ARTICLE NO. 80-4131

Orbit Transfer Propulsion and Large Space Systems

K. E. Kunz*
Rockwell International Corporation, Downey, Calif.

Propulsion system requirements for large spacecraft of the future are unique when compared to existing vehicles. Some of these characteristics were investigated and are reported here. An overall comparison of the relative merits of various types of propulsion systems was made for the orbit transfer of an advanced communications space platform from low Earth orbit (LEO) to geosynchronous equatorial orbit (GEO). A cluster of cryogenic propulsion modules appears to be superior in that fewer modules and launches are required than for either of the storable propellant or solid propellant modules. In addition, less trip time is required for the cryogenic propulsion module concept than for a solar electric system. Propulsion subsystem considerations for the modules include the selection of an optimum thrust-to-weight ratio and a system which gives a gradual thrust buildup without large transients. Other considerations include the location and arrangement of engines, number of engines per module and numbers of modules for a given mission. A thrust-to-weight ratio of 0.2 was found to be near optimum for the linear-shaped space platform. The use of multiple modules with multiple gimballed engines for each module provides the means for thrust vectoring and thrust-to-weight control. The structural concepts considered include an erectable pentahedral truss and a space-fabricated tri-beam platform design. These designs are based on the use of graphite-epoxy.

Introduction

WITHIN the decade, very large spacecraft will be assembled in low Earth orbit (LEO). In some instances, these spacecraft will consist of very large area elements such as antennas and solar arrays mounted on structural platforms which, in themselves, will be very large. These lightweight and very flexible spacecraft will require specially designed propulsive systems to transfer them from low Earth orbit to much higher orbits. Interactions between such structures and the propulsive thrust forces can have much influence on the overall spacecraft configuration, structural design, and control functions to a greater degree than that experienced heretofore with more conventional spacecraft. There is a need for options in propulsion systems to transport these large space systems to higher orbits such as geosynchronous equatorial orbit (GEO). This paper examines some of these structural/propulsive interactions, deduces significant propulsion requirements, and illustrates some concepts with design characteristics.

The structural/propulsive interactions examined are illustrated in Fig. 1, and consist of the total weight sensitivity to the thrust-to-weight ratio (T/W), the structural load amplification factor sensitivity to the thrust buildup rise time, and the location and arrangement concepts to obtain thrust vector control and enhance T/W control.

Large Space Systems

The magnitude of the size and weight characteristics of various large space systems designs is illustrated in Fig. 2, which shows sizes ranging from 100 to 1500 ft, and weights from 23,000 to 160,000 lb. These designs, resulting from prior studies, also illustrate the differences in configuration that are mission and function dependent. Representative examples include the electronic mail satellite (EMS) (antenna farm), a version of the On-Orbit-Assembled (OOA) spacecraft, and a large optical spacecraft.[1] Other characteristics shown on the figure include the T/W, assembly and operational altitudes, and propulsion summaries for each design.

Another example of the large space systems, resulting from more recent studies, is shown in Fig. 3, which also highlights some of the more significant components of the system.[2] This system is an advanced communications space platform, with an erectable pentahedral truss structural design, and is based on the use of a graphite-epoxy composite material to minimize structural deflections due to large temperature gradients. The overall platform length is 787 ft with a basic width of 40 ft, and has an operating weight in GEO of 133,400 lb (without orbit transfer propulsion). This design will be used as a model for all subsequent discussions.

The linear-shaped configuration of the platform with a length-to-width ratio of 20:1 is typical of platform designs resulting from certain operational constraints and other desirable attributes, and consisting of: 1) construction in LEO from the Space Shuttle Orbiter; 2) utilization of the Orbiter's remote manipulator system (RMS); 3) number of construction operations minimized.

The use of other construction options, such as the space-fabricated tri-beam design concept, results in platform designs that are comparable in configuration, size, and weight.

