives an analysis of the solar sail's potential with respect to SEP.

Performances for Mars and Mercury Rendezvous Missions

For the simulations, a flat sail with reflectivity coefficient $\rho = 0.9$ is used, whereas for SEP, a throttled engine is preferred. In this case the thrust and mass flow rates are characterized as polynomial functions of power available: mass flow rate (milligram/second) $= 0.47556 + 0.90209P$ and thrust (millinewtons) $= -1.9137 + 36.242P$ where P is the input power (kilowatts) to the power processing unit. The polynomial for solar array performance is

$$P = \left(\frac{P_0}{R^2}\right)\frac{\left(a_1 + a_2/R + a_3\big/R^2\right)}{\left(1 + a_4R + a_5R^2\right)}$$

where P_0 is the power at 1 astronomical unit, R is the solar range, and $a_1, \ldots, a_5$ are real constants.[12] Figure 1 shows the performances for Mercury missions. When the solar sail is used, a significant reduction in transfer time is possible by increasing the a_c, but over 1 mm/s^2 this reduction decreases. For the SEP case, the initial spacecraft mass considered is 2500 kg. Also in this case, it is possible to reduce the transfer time by increasing the input power, but over 10 kW this reduction decreases. Figure 2 shows the performances for Mars missions. As with the preceding case, a good reduction in transfer time is possible until $a_c = 0.6$ mm/s^2 for the solar sail and $P_0 = 12$ kW for SEP. To compare the performances using solar sail with those using SEP from the payload point of view, a different level of technology is considered in the next section. In particular specific mass (kilograms per kilowatt) for SEP and areal density (grams per square meter) for solar sail.

Comparison for Mercury Mission

Figure 1 gives the a_c, for a solar sail, and P_0, for SEP, for different transfer times. With these values and the initial spacecraft mass fixed (2500 kg in this case), it is possible to achieve a specified payload mass for different technological levels. For SEP the range from 0 to 30 kg/kW is considered, whereas for the solar sail the range varies from 0 to 30 g/m^2. Figure 3 shows the area where the solar sail is superior (upper-left zone) and the area where SEP is superior (lower-right zone). For example, considering an 800-day transfer time, we have the same payload using SEP with 7.8 kg/kW or a sail with 7.8 g/m^2. To the right of this point, SEP is better, whereas to the left, the solar sail is better. The achievable solar sail payload depends on the technological level more than does the SEP payload. This fact depends on the large size required for the sail, and the advantage in using solar sail increases with the technological level. With sail with the specific mass under 4.5 g/m^2, the SEP performances are inferior for every technological level or transfer time.

When 1500 kg is considered as the initial spacecraft mass, the solar sail increases the performances with respect to the SEP. In fact, the point with the same payload for different transfer times is translated to the right (for 800 days transfer from 7.8 to 8.4) and the area where the SEP is superior is reduced. In general, for a Mercury mission the solar sail provides a good opportunity, and this advantage increases with the level of technology and reduced initial spacecraft mass.

Comparison for Mars Mission

A similar analysis for a Mars mission is shown in Fig. 4. In this case the region where the solar sail is better than SEP is smaller and requires a very high level of technology for the sail ($<$1.5g/m^2). For a solar sail with specific mass $>$4 g/m^2, the SEP performances are the best for every transfer time. When the initial spacecraft mass is reduced to 1500 kg, the solar sail performance increases, as for the Mercury mission, but only a little improvement is noted. For a Mars mission, SEP shows good performance and would be preferred to the solar sail.

Conclusions

The minimum transfer time, using solar sail, and minimum fuel consumption, using SEP, for Mars and Mercury missions have been computed. The rendezvous problems have been studied using a hybrid direct/indirect method and the performances analyzed as a function of final mass, transfer time, and level of technology used. The comparison between the two propulsion systems shows the convenience of using the SEP for Mars mission, but when the sun is approach, the solar sail performances improve, and for the Mercury mission solar sail per formances are the best. In both cases, the solar sail performances, with respect to SEP, increase when the initial spacecraft mass is reduced.

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