Reducing the Cost and Risk of Orbit Transfer

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Use of an autonomous, Earth-referenced, low-thrust system for geosynchronous transfer or similar high-energy orbit transfer maneuvers can dramatically reduce costs and improve reliability over the use of independent high-thrust upper stages. Major cost savings are achieved by eliminating the duplication of components and subsystems between the spacecraft and upper stage. Eliminating components also reduces weight and increases reliability or, alternatively, allows for additional on-orbit propellant or equipment. Low-thrust transfer allows near-Earth deployment and checkout so that malfunctioning spacecraft can be retrieved or repaired. It provides a more benign transfer to the operational orbit from which retrieval is not normally possible. It also allows greater operational flexibility for storage in low Earth orbit following checkout or use of various orbits during different mission phases. Full autonomy during the critical sequence of perigee burns reduces both the cost and risk of ground operations. Finally, risk is further reduced by a fail-safe approach to orbit transfer that has fewer and less catastrophic failure modes than high-thrust inertially guided systems.

Introduction

THE thesis of this paper is that appropriately applied *autonomous low-thrust orbit transfer* has the potential for substantially reducing both the cost and risk of going from low Earth orbit (LEO) to geosynchronous (GEO), Molniya, or other high-energy orbits. In addition, the equipment needed for autonomy provides enhanced flexibility. For example, no duplication is required for orbit change maneuvers, avoidance maneuvers, or deorbiting to LEO for either reentry or Shuttle recovery and repair.

By *low thrust* we mean a propulsion system that produces a maximum acceleration low enough to permit the deployment of solar arrays, antennas, and other appendages in LEO so that the system can be thoroughly checked out before insertion in a high-energy orbit. Typically, this implies maximum accelerations on the order of 0.01 to 0.1 g. We wish to distinguish this from very low-thrust electric propulsion systems that provide less than 0.00001 g.[11] Many of the results and methods discussed here are applicable to electric propulsion systems. However, electric propulsion has other constraints, such as power requirements and extremely long transfer times, which require considerations beyond the scope of the present paper.

Low-thrust orbit transfer has been considered for some time.[1-8] It has been used in both planned ascent[9] and unplanned mission rescue operations.[10] In this paper, we will discuss the relative merits and demerits of low thrust for planned high-energy orbit transfer, with particular emphasis on two key features:

1) The use of the spacecraft itself to provide most of the housekeeping functions during transfer to reduce the weight, complexity, and therefore cost and risk of the transfer.

2) The use of a particularly simple autonomous transfer control mechanism to further reduce the complexity and cost of both the spacecraft and ground operations while simultaneously reducing the risk of orbit transfer.

Use of the spacecraft for housekeeping functions can imply either a relatively "dumb" upper stage that relies on spacecraft subsystems or, alternatively, the spacecraft flying itself from LEO to a high-energy orbit. In either case, obtaining the substantial cost, risk, and flexibility benefits that low thrust has to offer requires us to change to some degree the division of responsibility typical of space missions. We will discuss the implications of this change and why we believe this to be the largest single impediment to providing lower-cost, lower-risk, high-energy orbit transfer. This paper extends work initiated by two of the authors in the early 1980's.[1]

System Configuration

As illustrated in Fig. 1, there are three basic configuration options appropriate to low-thrust transfer. In option A, the

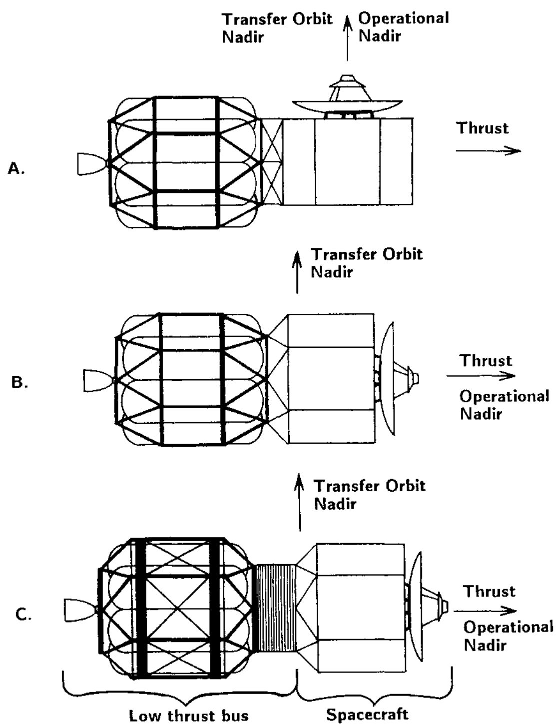

Fig. 1 Schematic representation of alternative low-thrust configurations.