Propulsion Comparisons

Various types of propulsion systems were sized for performing the orbit transfer of an advanced communications platform in the 100,000-lb weight category from LEO to GEO. These systems consisted of solar electric propulsion (SEP), inertial upper stage (IUS) large solid motor, and storable and cryogenic propulsion modules. Thrust-to-weight ratio characteristics of these systems vary from less than 10^{-4}

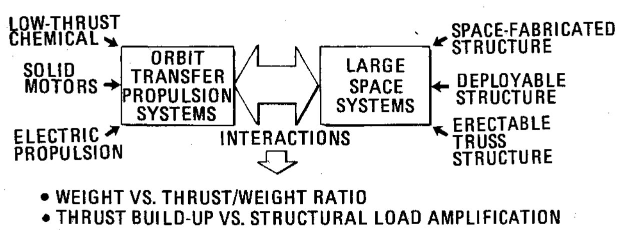


Fig. 1 Structural/propulsive interaction diagram.

Received July 23, 1979; revision received June 5, 1980.

Index categories: Spacecraft Configurational and Structural Design (including Loads); Spacecraft Propulsion Systems Integration.

*Member of Technical Staff, Propulsion, Test Requirements & Fluid Systems, Space Operations and Satellite Systems Division. Member AIAA.

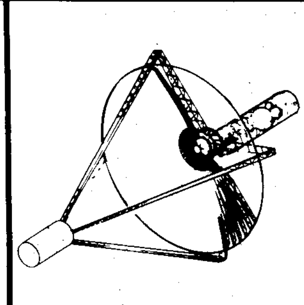
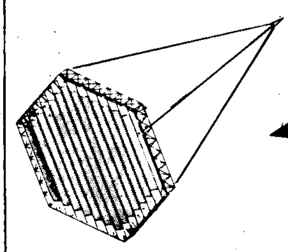
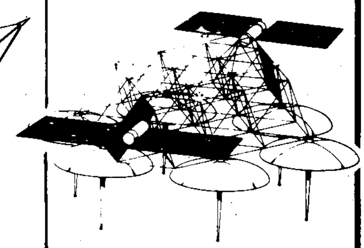
	 LARGE OPTICAL SPACECRAFT	 OOA	 EMS
SPACECRAFT WEIGHT (LB)	160,000	23,000	47,000 85,000
DIAMETER (FT)	100 200	730 1500	200 300
T/W	0.1 - 0.5	0.1	0.1 - 0.7
OPERATIONAL ALTITUDE (N.MI.)	1,000	GEO	GEO
ASSEMBLY ALTITUDE	LEO	LEO	LEO
OTV - NUMBER	1	3	1+
THRUST (LB), EACH	6,000	870	5,000 - 9,000
NO. THRUSTERS	2	24	2 - 3
PROPELLANTS	N_2O_4/MMH	N_2O_4/MMH	LO_2/LH_2

Fig. 2 Large space system characteristics.

