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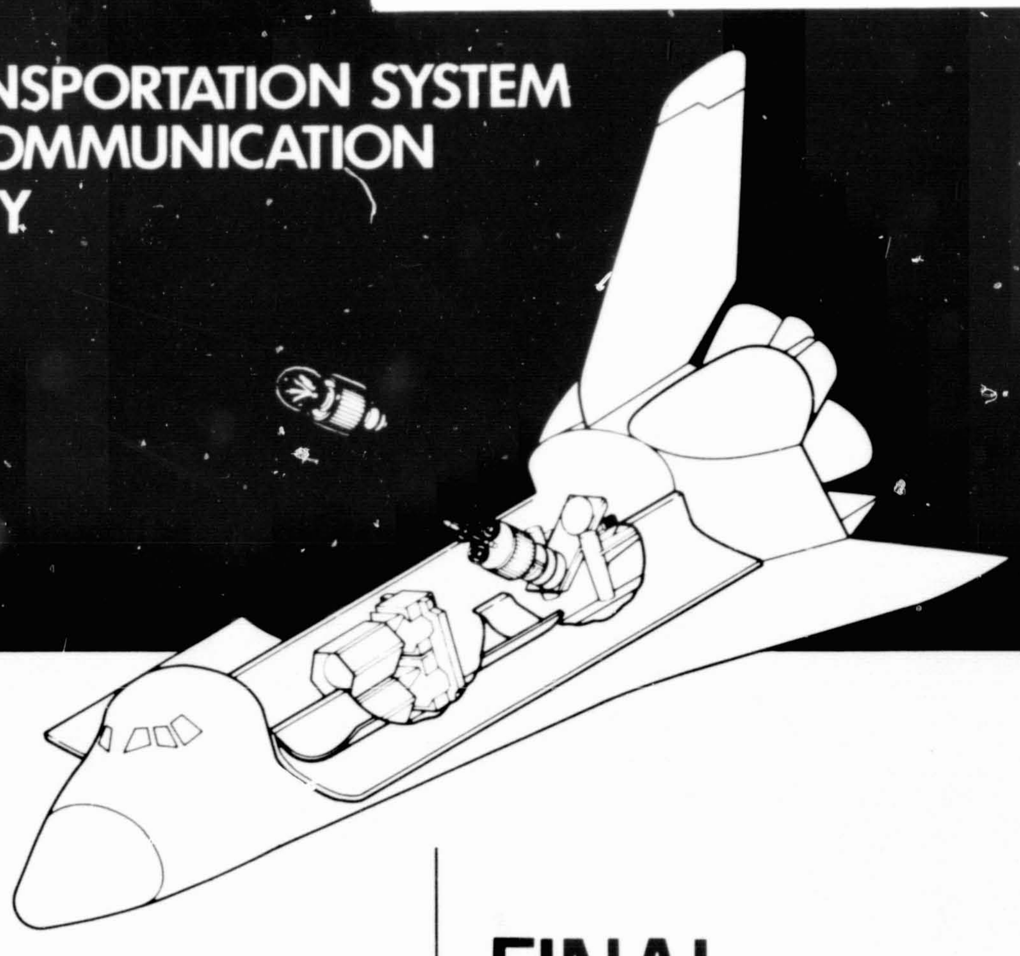
UTILITY OF SPACE TRANSPORTATION SYSTEM TO SPACE COMMUNICATION COMMUNITY



FINAL REPORT

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE AND COMMUNICATIONS GROUP

MARSHALL SPACE FLIGHT CENTER

Data Procurement Document No. 514
Data Requirement No. MA-04

Hughes Ref. No. D5221-2
SCG 50314R

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1. INTRODUCTION

The utility of the Space Transportation System to the space communication community depends on the service and the cost. The space communication community exists because it provides a useful service at a competitive cost.

Commercial space communications has existed slightly more than 10 years, beginning with the launch of Early Bird (Intelsat I) in 1965. During this time, the business expanded in number of in-orbit communication satellites, in service categories from purely international to national, and in user type from fixed point to mobile platform such as ships, airplanes, and spacecraft. This business growth results from the evolution of service-oriented system designs, which include the satellites, the earth and mobile-platform terminals, and the current expendable launch vehicles. Finally, these satellite system services continued to be offered to users at a competitive price.

The Space Transportation System (STS) offers the opportunity for maintaining, and perhaps accelerating, the growth of the space communication user community; however, to sustain this growth the new launch vehicle service must be available at a cost lower than the current expendable launch vehicles cost.

This report describes the results of the Hughes Aircraft Company, Space and Communications Group study contracted by NASA Marshall Space Flight Center on the "Utility of the STS to the Space Communication Community." The purpose of the study was to analyze a potentially cost effective technique of launching operational satellites into synchronous orbit using the STS. This technique uses an unguided spinning solid rocket motor (SRM) as the means for boosting a satellite from a low altitude shuttle parking orbit into a synchronous transfer orbit. The spacecraft is then injected into a geosynchronous orbit by an apogee kick motor (AKM) fired at transfer orbit apogee. The approach is essentially that used on all Delta and Atlas-Centaur launches of synchronous satellites with the shuttle orbiter performing the function of the first two stages of the Delta three stage launch vehicle and the perigee kick motor (PKM) performing the function of the Delta third stage.

The study concludes that the STS can be useful to the space communication community as well as to other geostationary satellite system users if NASA implements the recommended actions.

1.1 EFFECT OF USING PKM ON SHUTTLE TO SPACE COMMUNICATION COMMUNITY

Using a spinning SRM for perigee injection offers two potential ways to reduce synchronous launch costs. First, the volume of the payload vehicle will be minimized because the solid perigee stage is more compact than an equivalent liquid motor and because the AKM can be integrated with the spacecraft. A minimal payload volume will significantly improve multiple launch capability in the shuttle payload bay that will more often be volume limited than weight limited. Four or more Delta-class synchronous satellites, each with a PKM and AKM, can fit in the shuttle payload bay. Alternately, synchronous payloads can be launched with payloads that remain in the orbiter payload bay. Multiple launch on an upper stage requires payloads with similar orbit requirements.

The second potential source of cost reduction of synchronous launches is the low cost of the spin stabilized solid perigee stage. The cost of the procurement and integration into the shuttle of two solid rocket motors (PKM and AKM) is considerably less than that of an equivalent liquid motor. Also, the cost and complexity of an upper stage inertial guidance system (which will be shown to be unnecessary for synchronous launches) can be eliminated, although other equipment whose cost must be assessed will be required in its place. Because the SRM is relatively inexpensive it is reasonable that motors sized for several payload classes can be developed. Then a synchronous payload will utilize a share of the shuttle orbiter capability commensurate with its size.

1.2 DELTA LAUNCH SEQUENCE PATTERN FOR STS SEQUENCE

The NASA Thor Delta launch vehicle service is employed in the 1970 to 1980 period by approximately 70 percent of the geostationary payloads and approximately 50 percent of the commercial communication satellites. The preference for Delta results from two factors: 1) Delta costs are less than half the cost of the next larger launch vehicle, Atlas-Centaur, and 2) the Delta payload capability into geostationary orbit matches the requirements of many users. In summary, for a reasonable investment a Delta-launched satellite system can provide a useful service in a competitive market. The Delta proven launch sequence provides a model for a cost competitive STS (orbiter and upper stage) sequence (Figure 1).

The Thor Delta with strap-on SRMs and first and second stage engines places the Delta second and third stage plus payload into a nominal 100 n.mi. (185 km), 28^{o} inclination, circular parking orbit. The STS orbiter with strap-on solids and orbiter engines places the orbiter plus payload in a nominal 160 n.mi. (296 km), 28^{o} inclination, circular orbiter altitude orbit. With either the Delta or the orbiter, the next step is to prepare to inject the payload into an elliptical transfer orbit.

The injection into elliptical transfer orbit with the Delta is performed at the first equatorial crossing (the desired location of transfer orbit perigee). Transfer orbit injection is accomplished by the following sequence. A short second burn of the Delta second stage is followed by spinup of the third stage and payload in a second stage mounted spin table, and separation from the second stage and burn to depletion of the third stage solid rocket motor. Then, the payload, which is separatated from the Delta third stage, is in its transfer orbit with the perigee and apogee on the equator with a nominal 28° inclination and with the apogee at a nominal 19,400 n.mi. (35,800 km) synchronous orbit altitude.

The STS orbiter can use the same sequence by spinning up the payload vehicle consisting of an SRM upper stage and its payload, ejecting the payload vehicle at the proper time so that the PKM would fire at an equatorial crossing. The payload would separate from the PKM stage in a transfer orbit with perigee and apogee on the equator, with a nominal 28° inclination and with the apogee at a nominal 19,400 n.mi. (35,800 km) synchronous orbit altitude.

From the moment the third stage is ignited, the mission becomes identical to that of a Delta-launched spacecraft.

The Atlas-Centaur launch vehicle employs a similar sequence using a second burn of the Centaur for the same type transfer orbit injection. The spacecraft is spun up after separation by spacecraft mounted jets.