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spacecraft flies itself to a higher orbit using a low-thrust engine and tanks mounted on the "side" (nominally the roll axis) of the spacecraft. In this configuration, the spacecraft is oriented throughout the transfer orbit with the same nadir pointing face that will be used later for the operational orbit. This simplifies both the ground communications and Earth sensing activities and minimizes the need for additional equipment. Because the spacecraft will maintain a single nadir pointed face at all times, the cost of implementing solar array pointing and assessing thermal issues will normally be minimized. If the spacecraft includes nadir pointed instruments, this configuration ensures that without additional mission constraints, the instruments will never look directly at the Sun.

The principal drawback to option A is that it is different from the way spacecraft are ordinarily built. Most spacecraft are marketed and built on the basis of similarity to previous functioning designs. Spacecraft built in this fashion will reduce their overall design heritage to some degree.

An alternative is option B, in which the tanks and motor are moved to the nominally zenith pointing face of the spacecraft. This is more in keeping with traditional spacecraft design and puts the motor and tanks on the surface that is least in demand for instruments and appendages. Earth sensor positioning is possible for this configuration so that the same sensors can be used for both transfer orbit and operational orbit. Ground link antenna placement and possible timeline constraints due, for example, to instruments pointing toward the Sun, would need to be assessed on a mission-by-mission basis. Although this is not as straightforward as option A, it also poses no additional constraints relative to what the system would encounter in the use of a separate upper stage.

The third alternative (option C in Fig. 1) is to use a separate low-thrust upper stage. This provides a system that is relatively close to the traditional approach of a separate high-thrust upper stage. It can work well with pre-existing spacecraft designs. For example, Ref. 1 provides a detailed assessment of how a low-thrust bus in this configuration could be applied to orbiting the Tracking and Data Relay Satellite (TDRS). This configuration allows the system designer to take advantage of as many pre-existing subsystems as possible while incorporating additional hardware as necessary in the low-thrust bus.

Subsystems

In the proposed system, most of the subsystems and components used for orbit transfer operations will also be used for on-orbit operations. Attitude sensing in the transfer orbit will be done by conical Earth sensors (CES) and a yaw gyro, all of which will be used in the operational orbit. All TT&C, coast attitude control, power distribution, and housekeeping will be done by the spacecraft subsystems. Low-thrust engines in the 50–500-lb (200–2200-N) thrust range would be added to the normal complement of small thrusters typically used for orbit maintenance. Propellant tanks must be substantially increased in size to accommodate the delta V needed for major orbit transfers. The only piece of electronics not ordinarily onboard the spacecraft is the Autonomous Low Thrust Controller (ALTC), consisting of a single card of electronics that provides main engine firing control signals (U.S. patent pending). Thus, with the exception of this main engine firing control signal, the normal spacecraft subsystems provide all of the functions for orbit transfer.

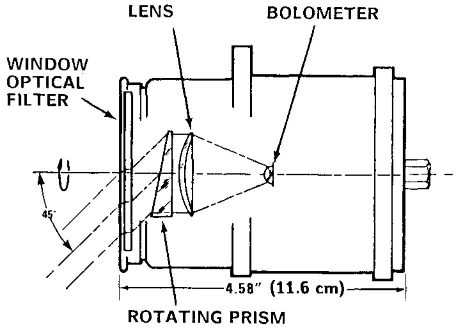

Fig. 2 Conical Earth sensor (CES) used for providing both Earth width data for autonomous orbit transfer and attitude data for both the transfer and operational orbits.