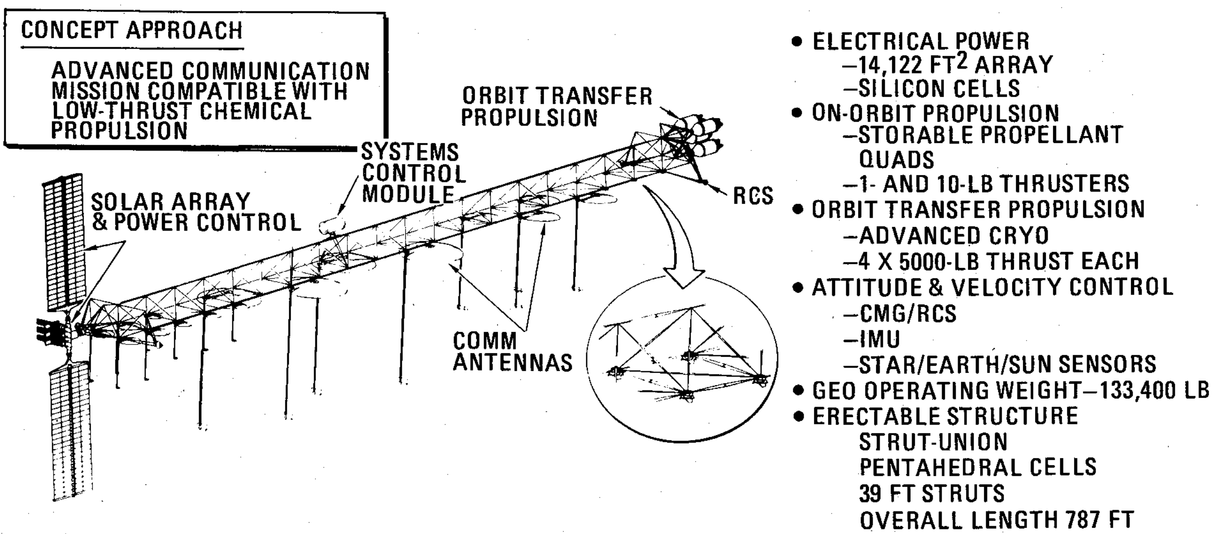


Fig. 3 Advanced communications space platform.

for SEP to 1.15 for the IUS. The required velocity increments as a function of T/W are shown in Fig. 4. This curve shows the velocity requirements for orbit transfer from a 28.5 deg inclined LEO of approximately 150 n.mi. altitude to GEO. For a T/W range of 0.1 to 0.2, a ΔV of 14,400 ft/s is required—based on a two-impulse burn and Hohmann transfer. It should be noted from the curve that ΔV requirements at T/W values less than 0.1 increase greatly, which is due to larger gravity losses ocurring with the longer trip times associated with less acceleration.

The system performance used in the sizing was based on the specific impulse of 5000 s for SEP, 295 s for IUS, 290 s for storable, and 467 s for cryogenic systems. For this comparison, considerations included the number of propulsion modules required, the number of orbiter launches required to bring this number of modules to LEO, any impact on the solar array operation or size, and any acceleration impact on the antennas during orbit transfer.

The results of this comparison are summarized in Table 1. Based on the number of modules required and the associated cost considerations of orbiter launches required, the cryogenic propulsion module and the SEP are clearly superior. Impacts on the solar array include the hinging requirement to allow folding of the solar array panels during the orbit transfer acceleration imparted by the cryogenic, storable, and the IUS systems. Another impact includes the more than doubling of the solar array panel area required with use of SEP to account for a 50% cell degradation due to radiation damage from long exposure to the Van Allen radiation environment, and to provide adequate power for reasonable trip times which can result in greater power required than for the communications function alone in GEO. Use of SEP does not impact the antennas due to acceleration, so that they may be fully deployed during orbit transfer; however, acceleration from the IUS requires that the antennas be retracted to a stowed position, which introduces an element of risk in requiring

Table 1 Orbit transfer propulsion comparison

	Cryogenic	Storable	IUS[a]	SEP
I_{sp}, s	467	290	295	5000
$(T/W)_{max}$	0.2	0.2	1.15	$<10^{-4}$
Number of modules[b]	4	11	20	N/A
Number of launches[c] (dedicated propulsion)	4	11	10	<1
Impact on solar array		Hinging required		>2×area
Antenna deployment for orbit transfer	Partial deployment		No deployment	Full deployment

[a] IUS first-stage modules
[b] Propulsion modules required for 100,000-lb platform
[c] Based on Shuttle Orbiter payload bay capacity

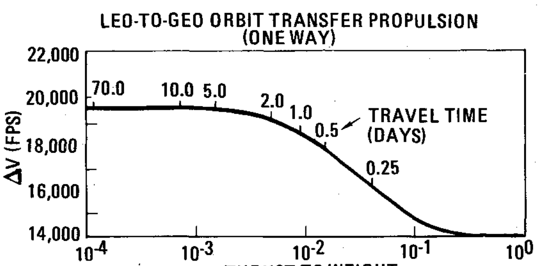


Fig. 4 ΔV requirements vs T/W ratio.