All commercial communication satellites, both spinners and three-axis, launched in the 1970 to 1980 period are placed in geostationary orbit from a spinning transfer orbit. During transfer orbit, the spacecraft orbit is determined using the spacecraft radio signal, and appropriate apogee firing parameters are computed. Also, the spacecraft attitude is determined on the ground using telemetered data from the spacecraft spinning sensors. An attitude maneuver is commanded to orient the AKM. This maneuver is performed by pulsing the spacecraft RCS precession jets at the proper spin phase. The jet pulses can be commanded on the ground using telemetered pulses from the spinning sun sensor and a ground-installed synchronizer. At the appropriate apogee (usually between the second and ninth), the apogee motor is fired to circularize the orbit and remove the inclination. After apogee injection, the spacecraft RCS is used to correct the orbital errors resulting from perigee or apogee firing. Three-axis spacecraft must despin and acquire proper orientation before beginning on-orbit operations.

This report discusses the perigee stage and orbiter hardware required to accomplish the perigee sequence for STS and the impact of this form of perigee injection and transfer orbit on potential geosynchronous operational payloads.

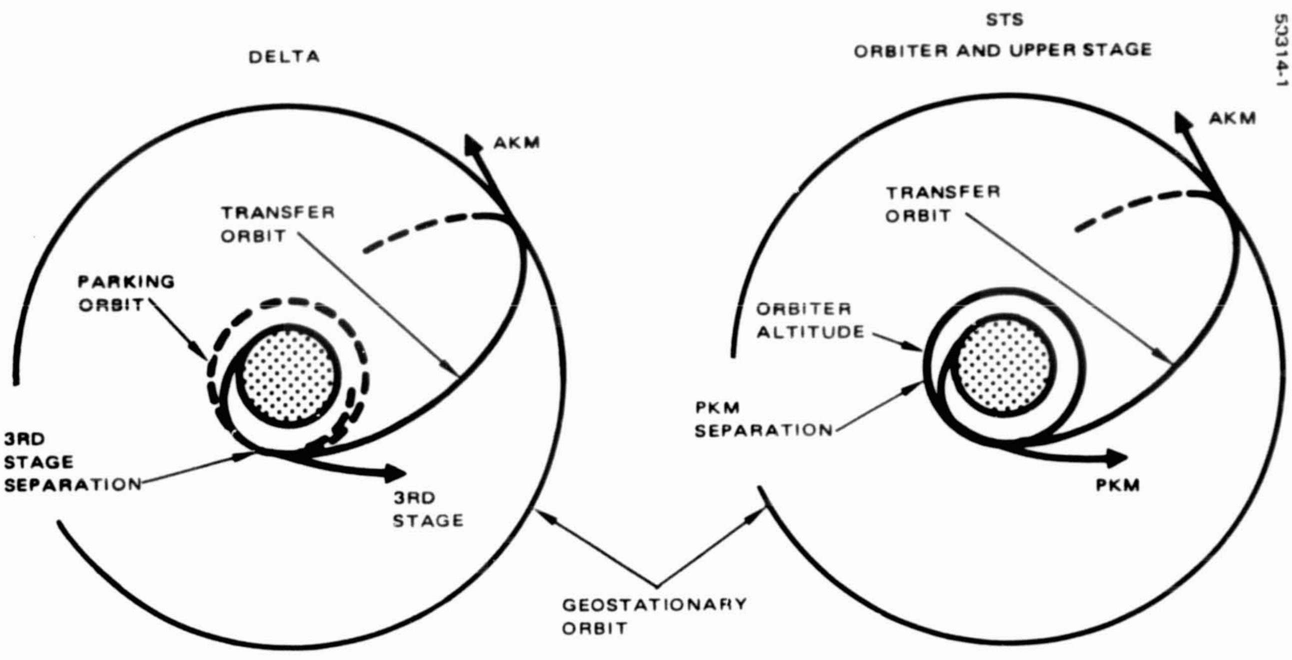


FIGURE 1. DELTA LAUNCH SEQUENCE PATTERN FOR STS SEQUENCE

2. STUDY OBJECTIVES

Specific objectives of the study are:

1) Determine the extent to which this technique can serve the shuttle geostationary mission model. The source provided for the model was the "Summarized NASA Payload Descriptions" of July 1974; however, some comments on the model are made in the next section.

2) Determine the impact on the spacecraft design and cost of launching the satellite with a spinning transfer orbit and AKM rather than boosting it into geosynchronous orbit with a three-axis stabilized vehicle.

3) Define the design characteristics and cost of the perigee stage.

4) Define an optimum technique for installing the payload and deploying it from the shuttle orbiter payload bay.

5) Analyze the accuracy of the approach and the impact of errors on synchronous payload capability.

6) Estimate cost associated with launching synchronous satellites with the shuttle PKM.

3. MISSION MODEL

3.1 GEOSTATIONARY SPACECRAFT

In order to generate a realistic model of the geostationary payload model for the first decade of shuttle operation, the geostationary payload model in the NASA STS Payload Data and Analysis (SPDA) document was compared to the list of geostationary payloads already launched or under construction for launch in the 1970 to 1980 period. It seems reasonable that the payload model for the next decade should represent a reasonably smooth growth from that of the present decade.

Data pertinent to this comparison are summarized in Table 1. The data are characterized by mission, launch vehicle purchaser, and payload class (Delta, Centaur, and Titan).

TABLE 1. GEOSTATIONARY SPACECRAFT
(1970 TO 1980 VERSUS 1979 TO 1991)

Mission	Launch Vehicle Purchaser	Payload Class					
		Delta		Centaur		Titan	
		70 to 80	SPDA	70 to 80	SPDA	70 to 80	SPDA
Scientific and experimental	N-rocket	4	NA	NA	NA	NA	NA
	Reimbursable	10	0	0	0	0	0
	NASA	2	0	0	8	1	3
Earth observation	N-rocket	0	NA	NA	NA	NA	NA
	Reimbursable	7	15	0	0	0	4
	NASA	2	0	0	0	0	14
Communication	N-rocket	0	NA	NA	NA	NA	NA
	DoD	0	NA	2	NA	8	NA
	Reimbursable	21	29	18	6	0	35
	NASA	0	0	0	4	0	0
Total	Reimbursable	38	44	18	6	0	39
	NASA	4	0	0	12	1	17
		1970 to 1980				1979 to 1991	
Grand total	Reimbursable	56				89	
	NASA	5				29	

Two features of the mission model for the present decade are apparent. First, this decade is dominated by Delta launches of which there are 42 compared to 18 Centaur and one Titan. Second, most of the geostationary launches in this decade are reimbursable (i.e., commercial, other U.S. government agency, or foreign) rather than NASA launches.

As shown in the SPDA, a sharp change in the mission model is indicated. First, the reimbursable model shifts from Delta, which grows only from 38 in the present decade to 44 in the next, to the larger payloads (Centaur plus Titan), which grow from 18 to 45. This shift is not consistent with the present nature of the commercial and foreign market. Even if this shift should be realized almost half of all reimbursable launches are still Delta-class payloads.

The second change between existing and SPDA projected models is the shift to NASA missions. The current ratio of reimbursable to NASA launches is over 11:1 (56 versus 5). In the NASA planning, it drops to about 3:1 (89 versus 29). This shift is also suspect; however, in either case, the large number of reimbursable launches means STS must provide a service at competitive cost.

The appearance of a competitor to the NASA monopoly for spacecraft launching to synchronous altitude is also evident with the N-rocket plan for four launches in this decade. This fact is important in light of the large numbers of reimbursable launches expected in the 1979 to 1991 period.

3.2 PROJECTED REIMBURSABLE COST

The projections of Delta and Centaur cost for the 1980 to 1990 time period are plotted with an assumed inflation rate of 5 percent per year. The cost bases are (Figure 2):

1) Delta $12.9 million for 1976 launch quoted to Indonesia

2) Centaur $25 million for a 1976 launch quoted to COMSAT

The NASA cost objective for the cost of an orbiter flight was established as $10.5 million in 1971 dollars. This has been quoted as the NASA direct cost equivalent to the cost carried in the NASA accounting for NASA usage of expendable launch vehicles. An additional factor must be added to arrive at [illegible]ivalent reimbursable cost. For the expendable vehicles, this factor [illegible] currently from 60 to 100 percent of the direct cost. A 60 percent fa[illegible]as assumed for the orbiter flight cost.

The plot of the orbiter cost per flight assuming 5 percent inflation and the 60 percent factor shows that the cost of the orbiter alone, not counting the upper stage, is twice the cost of Delta and approaches the cost of Centaur. According to this projection even if the cost of the Delta were double that shown it would be lower in cost than STS. Foreign competitors could also be expected to offer a less costly launch service.

3.3 MULTIPLE LAUNCH STS PROJECTED COST

The STS with its large payload capability offers the opportunity for multiple payload launches.

The previous data are plotted against both two-spacecraft and four-spacecraft multiple launch cases (Figure 3). The upper stage is assumed to cost $1 million. Data presented later in the report indicate that $1 million is an upper bound for the PKM/AKM concept.

Delta-class users would be able to launch their spacecraft at half their current cost because the STS could easily accommodate four Delta-class users. This depends on NASA's ability to reach initially established objectives in the STS orbiter cost and NASA's provision for simultaneous accommodation of multiple users.