Propulsion

The principal issue in the propulsion subsystem is the choice of low-thrust engine. A number of alternatives exist, including the use of 1, 2, or 4 inexpensive storable bipropellant engines.[1] Each of these options provides a substantial cost advantage relative to a single high-thrust solid motor or bipropellant engine. As a specific example, we will assume in this paper the use of two small gimballed engines. This option reduces the overall cost, provides attitude control during motor firing, and provides full propulsion redundancy. In addition, by turning off one of the two engines, this configuration allows the acceleration to be maintained at a low level near final burnout when the system mass is a minimum.

All mission phases will normally have some level of propellant reserve. If the transfer orbit propellant is the same as that used for the smaller on-orbit thrusters, this propellant reserve can be used to provide increased on-orbit life.[13] Alternatively, the on-orbit propellant reserve can be reduced to provide increased payload mass.

Attitude Sensing

Attitude sensing to maintain nadir pointing in the transfer orbit will be provided by the same sensor complement used on-orbit. This orientation is used to ensure that the thrust vector is maintained perpendicular to nadir. Yaw sensing for orbit inclination control is provided by a yaw gyro updated once or more per orbit from sun sensor data.

The primary roll and pitch attitude reference for the orbit transfer, as well as on-orbit, will be provided by CES's, as shown in Fig. 2. The specific sensor orientation will depend on the spacecraft configuration. In the case of spacecraft option A or option C with sensors added to the bus, the sensor fields of view will be oriented as shown in Fig. 3a. This figure shows the spacecraft-centered celestial sphere, with the heavy circles showing the field of view for three 45-degree cone angle CES's.[16] Two CES's are adequate to provide altitude-independent attitude data from below LEO to above GEO. In this case, a third sensor has been added to provide full redundancy. Figure 3a shows the sensing geometry from LEO, and Fig. 3b shows this geometry as seen from GEO. Comparing Figs. 3a and 3b shows that the sensing system provides good coverage, with no geometrical singularities over the entire altitude range.

For simplicity, we assume a bang-bang control system during the transfer orbit, with fixed control deadband in the sensed parameters. This will result in a fixed control deadband in roll and a control deadband in pitch, which is a function of spacecraft altitude as shown in Fig. 4. Although not critical to the success of the orbit transfer, this curve has a particularly convenient form. The control deadband is small when the engine is being fired for perigee burns, somewhat larger at intermediate altitudes, and small again at the operational altitude. As can be seen from the two curves in Fig. 4, the high-end altitude at which the deadband becomes small is sensitive to the tilt or mounting angle of the CES's. This can be adjusted easily to accommodate any desired operational altitude from intermediate altitudes such as GPS to supersynchronous. Thus, the system is robust in its ability to handle various final altitudes and to provide nadir pointing attitude control over the entire orbit transfer process.

Autonomous Low-Thrust Controller (ALTC)

In a low-thrust ascent to GEO or other high-energy orbit, orbit transfer is accomplished by a sequence of typically 10 to 30 extended burns centered on perigee, followed by a series of

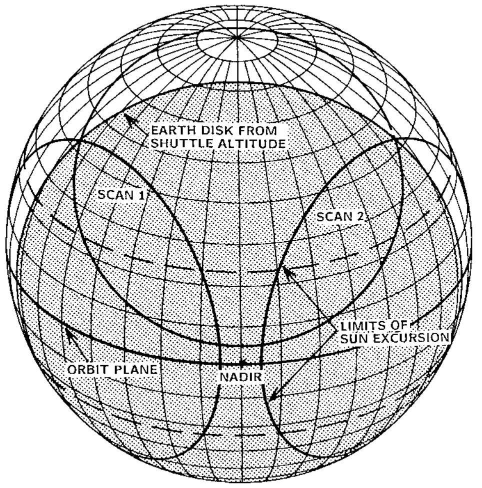

Fig. 3a Geometry of CES sensing system with Earth disk, as seen from Space Shuttle altitude.