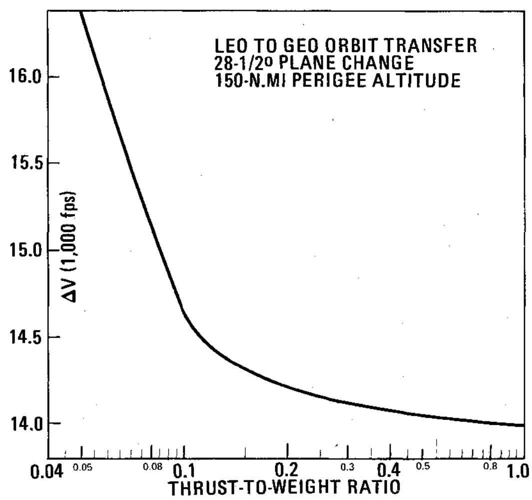


Fig. 5 Expanded scale of ΔV requirements vs T/W ratio.

automatic deployment in GEO rather than LEO. A moderate acceleration from the cryogenic and storable systems results in partial deployment (feed horns retracted, but with reflectors unfolded). Overall, both the SEP and the cryogenic propulsion modules are considered applicable for orbit transfer functions, and more detailed cost studies will be required to evaluate these systems and the effects of long trip times and solar array size for particular applications. However, the remainder of this paper will focus only on the cryogenic propulsion module for illustrative purposes.

Cryogenic Propulsion Module

It is desirable to utilize the largest module possible within the constraints of the Shuttle cargo bay in order to minimize the number of orbiter flights. A maximum gross weight for the orbit transfer propulsion module of 63,500 lb was established. The required number of modules is determined by the platform weight requirements, the velocity increments for orbit transfer, and propulsion specific impulse values. Propellant off-loading can be used in matching the platform weight requirements with the basic module. A platform weight of 133,400 lb is used as an example for the following discussions.

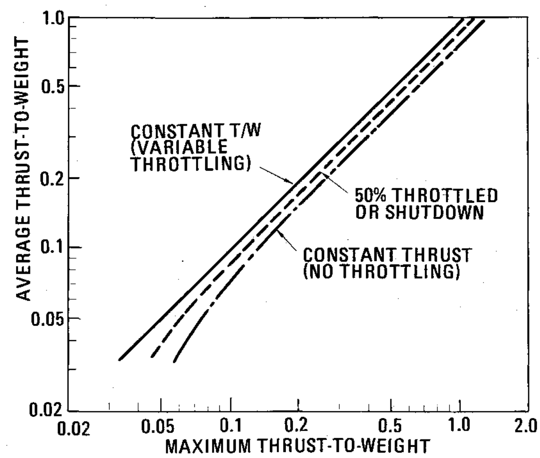


Fig. 6 Maximum vs average T/W ratio throttling effects.

ΔV vs T/W

The effect of T/W on ΔV was shown over a wide range of values in Fig. 4. For application to the cryogenic propulsion module, a narrower range of T/W is of more interest, such as that shown plotted to an expanded scale in Fig. 5. It is recognized that the single perigee burn data accentuates the effect at T/W values less than 0.1, compared to multiple burn effects; nevertheless, the trend would remain the same—that is, a marked increase in velocity increment is required at values less than 0.1 g.

In addition, it should be noted that for determining the propellant weight requirements, the ΔV requirements used here are expressed in terms of an *average* T/W value. Average refers to an average between the initial and burnout conditions, whereas the *maximum* T/W value (usually burnout conditions) is of interest to the structural designer. A correlation between average and maximum T/W values for a typical platform system is illustrated in Fig. 6, which also relates this correlation to generic types of engine thrust control. Variable throttling to low levels would be required if a constant T/W is needed. The maximum T/W value will determine the thrust requirements at burnout, and affect engine requirements for either the use of multiple engines with sequential shutdown to control T/W as propellants are consumed, or the use of fewer engines but with throttling requirements for T/W control.

Weight vs T/W

While too low a value of T/W yields excessively high ΔV requirements, too high a T/W can penalize the spacecraft in the area of structural weight. A small regime of appropriate T/W is indicated, to avoid both the propellant and system weight penalties of high ΔV (T/W too low), and the structural weight penalty of too large a T/W value. This is illustrated for a large advanced communications satellite (fabricated of graphite-composite material) in Fig. 7, which shows the satellite weight as a function of the maximum design T/W for two types of structure. When this T/W effect

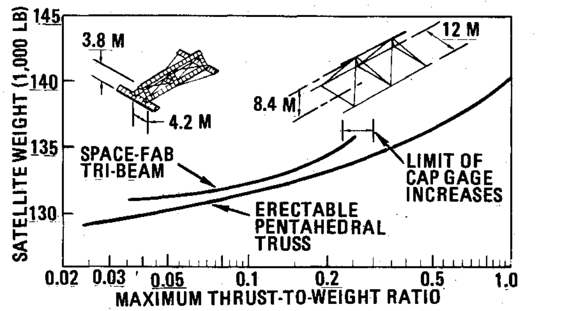


Fig. 7 Communications satellite weight vs maximum T/W ratio.