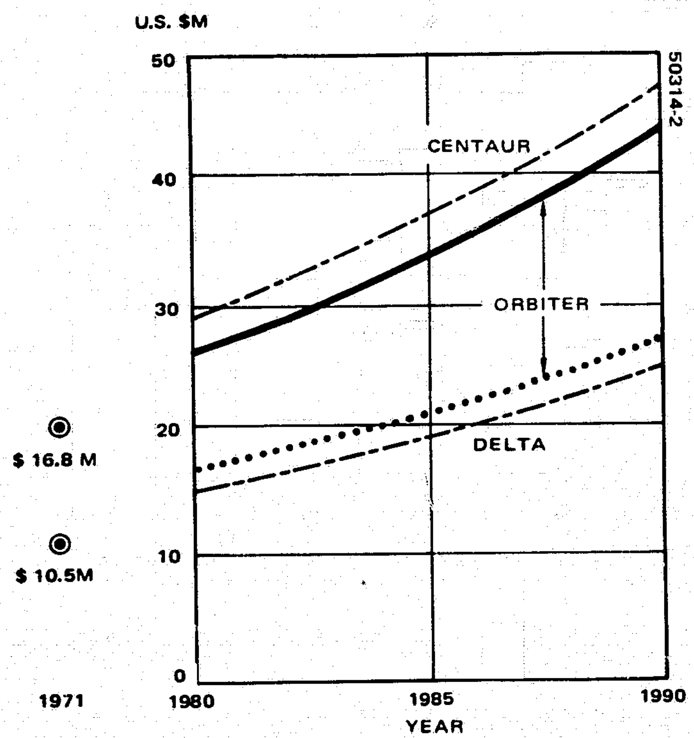


FIGURE 2. PROJECTED REIMBURSABLE COST

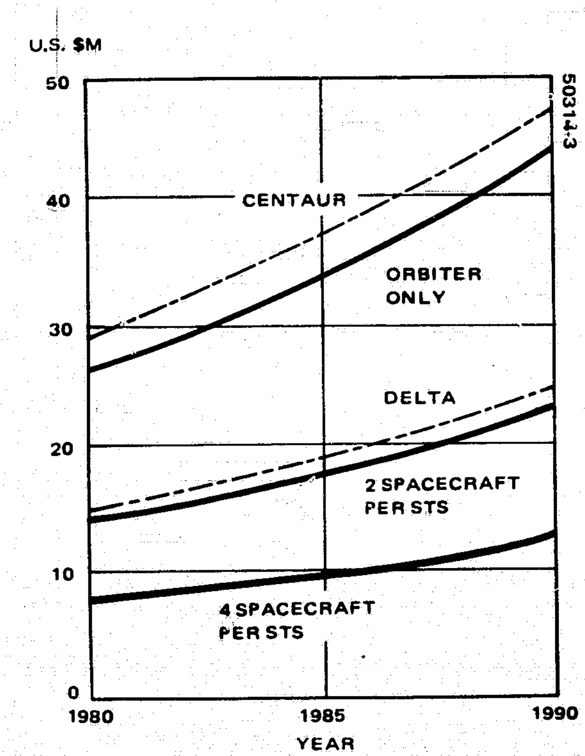


FIGURE 3. MULTIPLE LAUNCH STS PROJECTED COST

4. POSTPERIGEE FIRING PHASE – IMPACT OF SPINNING PKM ON SPACECRAFT

The baseline approach described in this report makes the perigee stage self-sufficient. Thus, the spacecraft is an inactive passenger prior to separation from the perigee stage. After separation, the spacecraft transfer orbit operations are identical to the transfer orbit operations on a Delta or Atlas-Centaur launched spacecraft. This is true for both spin stabilized and three-axis body stabilized spacecraft. Such three-axis spacecraft as the currently in-orbit Symphonie and the soon to be launched OTS, CTS, RCA DOMSAT, FLTSATCOM, and the Japanese Broadcast Satellite, all of which are designed to spin during transfer orbit and apogee firing, could be launched on a spinning PKM without modification (assuming that they are compatible with the shuttle ascent environment). Future spacecraft that have not yet been designed will require some equipment that would not be required if they were inserted into synchronous orbit by the Space Tug.

Areas potentially affected by a shuttle-PKM launch are:

1) Precession capability

2) Spin stability

3) Spinning attitude sensors

4) Spin balance and structure

5) Apogee kick motor

6) Transfer orbit solar power

7) Transfer orbit telemetry and command

These areas are discussed in the following sections. There is no thermal impact listed because spinning the spacecraft provides a benign thermal environment and reduces the complexity of spacecraft thermal control.

4.1 TRANSFER ORBIT REACTION CONTROL REQUIREMENTS (Figure 4)

Precession capability is required to reorient the spacecraft from the perigee firing attitude to the apogee firing attitude. This maneuver involves a rotation of about 120°. Also, active nutation control (ANC), which is required on those spacecraft that are unstable in transfer orbit, requires precession thrusters to reduce the spacecraft transverse angular rate. Three-axis spacecraft must also be despun after orbit insertion. All spacecraft designed for launch on Delta or Atlas-Centaur have the capabilities they would require on a spinning PKM launch because the transfer orbit associated with these boosters is indistinguishable (except in duration) from that of the shuttle-PKM launch. Thus, the use of a spinning PKM has no impact on the reaction control systems of these satellites.

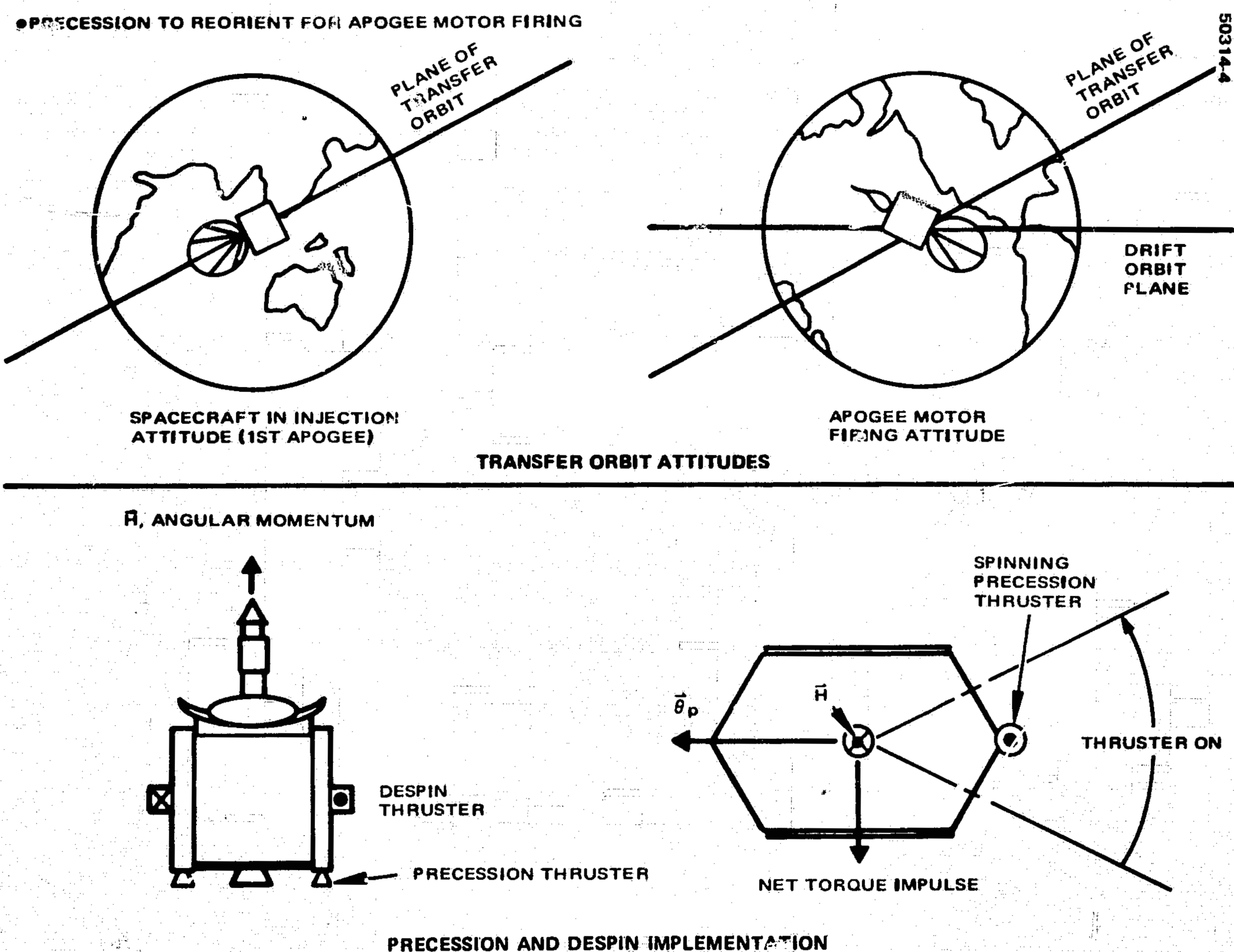


FIGURE 4. TRANSFER ORBIT REACTION CONTROL REQUIREMENTS

The impact on a spacecraft RCS of designing it for orbit insertion on a spinning PKM and AKM (or for a Delta launch), rather than on a Tug or other three-axis stabilized vehicle which places the spacecraft directly into synchronous orbit, depends on the characteristics of the spacecraft.