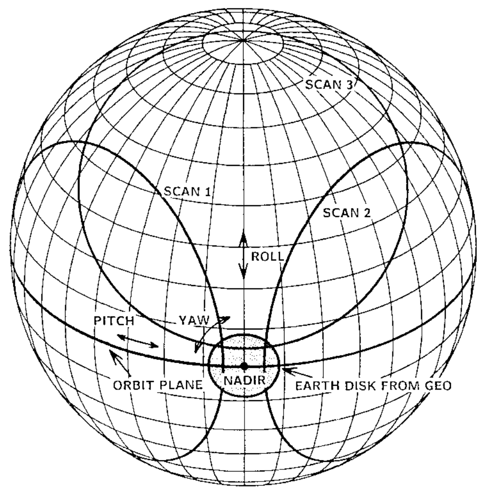

Fig. 3b Geometry of CES sensing system with Earth disk, as seen from geosynchronous altitude.

1 to 3 apogee burns. This process is shown schematically in Fig. 5. Note that this figure does not include either plane changes or rotation of the line of apsides. Both of these effects will need to be computed as part of the orbit transfer logic but will not affect the basic nature of the transfer.

The transfer sequence is initiated by an onboard timer, which provides the start and stop signals for the first burn, creating a somewhat eccentric intermediate transfer orbit. Subsequent perigee burns are controlled by the ALTC, based on the apparent size of the Earth sensed by the CES's. Specifically, the ALTC turns on the low-thrust motor whenever the size of the Earth sensed by two CES's exceeds a preset threshold. This is equivalent to firing the motor whenever the spacecraft is below a given threshold altitude. (A patent is currently pending on this motor firing control scheme.)

There are several advantages to this engine firing control scheme. The perigee burn sequence is fully autonomous and does not depend on ground communications, ground station coverage, or any ground-based (or onboard) orbit propagation. No ephemeris or orbit computation is required. Because the size of the Earth is changing rapidly in the region where the burns are initiated and terminated, a firing accuracy of 5–7 s relative to true perigee can be achieved, depending on the eccentricity of the particular intermediate transfer orbit. (See Fig. 6.) Burns are made relative to true, sensed perigee rather than any computed or predicted time and are therefore independent of any variations in the actual delta V imparted during a given burn or burn sequence. Thus, the system is not only autonomous but also less susceptible to data, communications, or computation errors.

The Earth sensors provide the motor control signals, as well as the attitude control signals. The spacecraft remains nadir-oriented during the potentially long arc over which the maneuver occurs. Thus, the thrust is applied perpendicular to nadir, rather than in a fixed direction in inertial space, and provides

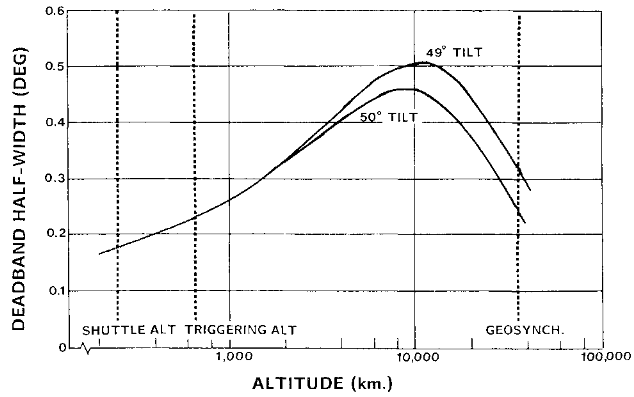

Fig. 4 Pitch deadband vs altitude for triggering on fixed scan wide of ±0.5-scan-deg. Tilt refers to the mounting angle of the CES relative to nadir.