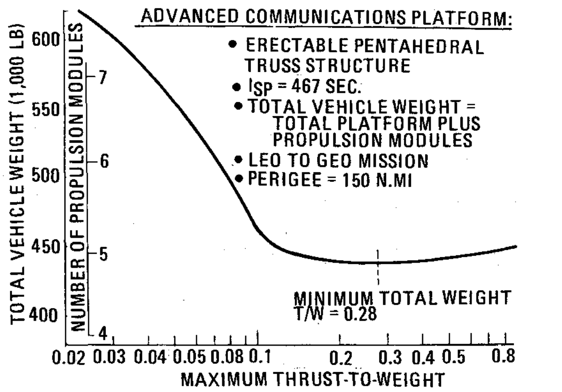


Fig. 8 T/W ratio impact on total vehicle weight.

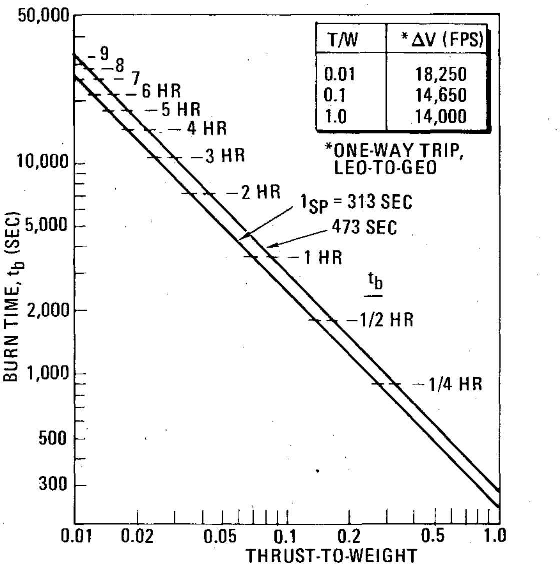


Fig. 9 T/W ratio effect on engine burn time.

is combined with the ΔV effect, an optimum T/W is indicated at 0.28, as shown in Fig. 8, for the erectable pentahedral truss-type structure. A maximum design T/W of 0.2 was selected for this particular spacecraft. The curve is relatively flat between T/W values of 0.2 and 0.4 g. A total of five propulsion modules is also indicated. The study also indicated the optimum T/W is highly dependent on the type of structure selected and the direction in which the thrust is applied.

Burn Time vs T/W

The burn time of low-thrust engines operating on long missions at low T/W ratios will impose a new technology requirement for future space engine developments. For example, the effect of T/W on engine burn time for the example LEO-to-GEO one-way mission is illustrated in Fig. 9. At T/W values of 0.1 to 0.2, burn times of 1200 to 2200 s are required. Thus far, burn durations per mission of this magnitude have not been required for bipropellant thrusters, although lower performance, low-thrust, monopropellant thrusters have fired for hours, such as for the ATS-6 satellite. The actual burn times, including effects of throttling or engine shutdown, would be increased somewhat over those shown on the referenced figure, which was based on a simplified approach. This example, however, does illustrate how burn time is affected by mission-related parameters of T/W and ΔV.

Propellant Storability

Propellant storability is a requirement for the entire elapsed time from propellant tanking to burnout. The use of cryogenic propellants requires adequate insulation for tanks to minimize boiloff propellant losses. Transit times to LEO and, subsequently, to GEO are relatively short (measured in hours), so that the elapsed time that impacts boiloff the greatest is the time required in LEO to accumulate the necessary number of propulsion modules. This elapsed time may be on the order of eight weeks, based on the following simplified scenarios:

1) A single Space Shuttle Orbiter is dedicated to the construction of the platform spacecraft.