A spacecraft that is spin stabilized on orbit will usually be able to use the on-orbit precession capability for apogee firing orientation and ANC; however, in some cases ANC may require a higher thrust level than is suitable for on-orbit operation.

A spacecraft that is three-axis body stabilized on orbit will have thrusters that are properly oriented for transfer orbit precession (east-west stationkeeping thrusters) and for despin (north-south stationkeeping thrusters). On most three-axis spacecraft the thrust level of these thrusters will be adequate for apogee reorientation, but not for ANC.

The impact, then, of despinning an RCS for spinning orbit insertion is:

Spinners that are stable in transfer orbit	No impact
Most three-axis spacecraft that are stable in transfer orbit	No impact
Spinners that are unstable in transfer orbit	Thrust level of precession thrusters is driven by ANC requirements. Normally, this has no impact but, in some spacecraft, higher level thrusters may be added.
Three-axis spacecraft that are unstable in transfer orbit	A pair of high level thrusters must be added for ANC. The cost associated with this addition is $70,000 to $90,000.

4.2 SPACECRAFT STABILITY IN TRANSFER ORBIT (Figure 5)

In transfer orbit, the spacecraft will be spinning about the thrust axis of its perigee kick motor. If the inertia of the spacecraft about this spin axis is less than that about one of its transverse axes, the spacecraft will not be spin stable and will require active nutation control. Most spin stabilized spacecraft, such as Anik and WESTAR, are also stable in transfer orbit. All of the Delta- and Centaur-launched three-axis body stabilized spacecraft such as CTS, OTS, RCA DOMSAT, Symphonie, and FLTSATCOM now on orbit or under construction are spin stable in transfer orbit.

Spacecraft that will be unstable in transfer orbit are those that mount their AKM external to the spacecraft body and do not become stable until the AKM is separated, and spacecraft that are long and thin because of booster

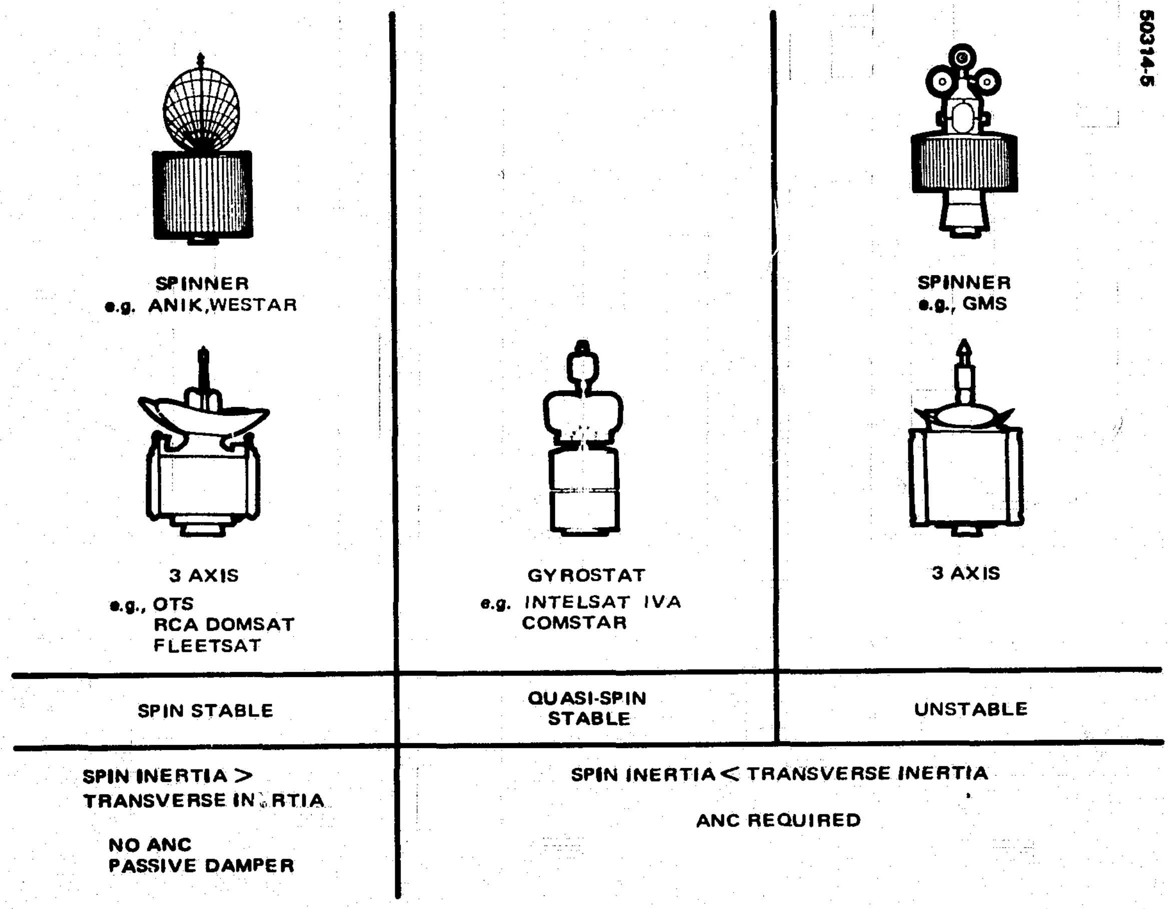


FIGURE 5. SPACECRAFT STABILITY IN TRANSFER ORBIT

shroud diameter limitations. Examples of the former are the meteorological satellites that must separate their AKM to expose a sensor cooler. Examples of the latter are the Gyrostats, such as Intelsat IV, IVA, and COMSTAR, which are stable in transfer orbit but can become unstable if large transient nutation occurs.

4.3 ACTIVE NUTATION CONTROL IMPLEMENTATION

The operation of a typical ANC system is shown in Figure 6. When the body spin axis nutates about the angular momentum vector, there is an angular rate about the body transverse axis. An accelerometer mounted on the outer periphery of the body with its sensitive axis parallel to the body spin axis measures a centrifugal acceleration that results from this transverse rate. The spin axis rotates about the angular momentum vector at a frequency less than the spin frequency. Each spin cycle, the accelerometer output will follow a sinusoid that peaks when the accelerometer is in a plane containing the angular momentum vector and the spin axis. When the accelerometer output exceeds the threshold, a jet located approximately 90° away

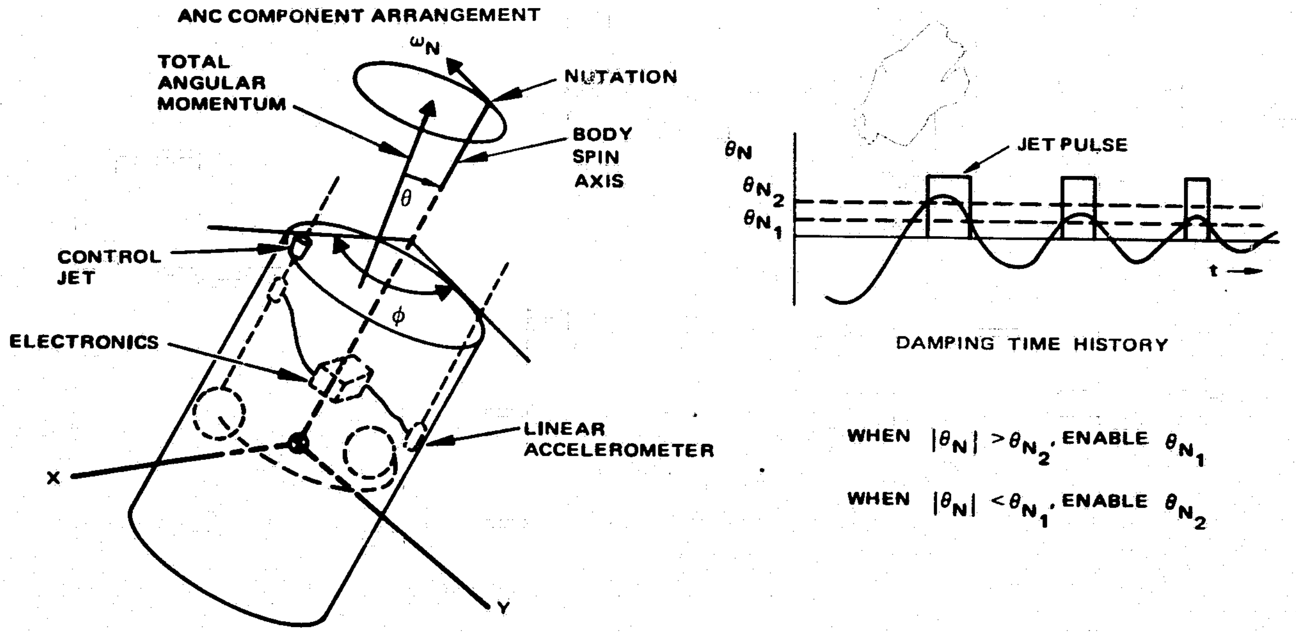


FIGURE 6. STABILITY IMPLEMENTATION

from the accelerometer, but with its thrust direction parallel to the accelerometer axis, will be fired to cancel the transverse rate. A dual threshold can be used to minimize the number of jet firings. Normally, the high threshold is active. Once the high threshold is exceeded, the lower threshold is turned on. When the signal drops below the low threshold, the system reverts to the high threshold. Periods of 10 minutes to 1 hour will occur between activations of the high threshold. The nutation will be reduced below the low threshold within seconds.