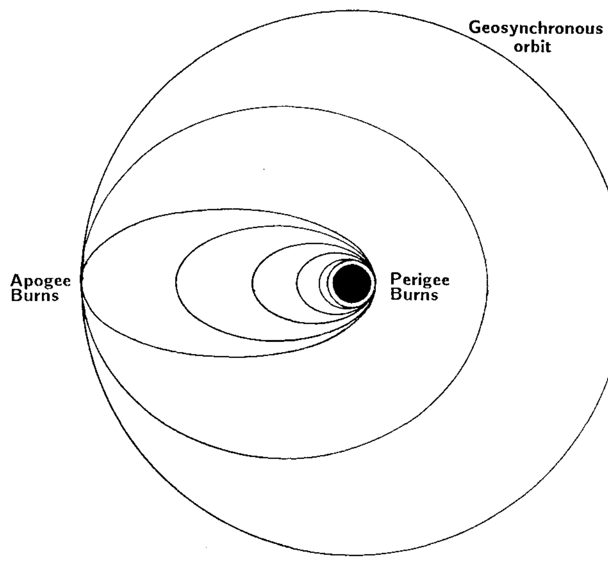

Fig. 5 Schematic representation of low-thrust orbit transfer.

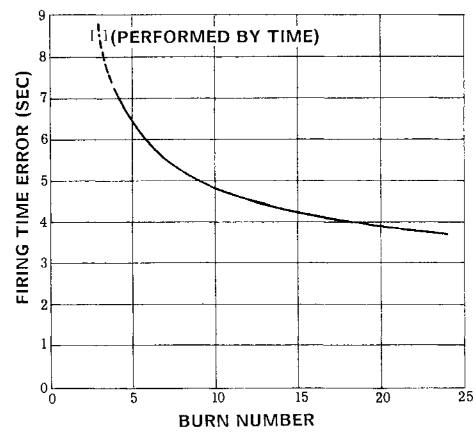

Fig. 6 Firing time error vs burn number for ±0.5-scan-deg control deadband and 24-burn sequence.

for an efficient transfer. (Efficiency of orbit transfer is discussed in the next section.)

Because the same sensors are being used for both attitude control and motor firing signal, the system has very benign failure modes. Anything that causes a loss of control of the spacecraft, such as unanticipated shutdown of one of the engines, will also cause a reduction in the size of the Earth sensed by at least one of the CES's. Since the firing logic requires both signals to be large, the system would immediately shut down the low-thrust motors. This means that the propellant supply would not be wasted and the transfer would not be perturbed by continued firing in an improper direction. Thus, if control can be restored, the system has the potential for proceeding to the proper final orbit, using the redundant engine or even the small control thrusters, if necessary. Overall, the system has fewer failure modes and provides a safer haven if a failure should occur than with high-thrust, one-shot transfers.

Transfer Orbit Timeline

As a specific example, we will consider here the process of transfer to geosynchronous orbit. However, the autonomous low-thrust control process can be applied equally well to transfer to other high-energy orbits such as GPS, Molniya, or those above GEO.

The orbit transfer process begins with insertion into LEO by the Shuttle or an expendable launch vehicle. The autonomous low-thrust transfer process essentially replaces the upper stage of the expendable booster or the transfer vehicle carried in the Shuttle. While the spacecraft is in LEO, the solar arrays, antennas, and other appendages are deployed, and the system is tested as fully as possible. Thus, the deployment process, the payload, and the majority of the spacecraft subsystems can be verified after having undergone their most strenous mechanical environment and at an altitude where there is at least the potential for repair or recovery.

Ordinarily, the spacecraft's low Earth orbit will be nearly circular and at an inclination other than the desired end value. For energy optimization, the plane change will be done partially during perigee burns, but predominately during apogee burns. The ability to combine in-plane and out-of-plane components is not affected by the use of low thrust. (Yaw control is required for orbit plane control, irrespective of the maneuver strategy.) A plane change during the transfer significantly increases the complexity of the planning but does not affect the conceptual approach and will not be addressed further.

Because the initial orbit is nearly circular, perigee is not well defined, and the ALTC cannot yet take control. The first burn or, in some cases, the first two burns would be controlled by a timer onboard the spacecraft. Conceptually, it is convenient to think of the first burn as being centered on the equator so that apogee will ultimately be near the equator. In practice, the timing of the first burn is preplanned to account for rotation of the line of apsides during the course of the transfer process. This mitigates the need to use a more complex burn strategy that would keep the line of apsides from rotating.[14,15] During the initial burns, the Earth size would still be tested to ensure that attitude was being correctly controlled.