2) Multiple flights are required to transport material and subsystem modules to LEO.

3) The orbiter requires a two-week turnaround period between flights.

4) A total of five propulsion modules are required, and this determines the orbiter flights required to transport them to LEO.

From this example, it can be seen that the fifth module arrives in LEO eight weeks after the first module. Contingencies and margin allowances may be accounted for by assuming all modules have eight weeks of propellant boiloff, although only the first module has experienced the entire eight-week holding period.

Insulation concepts from prior studies of LO_2/LH_2 propulsion modules include the use of multilayer insulation externally applied to nonintegral propellant tanks that are supported within an outer shell by fiberglass struts that act as heat blocks.[3-5] Insulation materials such as layers (3/4 to 1 in. total thickness) of double aluminized Mylar and use of fiberglass tank supports will limit boiloff rates to 2.3 lb/h for tanks designed to contain 56,000 lb of LO_2/LH_2. This results in boiloff rates of 0.7% per week with 265 lb of insulation. For the eight-week holding period, a boiloff allowance of 6% was used with the above insulation concept. Alternative insulation materials, such as layers of double goldized Kapton, would provide lower boiloff rates and less insulation weight for an increase in material costs. This latter concept may prove more desirable for single-module applications with reusable requirements.

Thrust Buildup

The rate of application of the propulsive thrusting force to "soft" structural platform designs is of importance to avoid excessive weight penalties. The thrust rise time during the engine starting transient must be long enough to keep the structural load amplification factor as close to a value of 1 as possible, since a value of 2, for example, could result in a 25-40% structural weight increase of the vehicle. The interaction between the engine thrust rise time and the amplification of structural loading is shown in Fig. 10. Amplification factors varying from 1 to 2 can occur, depending on the shape of the thrust/time relationship (see inset), and the ratio of the period of thrust buildup τ to the period of the structure T.[6] The structural period T is associated with the lowest modal frequency. The lowest amplification factor results from a linear thrust rise shape, $a=0$. For a linear buildup of thrust in time τ, and with the function τ/T greater than 2, the amplification factor will not exceed a value of 1.2.

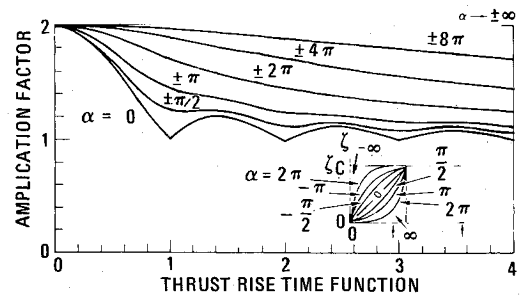


Fig. 10 Thrust rise time effects on amplification factor.

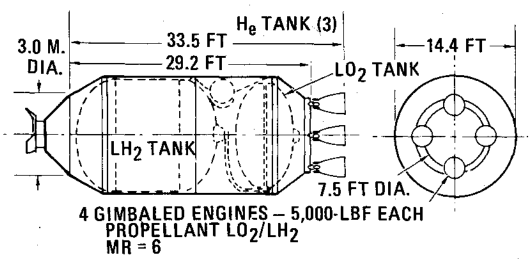


Fig. 11 Low-thrust propulsion module.

Module Description

The low-thrust propulsion module is shown in Fig. 11 with some details and overall dimensions. A single oxidizer tank, fuel tank, and helium pressurization gas tanks are located within a structural shell that acts as a micrometeoroid shield. The design features the use of nonintegral propellant tanks with multilayer insulation for control of boiloff. Tank supports are fiberglass struts. Based on prior studies, an allowance for 1-in. thick MLI would control boiloff of LO_2/LH_2 propellants to 0.7% per week of on-orbit holding time. An allowance of 6% boiloff was assumed for an eight-week period required to transport all five modules to LEO.

The overall dimensions of the module are compatible with orbiter payload bay size, and the overall length is within the 35-ft length target for module design. This is accomplished in part by the use of multiple 5000-lb thrust engines which are short, and which eliminate the need for nozzle retraction mechanisms.