The ANC consists of redundant accelerometers, redundant electronics, and redundant thrusters. The accelerometers are off the shelf items costing from \$1,000 to \$2,000 when purchased in moderate quantity. For small orders, they can cost about \$5,000. The cost of the electronics depends on the ANC logic selected. A cost of \$40,000 to develop the circuitry and about \$15,000 for redundant electronics is typical. In addition, two to three man-months of analysis and some computer time are required to determine ANC parameters and verify the stability of the vehicle, as given in Table 2.

TABLE 2. ACTIVE NUTATION CONTROL COST

	Nonrecurring, \$	Recurring, \$
Accelerometer	—	2,500 to 5,000
Electronics	50,000	10,000
Analysis	20,000	—
ANC total	70,000	15,000

4.4 SPINNING ATTITUDE SENSORS

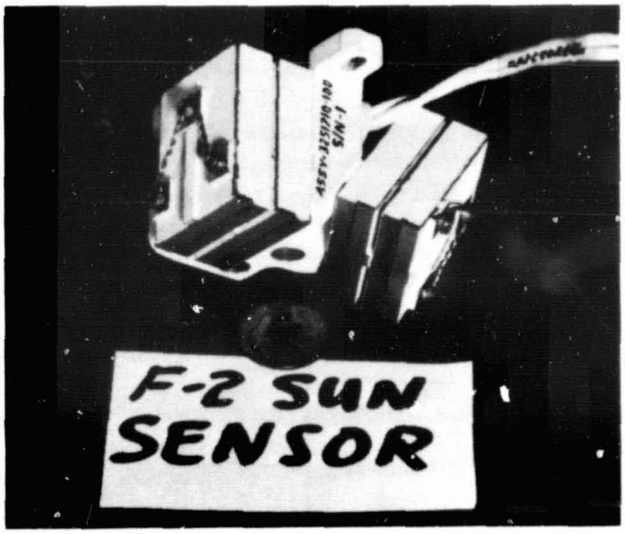
The sensors shown in Figure 7 are used on the Hughes HS 333 class satellites (Anik, WESTAR, and Palapa). Other Hughes satellites use higher precision sensors; however, the more expensive sensors are dictated by on-orbit requirements. The accuracy requirements for transfer orbit operations are not stringent.

The data acquired and the earth-sun-spacecraft geometry are the same for all geosynchronous transfer orbits, independent of the on-orbit mission. The sensors are not used for onboard spacecraft control and interface only with the spacecraft telemetry system and mounting bracketry. Thus, the same sensors can be used for all launches and no development cost is incurred.

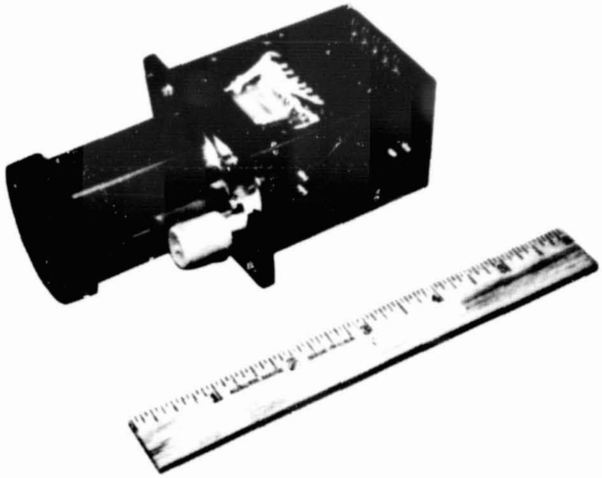
The sensors are available at $7,000 for the internally redundant sun sensor and $44,000 for a pair of earth sensors.

4.5 APOGEE KICK MOTOR

Recurring costs for some pertinent AKM sizes are shown in Table 3. Each of the motors listed in the table is in existence or soon will be. The Delta 2914 AKM was developed for CTS. It can provide a synchronous payload of about 750 pounds (341 kg). The Delta 3914 motor is being developed for the RCA SATCOM. It provides slightly over 1000 pounds (455 kg) on orbit. The cost shown for the Minuteman III motor does not include upgrading, which might be required for this application.



SUN SENSOR



EARTH SENSOR

FIGURE 7. SPINNING ATTITUDE SENSORS (PHOTOS A31102 AND A31878)

TABLE 3. APOGEE KICK MOTORS COST

	Nonrecurring, $	Recurring, $
Delta 2914 class (TE-M-616)	—	~130,000
Delta 3914 class (STAR-30, SVM-7)	—	~150,000
Atlas-Centaur class (TE-M-364-19)	—	~190,000
Quarter-shuttle class	3,500,000	~300,000
Half-shuttle class (Minuteman III)	—	~160,000

There are no nonrecurring costs for the above motors. If a new AKM were developed for a payload not covered by this list, a nonrecurring cost of about $350, 000 could be expected. Also, the recurring cost of the first buys would be higher than the ones in the table.

Spacecraft that do not match the capability of one of these motors can augment the capability of an undersized motor by adding propellant to the spacecraft RCS to provide part of the apogee velocity increment. This hydrazine preburn is used on Intelsat IV, IVA, COMSTAR, and MARISAT.

It is difficult to estimate the structural cost of including an AKM in the spacecraft. Current spacecraft already include the AKM. For new spacecraft, this cost will depend on the spacecraft configuration that is appropriate to the mission. For some configurations, mounting an internal AKM is convenient. On others, it may impact the structural design and require larger spacecraft volume. In some cases, the spacecraft designer may choose to mount the AKM externally to the spacecraft. This will be the case on satellites such as synchronous meteorological satellites where a sensor precludes installation of the AKM within the body.

4.6 EFFECT OF SPINNING LAUNCH ON SPACECRAFT

The spinning nature of the PKM/AKM launch technique affects the spacecraft in a number of ways. Spin provides a benign thermal environment by integrating the solar flux over the spacecraft body thus preventing hot spots, and makes possible the use of simple, lightweight, inexpensive optical sensors for AKM pointing. On the other hand, spin imposes centrifugal loads on the spacecraft and introduces wobble of the spacecraft if the spacecraft is not spin balanced. A spin stabilized spacecraft provides for these effects for on-orbit operation so only the effect on three-axis body stabilized spacecraft must be considered.

Centrifugal loads on the spacecraft due to spin do not impact the structural design, which is driven by the booster acoustic and vibrational loads. Centrifugal loads due to spin represent modest static loads. For example, a satellite that uses the full shuttle diameter has a centrifugal acceleration of only 2 g at its periphery when spinning at 30 rpm. The 10 g centrifugal load experienced on Anik at 100 rpm does not affect the design.

Three-axis satellites tend, because of constraints on the location of equipment, to be more poorly spin balanced than spinners. It would then require considerable weight to achieve the spin balance provided spinners for their on-orbit operation; however, analysis indicates that a considerable degree of spin unbalance is tolerable for transfer orbit operations. The wobble caused by the placement of payload hardware will cause injection errors at perigee and apogee that do not add significantly to the total synchronous injection error. Thus, neither excessive balance weights nor spin balance tests appear to be required for three-axis satellites launched by the spinning PKM/AKM approach.

4.7 TRANSFER ORBIT SOLAR POWER

As shown in Figure 8, the geometric relationship between the solar panels of a spinning spacecraft and the sun direction are similar in transfer orbit and synchronous orbit. Thermal control constraints on the launch windows place transfer orbit eclipse near perigee and place the sun within 25° to 30° from the spacecraft spin plane when the spacecraft is in the apogee firing orientation. Thus, for a spinning spacecraft, no special provisions are required for transfer orbit solar power.