Once an elliptical orbit has been established, the ALTC will be initialized, and it will provide thruster firing control signals until the ALTC senses that a predetermined apogee altitude somewhat below the final altitude has been reached. The final perigee burn will generally be shorter than previous burns and will be computed either on the ground or in an onboard computer to correct the final apogee height and line of apsides. It can then be triggered either by time or by resetting the threshold altitude appropriately. Following the perigee burn sequence, a much smaller number of apogee burns will be required to raise perigee to the desired altitude and complete any desired plane change. (A smaller number of burns is required because of the lower speed and longer time spent in each apogee passage.) Depending on the degree of onboard computation, the apogee burns can be computed either onboard the spacecraft or on the ground. Because of the high apogee altitude, ground commanding and communications are no longer a problem. Thus, in the most economical implementation, the perigee sequence would be controlled by the ALTC, and the apogee sequence by ground command. However, full autonomy is feasible if it is desired to meet mission requirements.

The transfer time and transfer efficiency (i.e., losses relative to a Hohmann transfer with impulsive burns at apogee and perigee) depend on both the acceleration provided by the motor and on the length of the burn arc about perigee. Representative data for both transfer time and efficiency are shown in Figs. 7 and 8. Note, that for most cases, the losses due to a finite burn duration are of the order of a few hundred ft/s or less and are not a major consideration for transfer strategy. For a 5000-lb (2300-kg) spacecraft, 100 ft/s (30 m/s) represents approximately 50 lb (23 kg) of on-orbit weight.

Cost, Risk, and Reliability

Our goal is to obtain orbit transfer at minimum cost with minimum risk. Ordinarily, these are opposing goals but, in the case of low-thrust transfer, both cost and risk can be substantially reduced relative to a high-thrust upper stage.

Cost

Table 1 provides a rough estimate of the cost and weight savings relative to a wholly independent high-thrust solid motor upper stage of a spacecraft using the autonomous low-thrust transfer, with support from the spacecraft subsystems

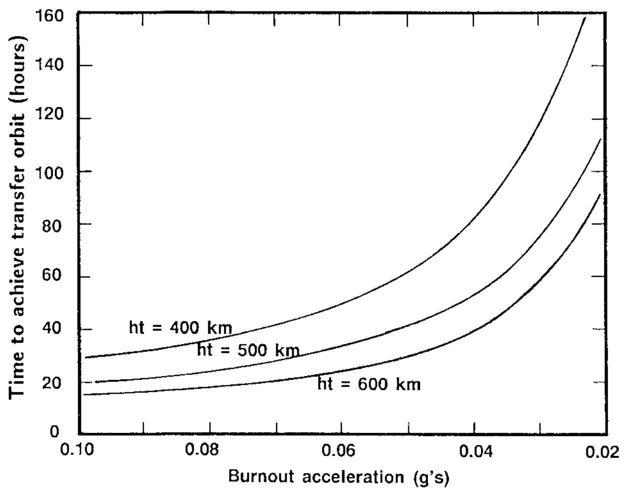

Fig. 7 Time to achieve transfer orbit.

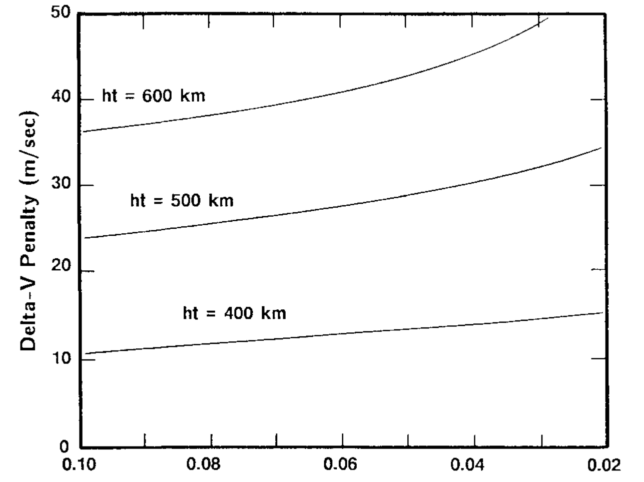

Fig. 8 Efficiency of low-thrust burns relative to Hohmann transfer.