The design weight summary with maximum propellant loading is shown in Table 2. The inert weight includes allowances for subsystems such as structure, thermal control, avionics, propulsion, residual fluids and contingencies, based on prior studies of Space Tug and Orbit-to-Orbit Shuttle (OOS).[3,5] With maximum propellant loading, the five modules are capable of transporting a maximum payload of 140,000 lb from LEO to GEO. The propulsion module may be offloaded to accommodate lower payload weight.

Each of the four engines include provisions for two-axis gimballing for thrust vector control (TVC). The engine is a staged-combustion design based on the technology development of the Advanced Space Engine.[7] The performance and size of the 5000-lb_f thrust engine is summarized in Table 3.

Module Cluster Arrangement

A cluster of five low-thrust propulsion modules is provided for the orbit transfer of the antenna platform. This cluster configuration is shown in Fig. 12. A thrust structure is provided for transmitting thrust loads from the five modules into the platform structure. Module mating to the thrust structure is via docking ports.

The five propulsion modules are operated in a 3-2 firing/staging sequence. The total firing time of each module is approximately 20 min. at full four-engine thrust per module. In actual practice, durations slightly longer will result when paired engines are shut down to control T/W, and

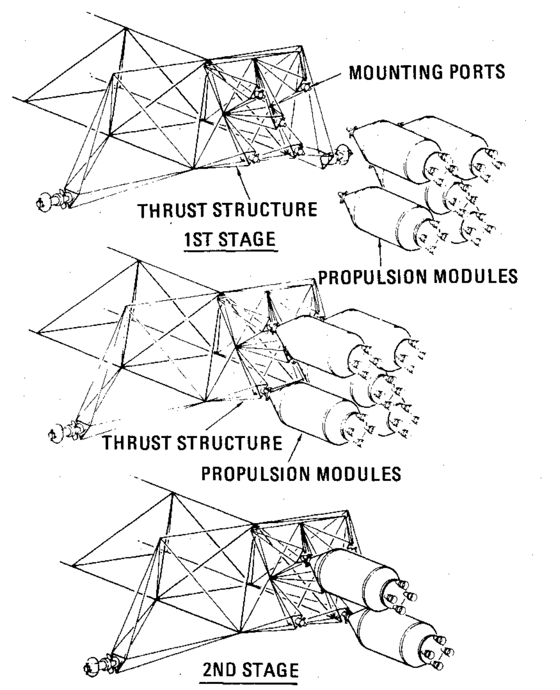


Fig. 12 Module cluster arrangement.

Table 2 Low-thrust propulsion module weight summary

Item	Weight, lb
Maximum gross weight	63,500
Maximum propellant load	55,815
Inert weight	7,685
Stage mass fraction	0.879
Propellant boiloff (6%)	3,349
Usable propellant (after boiloff)	52,466

Table 3 Engine performance summary

Item	Value
Thrust, lb	5000
Chamber pressure, psia	1500
Nozzle expansion area ratio	400:1
Propellants	LO_2/LH_2
Mixture ratio (O/F)	6:1
Specific impulse, s	467
Overall length, in.	52
Nozzle exit diameter, in.	30
Weight, lb	110

when sequential startup and shutdown by engine pairs are done in 10-s intervals to reduce the dynamic amplification of the platform structure during these thrust load transients.

The initial three modules require a single firing for the perigee burn, then are stage off at burnout, and the remaining two modules are fired to achieve the remaining perigee burn ΔV. A second start for the two modules is then required for the apogee burn to circularize the orbit at GEO.

With multiple engines per module, the T/W is controlled to remain below 0.2 by sequential shutdown of engines in pairs. For example, during the three-module burn, the initial T/W is 0.14, with all engines firing (60,000 lb_f thrust), and it remains below 0.2 until just prior to burnout; then, if two engines are shut down on each module, the T/W would be reduced to 0.11 at burnout. For the remaining two-module burns made subsequently, the initial T/W is 0.16 with all engines firing (40,000 lb_f thrust), and it remains below 0.2 for the initial half of the burn ΔV, then two engines per module are shut down to bring the burnout T/W to 0.13. During the orbit transfer