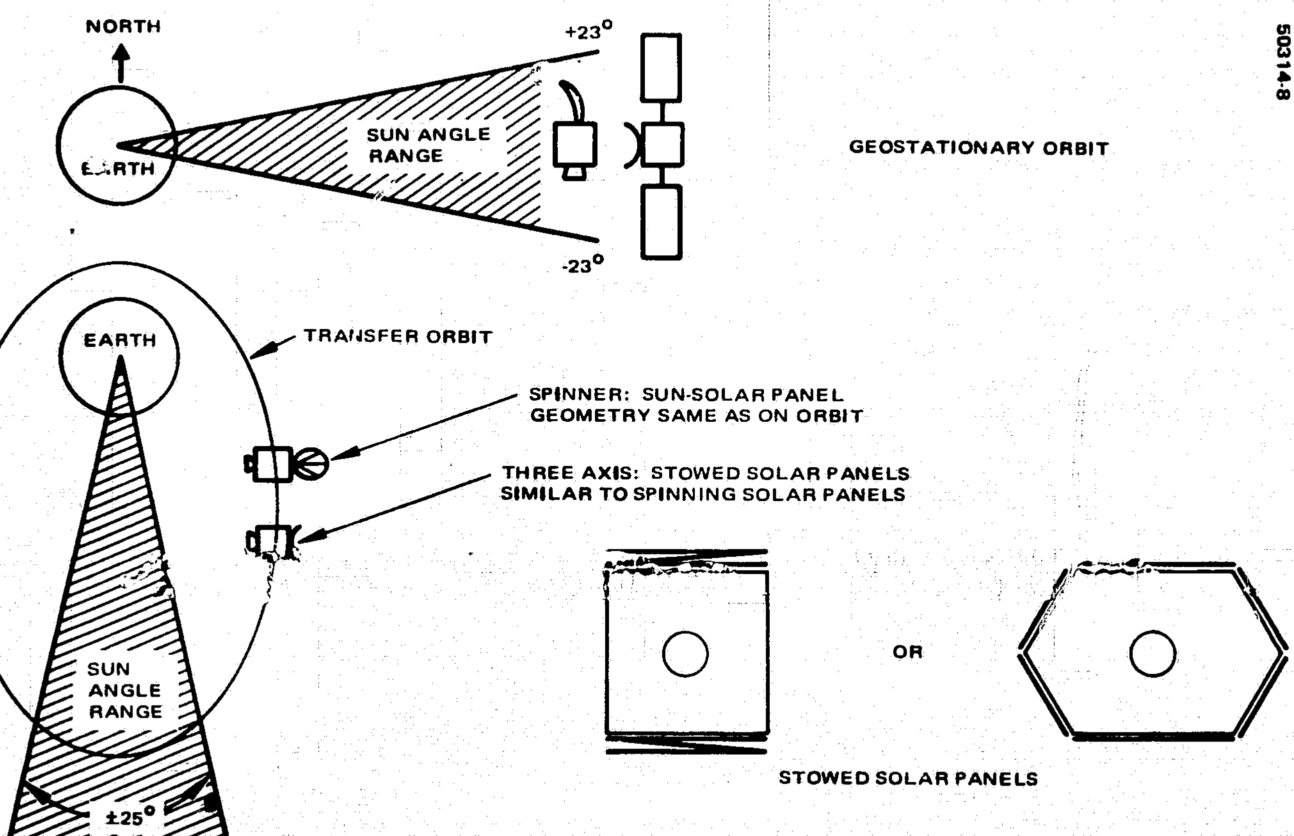


FIGURE 8. TRANSFER ORBIT SOLAR POWER

The solar panels of three-axis satellites must be stowed during apogee motor firing. Transfer orbit panel configurations typical of operational three-axis satellites are shown in Figure 8. In the stowed configuration, the panels are similar to those of the spinning satellite. The panel area exposed to the sun in these stowed configurations is adequate for the low level of power required during transfer orbit.

4.8 SPACECRAFT TELEMETRY AND COMMAND (T&C)

No modification of the spacecraft T&C system is necessary because no T&C link is required between the orbiter and spacecraft. No link is required for spacecraft operation because the spacecraft is an inactive passenger until separation from the perigee stage and postseparation functions can be controlled from the payload mission ground station through the on-orbit T&C system as they are on current expendable booster launches. Nor is a telemetry link to the orbiter required for spacecraft checkout after deployment from the orbit, because retrieval capability is not desirable for the class of spacecraft considered in this report. These spacecraft, which are relatively inexpensive (spacecraft cost is comparable to launch cost) are simple and reliable. None of the more than 20 Hughes communication satellites launched would have been returned to earth from parking orbit even if the capability for checkout and retrieval had been available. Finally, there do not appear to be any safety-critical spacecraft functions during the period between deployment and PKM firing. The AKM is in a fail-safe condition and RCS propellant temperatures will remain near the orbiter payload bay temperature.

4.9 SPACECRAFT SYNCHRONOUS ORBIT CORRECTION

After apogee injection of the spacecraft, it will be necessary to fire the spacecraft RCS to correct the orbital errors resulting from both perigee and apogee injection. Fuel is also required to move the spacecraft longitude from the longitude at apogee injection to the station longitude. The relation between RCS fuel needed to correct orbital errors and the various error sources will be discussed in Section 7, Guidance Considerations.

5. PERIGEE STAGE

5.1 PERIGEE STAGE SUPPORT

The perigee stage (Figure 9) includes several elements in addition to the solid rocket motor. The stage will require, on most launches, an active nutation control system for spin stability and a means of commanding the stage events (ANC on, PKM arm, PKM fire, and spacecraft separation). No stage telemetry is required. The baseline concept is to make the perigee stage self-supporting so that no interface other than the basic structural interface is required between the stage and spacecraft. This choice provides an upper bound to the cost associated with the launch. If the spacecraft design is such that it can provide any of these functions in a more cost effective manner, the user could elect the dispense with the perigee stage hardware involved in this function. For example, if a spacecraft required active nutation control for transfer orbit operation, the same ANC with minor modifications could stabilize the payload before perigee firing eliminating the costly perigee stage reaction control system.

No spinup capability is required on the perigee stage because the baseline deployment concept provides for spinup before separation.

5.1.1 Perigee Stage ANC

The payload composed of the perigee stage and the spacecraft (including AKM) will generally not be spin stable. Consequently, after separation from the orbiter, the nutation angle of the spinning payload will grow exponentially with time until the PKM is fired. The nutation angle at the time of PKM firing will depend on the initial nutation angle resulting from tipoff transients at separation and on T_D, the dedamping time constant of the vehicle. T_D for the payload vehicle will vary according to the mass properties of the spacecraft, configuration of the spacecraft propellant tanks, and spin speed. At 30 rpm, values of T_D range from 4 to 20 minutes for vehicles with solid motors. This range is also the potential range for the period between separation and PKM firing. For a vehicle that is spinning freely for a single dedamping time constant, the PKM pointing error will increase by over 2^o. Thus, at least some users, probably including all vehicles with liquid apogee motors, will require active nutation control of the perigee stage.

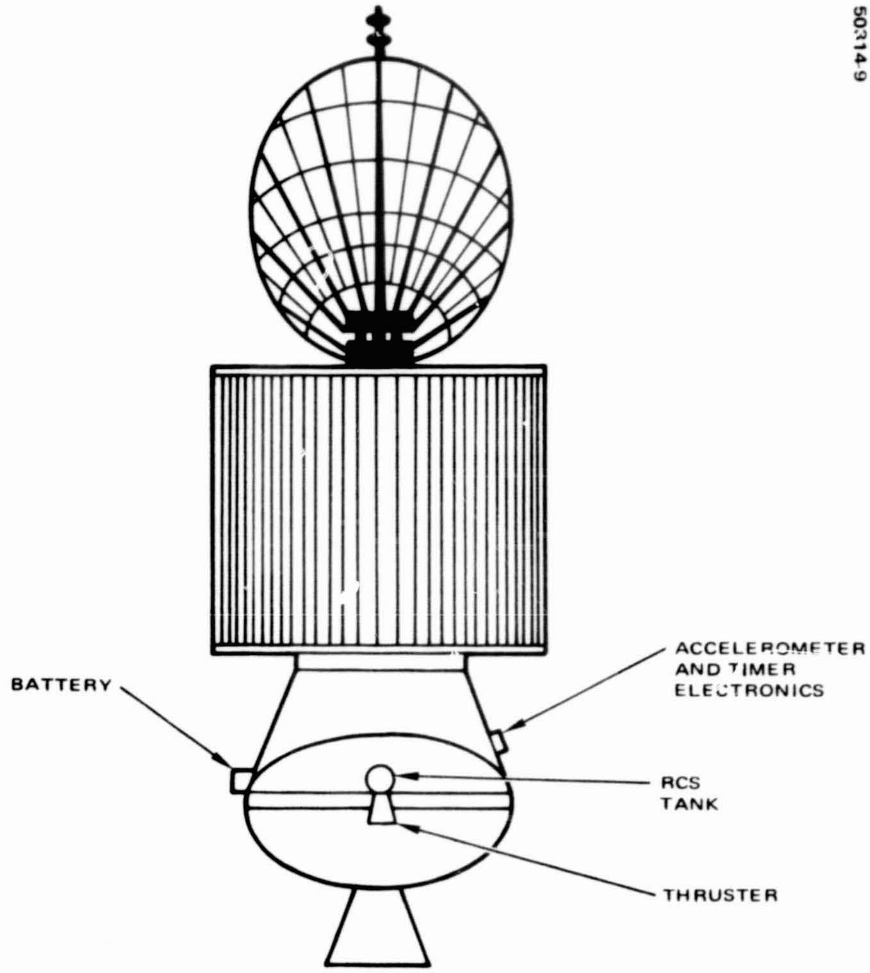


FIGURE 9. PERIGEE STAGE

The perigee stage active nutation control characteristics and costs (Table 4) are similar to those described for spacecraft transfer orbit ANC; however, unlike the spacecraft that uses the on-orbit RCS, the perigee stage will require the addition of an RCS for the ANC function. Cold gas was selected over hydrazine for lower cost and also to avoid orbiter jet impingement problems. Some recurring analyses are also included for the perigee stage because the stage can be expected to carry a variety of payloads.