Table 1 Rough estimates of cost and weight savings with autonomous low-thrust control system relative to a wholly independent high-thrust solid motor upper stage

Area	Independent upper stage	Autonomous low-thrust system	Cost savings, $M	On-orbit weight savings,[a] lb
Propulsion	Large solid motor & second solid or biprop(s)	2 small biprop engines	8	560
TT&C	Independent sys.	Use spacecraft system	2	80
Guidance	Independent inertial guidance	Auto. low-thrust controller & on-orbit sensors	3	80
Control	Independent RCS	Use spacecraft system	2	180
Power	Independent batteries	Spacecraft solar array	3	550
ASE	Heavy, accurate	Moderate, minimal ejection accel. req.	4	1380
System I&T	Required	Part of spc I&T	3	—
I_{sp}	~295 s	~310 s	—	370
Finite burn losses (Fig. 8)	Small	100–200 ft/s (30–60 m/s)	—	—100
Totals			$25M	3100 (1400 kg)

[a]Based on 5000-lb (approximately 2300-kg) payload.

as described earlier. (The table assumes a 5000-lb (2300-kg) payload to geosynchronous orbit.) In the case of both systems, costs are highly variable and are dependent on both system and subsystem requirements. Consequently, the estimates shown are intended only to provide an approximate assessment of where the major cost and weight savings arise and a rough order of magnitude of the total dollar value of these savings. With this caveat in mind, we see from Table 1 that the total estimated savings are on the order of $25 million. If we assume that on-orbit weight is valued at $10,000/lb, then the weight savings represents an additional $30 million.

As one would expect, the largest individual saving is in the propulsion subsystem. The independent stage requires a large, nonredundant motor for insertion into the transfer orbit and a second nonredundant motor for the apogee burn or a single, expensive high-thrust liquid engine. The proposed low-thrust configuration uses two small, economical bipropellant engines, either of which can provide both perigee and apogee burns. This configuration provides full redundancy at much reduced cost and weight, taking into account the added propellant tanks required for the low-thrust system.

The smaller bipropellant engines of the low-thrust system have a significantly higher I_{sp} than do the large solid rockets. This implies a greater on-orbit weight by 7% or more, depending on the optimization of the low-thrust motor. This advantage is lost if a high-thrust bipropellant engine is used.

The table assumes that the solid motors for the independent stage are sized correctly for the transfer at hand, as will generally be the case for transfer to GEO. However, using the same engine for nongeosynchronous missions (e.g., GPS or a Molniya orbit) can result in much more significant weight penalties for the independent stage. These losses are very mission-dependent but may well be as large or larger than any of the other items.

The independent upper stage includes separate guidance control, reaction control systems (RCS), TT&C, and power subsystems. The latter two are required to support the needs of both the stage and the spacecraft. The spacecraft subsystems are typically unusuable while the spacecraft is in its stowed configuration, ordinarily maintained during the entire orbit transfer process. In the low-thrust transfer, the opposite situation exists. The spacecraft is deployed in low Earth orbit, so that both the TT&C and power systems can be functioning normally and can provide services for both the spacecraft and the low-thrust subsystem. (There are differences in payload power requirements and eclipse durations during transfer that may or may not counterbalance for a given mission.) Maintaining a single face toward the Earth at all times makes the communications link less difficult, and maintaining a deployed configuration reduces thermal anomalies relative to the stowed configuration.

The guidance and control systems of the two approaches differ greatly. Independent high-thrust upper stages require sophisticated and relatively expensive inertial guidance systems because they have only limited opportunities to establish the correct transfer orbit. They typically have full three axis reaction control systems (RCS) to maintain control during coast and to re-establish orientation for subsequent burns. In contrast, the low-thrust system uses Earth sensors that can be used in the operational orbit and the ALTC, which is a very small, one-card electronics box. It requires a directional gyro (correctable by periodic sun sensing) to control yaw. For those spacecraft that carry a gyro complement, this will be available at no additional cost. The spacecraft can be controlled using the spacecraft RCS during coast periods and probably during burn periods if an integrated design is developed.