Table 4 **Module cluster summary**

Item	Value
Propellants	LO_2/LH_2
Total impulse, lb-s	1.16×10^6
Number of modules	5, parallel
Firing/staging sequence	3/2 modules
Number of engines	20
Thrust (each), lb_f	5000
$(T/W)_{max}$	0.2
Ignition weight (each), lb	57,303
Five modules,[a] lb	286,515
Boiloff, %	6
Loaded weight (each), lb	60,280
Five modules, lb	301,400

[a] For antenna platform weight of 133,400 lb

Table 5 **Conclusion summary**

Area	Characteristics
1) Minimum weight	Low T/W (0.1-0.3)
2) Low T/W	Long engine burn times (1000-2000 s)
3) Minimum structural load amplification	Long thrust rise time, engine throttling, multiple low-thrust engines
4) Storability of cryogenic propellants for long periods (60 days)	Multilayer insulation, fiberglass tank supports, boiloff losses 0.1% per day
5) Multiple low-thrust engines	Facilities T/W control, TVC, minimum structural load amplification

phase, the antenna feed horn assemblies are in a retracted position, and the solar arrays which have been deployed are in a tied-down folded position for acceleration of this phase.

Thrust vector control is provided by the gimballed engines which are ganged in pitch and yaw and differently gimballed with the outer modules for roll control. This mode of TVC with the multiple engine configuration provides control under all conditions of 3-2 module staging and paired engine operation for T/W control or structural dynamic deamplification. Multiple engines provide the necessary flexibility for meeting these varied conditions.

Other characteristics of the module cluster are listed in Table 4, which also shows the offloaded propellant condition of the low-thrust propulsion module.

Conclusions

In examining orbit transfer propulsion for transporting large space systems to high orbit, a number of conclusions follow from an analysis of the structural/propulsive interactions, the operational modes and requirements, and the system design processes. These conclusions, summarized in Table 5, are as follows:

1) Minimum weight for the total system is obtained at low thrust-to-weight values between 0.1 and 0.3 for the structure type and spacecraft geometry assumed during this study.

2) Long engine burn times (1000-2000 s) per mission are required when low thrust-to-weight ratios are necessary.

3) Minimizing structural load amplification requires relatively long engine thrust rise times, which implies that engine throttling together with the step-throttling of a number of low-thrust chambers, is required.

4) The storability of large quantities of cryogenic propellants in orbit for long periods, such as 60 days, appears feasible by using already developed techniques with multilayer insulation and fiberglass tank supports, and which should limit boiloff losses to 0.1% per day.

5) Multiple low-thrust engines facilitate thrust-to-weight ratio control as propellants are consumed by use of sequential engine shutdown of opposing pairs. Thrust vector control also is facilitated by use of multiple engines that are two-axis gimballed.

References

[1] Proposal for a "DOD/STS On-Orbit Assembly Concept Design," Satellite Systems Division, Rockwell International Corporation, Downey, Calif., SD 77-AP-0056-1, Vol. 1, July 1977.

[2] "Space Construction System Analysis," Satellite Systems Division, Rockwell International Corporation, Downey, Calif., PD-79-08, March 1979.

[3] Hanley, G.M., et al., "Orbit-to-Orbit Shuttle (Chemical) Feasibility Study," Space Division, North American Rockwell Corporation, Downey, Calif., SAMSO TR-71-238, Vol. II-2, Oct. 1971, pp. 3-55.

[4] "DOD Upper Stage/Shuttle System Preliminary Requirements Study," Briefing Manual, McDonnell Douglas Astronautics Co., Huntington Beach, Calif., SAMSO TR-72-202, Aug. 1972, p. 168.

[5] "Baseline Space Tug Configuration Definition," MSFC 68M00039-2, NASA, July 1974, pp. 75, 76, 94-96.

[6] Ayre, R.S., "Transient Response to Step and Pulse Functions," *Shock and Vibration Handbook*, 2nd Ed., McGraw-Hill, New York, 1976, pp. 8-20.

[7] "OTV, Large Space Stuctures and Other Space Applications, Propulsion," Rocketdyne Division, Rockwell International Corporation, Canoga Park, Calif., BC 78-92, June 1978.