The cost of the ANC can be avoided for payloads that have relatively low energy dissipation because nutation will grow slowly for these payloads. The cost of ANC for payloads that require this function can be reduced in most cases by implementing the ANC in the spacecraft. If the spacecraft has ANC for transfer orbit, this ANC, with minor modifications, can provide the perigee stage stabilization. If the spacecraft does not have ANC, a considerable savings can still be made by using the spacecraft RCS rather than adding a cold gas RCS to the perigee stage.

TABLE 4. PERIGEE STAGE ANC COST

	Nonrecurring, $	Recurring, $
Analysis	25,000	10,000
Accelerometer	—	3,000
Electronics	250,000	30,000
Battery	—	1,000
RCS	300,000	150,000
	575,000	195,000

5.1.2 Perigee Stage Support — Telemetry and Command

The IUS specification requires that the payload have telemetry and command capability for safety-critical functions over a range of 20 n. mi. from the orbiter. The conclusion of this study is that there are no safety-critical telemetry or command functions for the perigee stage or spacecraft. The only potential postseparation threats to the orbiter from these payloads appear to be related to propulsion and possible recontact between the orbiter and payload. The propulsion threat is primarily from possible premature ignition of the PKM or AKM. Premature ignition does not appear to be detectable from telemetry nor preventable by radio command. Premature ignition is best prevented by a fail-safe design of the safe/arm system and of the motor firing control.

Fail-safe arm and firing commands can be provided either with an RF link or a timer. A timer has been selected because of its lower cost and because it relieves the orbiter crew of a set of mechanical, time based functions which require no human judgment. Ignition of the PKM before adequate payload orbiter separation is achieved is precluded by the following.

1) Redundant commands from independent timers required for arm and fire

2) Activation of both a separation switch and a g switch, which senses separation impulse is required

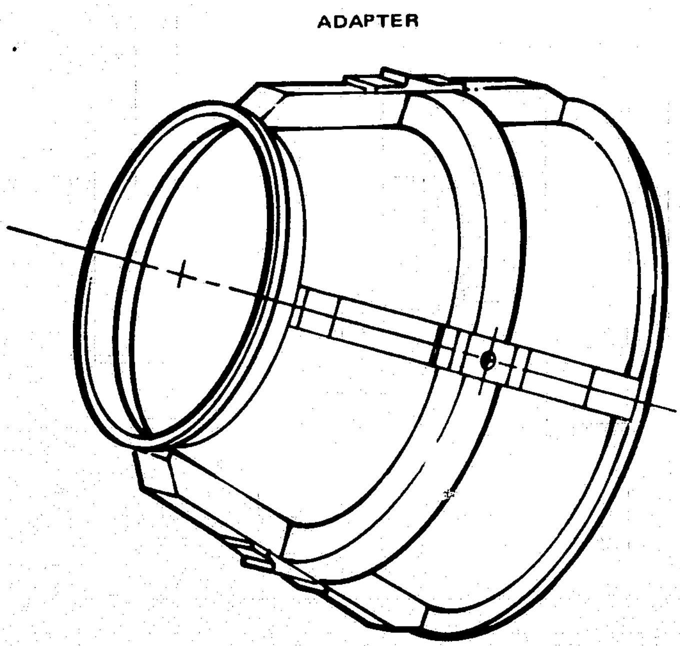
3) Required payload separation (3000 feet) (915 m) provided by separation springs without need of shuttle maneuver

The timer, which could be similar to the sequences used on Intelsat IVA and COMSTAR to control the spinup sequence after separation from the Atlas-Centaur, would require about $100,000 for development and $25,000 for a pair of redundant units. An S band radio link, if it were required, would require $150,000 to develop and $50,000 per unit. Redundant units would probably be required.

5.2 PERIGEE STAGE – SPACECRAFT INTERFACE

The spacecraft to perigee stage adapter (Figure 10) is similar to the adapter between the Delta third stage and spacecraft. The conical aluminum adapter is fabricated with three or four circular rings, depending on payload size, four longerons, and conical skin. The adapter provides the mechanical interface with the shuttle orbiter cradle and supports both PKM and spacecraft.

The separation between the PKM and spacecraft could be the same as on Delta launches. A Marmon clamp, which holds the spacecraft to the adapter, is released by the perigee stage timer about 1 minute after nominal motor burnout. Four axial springs separate the spacecraft from the PKM. A perigee stage weight is released to destabilize the spinning PKM so that it does not recontact the spacecraft in the event of PKM "chugging" (continued intermittent thrust).



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SPACECRAFT ON-ORBIT WEIGHT (lb)	(kg)	ADAPTER WEIGHT (lb)	(kg)	COST NON-RECUR. ($K)	COST RECUR ($K)
1000	454	85	39	250	150
2000	908	220	100	300	200
6000	2724	590	269	400	300

SEPARATION

- SEPARATION IDENTICAL TO DELTA
- 40 sec AFTER NOMINAL BURNOUT

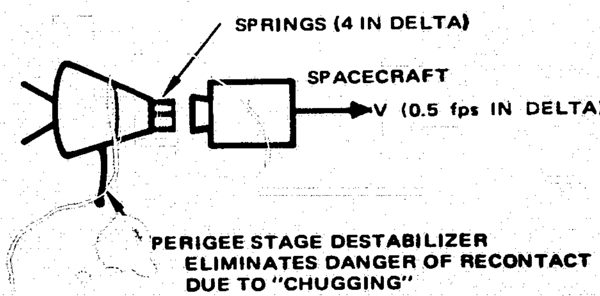


FIGURE 10. PERIGEE STAGE – SPACECRAFT INTERFACE

5.3 PROPULSION

5.3.1 Propulsion Technology

It can be seen in Table 5 that the current technology provides nearly the maximum performance obtainable within the constraints of shuttle safety and minimum development risk. Although the risk associated with the short term low risk category is small, performance is not critical. Thus, the analysis in this report is based on the use of current technology.

5.3.2 Perigee Stage Requirements

The requirements on the perigee kick motor are straightforward. The PKM must provide a velocity increment of 8000 fps. The payloads to which the PKM must impact this velocity increment are discussed below.

5.3.3 Perigee Stage Payload Sizes

Motor sizes for a variety of payload weights are given in Table 6. The Delta 2914 launched payloads are nominally 750 pounds (341 kg) and the STS with orbiter and PKM/AKM can launch these payloads with the Delta third stage motor, the TE-364-4, and the PKM. The STS could launch larger payloads (e.g., up to 820 pounds (373 kg), with the TE-364-4 by raising the orbiter to a higher altitude (i.e., 400 n. mi. (740 km), at the expense of 20,000 pounds (9,091 kg) of additional orbiter fuel. If half of the orbiter fuel is attributed to each payload, then the shuttle payload per synchronous payload would be increased by 10,000 pounds (4,545 kg) for a total of 15,000 pounds (6,818 kg). The 750 to 820 pound (341 to 373 kg) payload cases, therefore, do not require a PKM development; the TE-364-4 is adequate.

TABLE 5. RECOMMENDED TECHNOLOGY FOR PKM

Status	Technology	Potential Weight Reduction, % PKM	Potential Weight Reduction, % Shuttle Payload
Options			
Present	Titanium or fiberglass case		
	Carbon phenolic nozzle		
	Class II propellants		
Short term low risk	Kevlar case 1	~2%	1
	Carbon-carbon nozzle		
	Class II propellants		
Long term	High strength kevlar case	~10	6
	High carbon-carbon nozzle		
	Class VII propellants		
Conclusion			
Relatively small weight savings not critical item			
Maintain use of present technology			

TABLE 6. PERIGEE STAGE PAYLOAD SIZES

ΔV = 8000 fps			
Synchronous Payload (BOL), lb (kg)	PKM Weight, lb (kg)	Shuttle Payload, lb (kg)	Basis
750 – 820 (341–373)	2500 (1135)	5000 – 15,000 (2270–6810)	TE-364-4
750 (341)			2914 Delta
1000 (454)	3400 (1544)	5700 (2588)	Delta 3914 class
2100 (953)			Centaur
2160 (981)	8050 (3654)	13,200 (5993)	Minuteman III
2850 (1294)	9550 (4336)	16,250 (7378)	1/4 orbiter
3200/1600 (1452/726)			Titan IIIC
6150 (2792)	18,800 (8535)	32,500 (14,755)	1/2 orbiter

The Delta 3914 capability matches a nominal 1000 pounds (455 kg) spacecraft in synchronous orbit and this exceeds the TE-364-4 capability. A new PKM with a weight of 3400 pounds (1545 kg) must be developed to match this capacity.

The Atlas-Centaur has a 2100 pound (955 kg) spacecraft launching capability and the existing and space qualified Minuteman III solid rocket motor can provide this capability. A new motor development in this class (namely Centaur) is not required because the Minuteman III 8050 pound (3659 kg) motor can be used.

An analysis was made of the maximum payload weight the orbiter could accommodate assuming the orbiter bay volume was divided into two and four segments. In the quarter-shuttle bay case, the satellite weight could grow to 2850 pounds (1294 kg) and the PKM would weigh 9550 pounds (4336 kg). On the half-shuttle bay case, the spacecraft could be 6150 pounds (2792 kg) and require a PKM weighing 18,800 pounds (8,535 kg). Although it is difficult to envisage what mission these large spacecraft might have, it does denote the large weight the PKM/AKM concept can accommodate.