Finally, present independent upper stages require heavy and expensive airborne support equipment (ASE) to achieve ejection from the launch vehicle. Low-thrust systems do not have precise ejection requirements because they will undergo a deployment and acquisition sequence after ejection. This allows both reduced cost and weight savings for the ASE.

Discussion: Risk, Reliability, and Responsibility

Low-thrust transfer increases the transfer time from approximately 6–8 hours to 2–5 days, depending on the motor

Table 2 Comparison of autonomous low-thrust transfer with independent high-thrust upper stage

Characteristic	Independent high-thrust upper stage	Autonomous low-thrust transfer
Impact of motor failure	Mission failure	No impact—use backup
Impact of control failure during burn	Mission failure	No impact—use backup
Impact of temporary command link loss	Possible degradation	No impact
Capable of autonomous transfer	Yes	Yes
Ephemeris prediction necessary	Yes	No
Low orbit deployment and checkout	No	Yes
Potential for low- or intermediate-altitude storage	No	Yes
Total transfer time	6–8 h	2–5 days
Peak accelerations	$>2.5\ g$	$<0.1\ g$

used and the transfer process. Normally, this time increase is not a problem. Ground contact time is limited when the system is near perigee. This is not a problem for the primary transfer mode because the perigee burns are fully autonomous. However, ground contact may be important for backup mode operation. The percentage of time spent in eclipse will normally be increased while the spacecraft is in low Earth orbit. This may or may not be a problem, depending on the specific spacecraft and the power requirements of heaters compared to payload operation.

Table 2 summarizes the principal issues associated with orbit transfer risk and reliability. We wish to emphasize that we are not arguing that the components themselves are inherently more reliable in the low-thrust process. Rather, there are fewer components and fewer nonredundant, critical components. Most important, the transfer process itself is inherently more benign and more reliable, as can be seen by examining the consequences of various potential failures in Table 2.

High-energy orbit transfer represents well-developed technology. Nonetheless, the traditional high-thrust orbit transfer is an inherently high-risk process. The large motors or engines are potential single-point failures subject to substantial stresses. High thrust subjects the spacecraft to its most severe mechanical environment at the same time that it carries it out of reach of recovery or repair. Precise pointing and timing are required, and these must be accomplished in a relatively short time because most spacecraft have a short life span when on orbit in the stowed configuration. There is very little time or potential for checking spacecraft systems or deployment mechanisms before the spacecraft goes out of reach. In most cases, a control or motor failure during the burn will result in loss of the mission. On the positive side, cases have now occurred of both recovery[18] and mission completion[10] following the failure of high-thrust upper stages, but at considerable expense and risk.

In contrast to the high-thrust upper stage, the autonomous low-thrust transfer is a much more benign fail-safe process. It is possible to deploy the spacecraft and check out essentially all subsystems before moving it to an inaccessible orbit. The transfer process itself presents a very mild mechanical environment for the spacecraft. The critical perigee burn sequence is controlled autonomously and does not require any ground commanding. All critical transfer components (motor, sensors, control actuators, and firing logic) can be made fully redundant at lower cost than the nonredundant high-thrust counterparts. If a propulsion or control failure should occur during a burn, the sensed size of the Earth will cause the ALTC to shut off the propulsion system without wasting needed propellant or degrading the orbit. Consequently, if the failure can be rectified (by using the backup sensors, actuators, motors, or even the on-orbit RCS thrusters), then full mission completion is to be expected.

The principal drawback to the low-thrust transfer process is the significant change required in both responsibility and funding patterns. The low-thrust transfer process represents a shift in both of these elements, from the launch group to the spacecraft group, in both the funding and engineering communities. Such a shift in a well-established pattern frequently meets strong resistance and represents a lessening of design heritage. Nonetheless, this shift in responsibility has been identified by the spacecraft insurance industry as a major element in reducing space mission insurance costs.[17] If we can accommodate a new way of doing business, then we can cut costs and increase reliability by using the spacecraft itself as the primary transfer vehicle.

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