5.3.4 Propulsion Configurations

The focus of the study was on the PKM and related functions because the AKM function has been broadly practiced. Two aspects of the AKM system design are significant in the PKM/AKM concept and involve the type of AKM, liquid versus solid, and its attachment to the spacecraft, integral and nonintegral (Figure 11).

Most current spacecraft, because of the expendable launch vehicle shroud limitations, use an integral solid AKM configuration. The Europeans with the Symphonie, launched in 1974, pioneered the first integral liquid AKM.

The SMS launched in 1974 is an example of the nonintegral solid AKM in that the AKM was jettisoned after firing in order to expose a sensor radiation cooler to cold space.

The only design shown that has not flown is a solid PKM and liquid AKM, which are nonintegral. This design, however, has the feature that the combined PKM/AKM stage could be very compact in length and spin stable in the PKM/AKM staging. This type stage design would be an appropriate consideration for development and use with payloads currently launched by Titan IIIC Transtage.

The nonintegral solid or liquid AKM could also be considered for low orbit spacecraft where their orbit is higher than the orbiter directly achievable altitude.

The most important consideration is the flexibility an STS user has with the PKM/AKM concept. The STS offers the payload supplier and purchaser a variety of options the current expendable launch vehicles do not have and which would be limited by a government-furnished upper stage.

5.3.5 Typical Perigee Kick Motors

Figure 12 shows motor configurations and performance parameters for solid rocket motors compatible with synchronous payloads of 1,000, 2,000, and 5,500 pounds (454, 909, and 2,497 kg). These data are representative of data obtained from several SRM manufacturers in order to assess the range of PKM performance and characteristics available for potential shuttle payloads.

The parameters listed in the figure and the configuration will vary according to manufacturer and design criteria. For example, if a shorter motor is desired to achieve better shuttle packing it can be achieved by nozzle submergence at the cost of increased inert weight and lower maximum operating pressure (MEOP). The lower MEOP results in lower I_{sp}. Also, if a

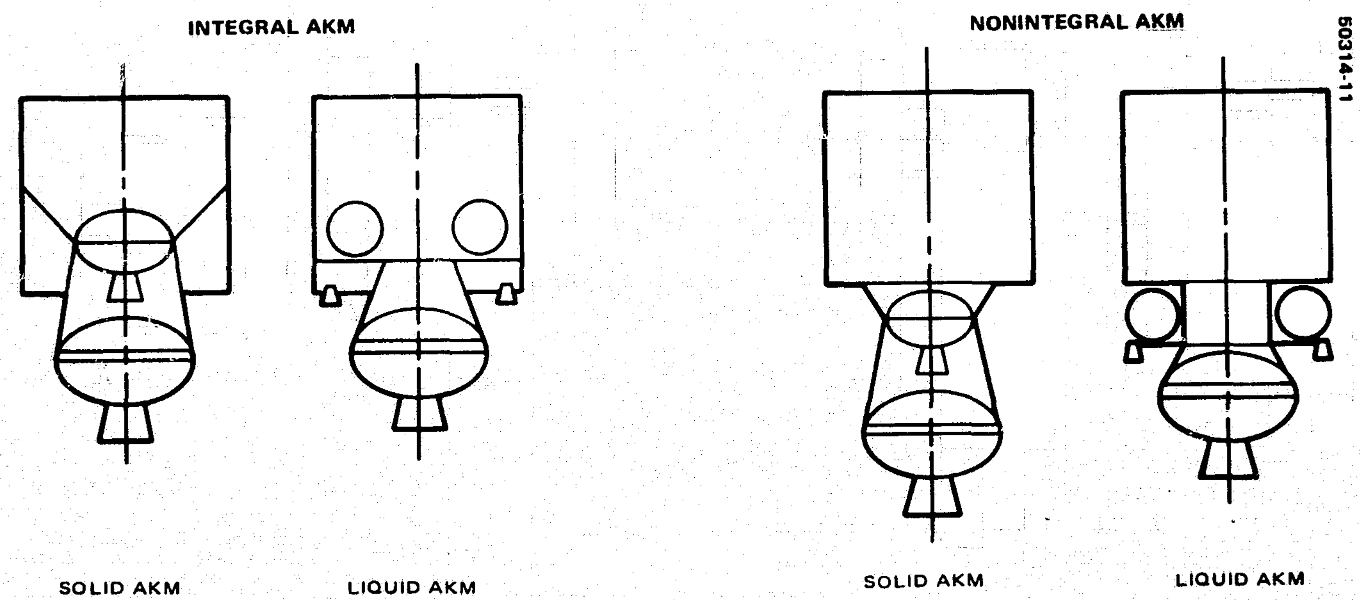
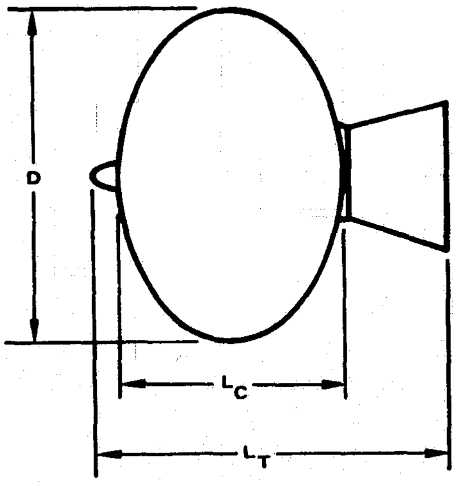


FIGURE 11. PROPULSION CONFIGURATION



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1000 lb SYNCHRONOUS PAYLOAD	NOMINAL NOZZLE SUBMERGENCE		MAXIMUM NOZZLE SUBMERGENCE	
OVERALL LENGTH, L_T (in.)	66.0	(1.67 cm)	43.5	(110 m)
CASE LENGTH, L_C (in.)	38	(0.96 m)	37	(0.94 m)
OVERALL DIAMETER, D (in.)	49.7	(1.26 m)	52.5	(1.33 m)
TOTAL WEIGHT (lb)	3,160	(1,435 kg)	3,450	(2,566 kg)
BURN TIME (sec)	60		120	
MAXIMUM THRUST (lbf)	19,800	(88,100 N)	11, 000	(48,950 N)
MAXIMUM ACCELERATION (g)	6.6		2.9	
2000 lb SYNCHRONOUS PAYLOAD				
OVERALL LENGTH, L_T (in.)	81.7	(2.08 m)	52.7	(1.34 m)
CASE LENGTH, L_C (in.)	46	(1.17 m)	47	(1.19 m)
OVERALL DIAMETER, D (in.)	63.5	(1.61 m)	62.0	(1.57 m)
TOTAL WEIGHT (lb)	6,210	(2,819 kg)	6,840	(3,105 kg)
BURN TIME (sec)	75		120	
MAXIMUM THRUST (lbf)	31,300	(139,000 N)	21,000	(93,500 N)
MAXIMUM ACCELERATION (g)	5.2		2.8	
5500 lb SYNCHRONOUS PAYLOAD				
OVERALL LENGTH, L_T (in.)	110	(2.79 m)	84	(2.13 m)
CASE LENGTH, L_C (in.)	62	(1.57 m)	64	(1.63 m)
OVERALL DIAMETER, D (in.)	86	(2.18 m)	90	(2.29 m)
TOTAL WEIGHT (lb)	15,400	(6,992 kg)	16,100	(7,309 kg)
BURN TIME (sec)	100		120	
MAXIMUM THRUST (lbf)	57,200	(254,000 N)	44,500	(198,000 N)
MAXIMUM ACCELERATION (g)	3.9		2.4	

FIGURE 12. SOLID ROCKET MOTOR COMPATIBILITY WITH SYNCHRONOUS PAYLOADS

lower maximum thrust is required it can be achieved by reducing MEOP and hence I_{sp}. Case length versus diameter is also subject to tradeoff. The range of length/diameter ratios shown are nearly optimum from the motor weight standpoint.

5.3.6 PKM Cost Summary

Table 7 summarizes the costs for three previously described motor sizes. The sizes shown are for Delta-class, quarter-shuttle, and half-shuttle payloads. In addition to the basic development and qualification costs in the table there will be special tooling costs associated with the manufacture of new motors. These tooling costs should be less than $1.5 million. The development costs shown are based on the assumption that the several sizes would not be developed in parallel programs. Parallel programs would result in significantly lower costs. The unit cost includes a fee (based on a five motor buy) for one motor being fired as an acceptance round.

The Minuteman III unit cost shown is for the motor as qualified for its ballistic missile application. Although it is qualified for space firing, some upgrading of quality control and safety margins may be required.

5.3.7 PKM Development Plan

The development program shown in Figure 13 can be met by all motor manufacturers. The development program could be reduced to 24 months by a motor manufacturer currently fabricating solid rocket motors in the size class being purchased.

TABLE 7. PKM COST SUMMARY (1975 DOLLARS)

Synchronous Orbit Payload, lb	Development and Qualification Cost, $*	Unit Cost, $
1000	3,500,000	380,000
2800	4,000,000	460,000
5500	5,600,000	545,000
TE-364-4	—	190,000
Minuteman III	—	160,000

*Assumes three development and five qualification tests.

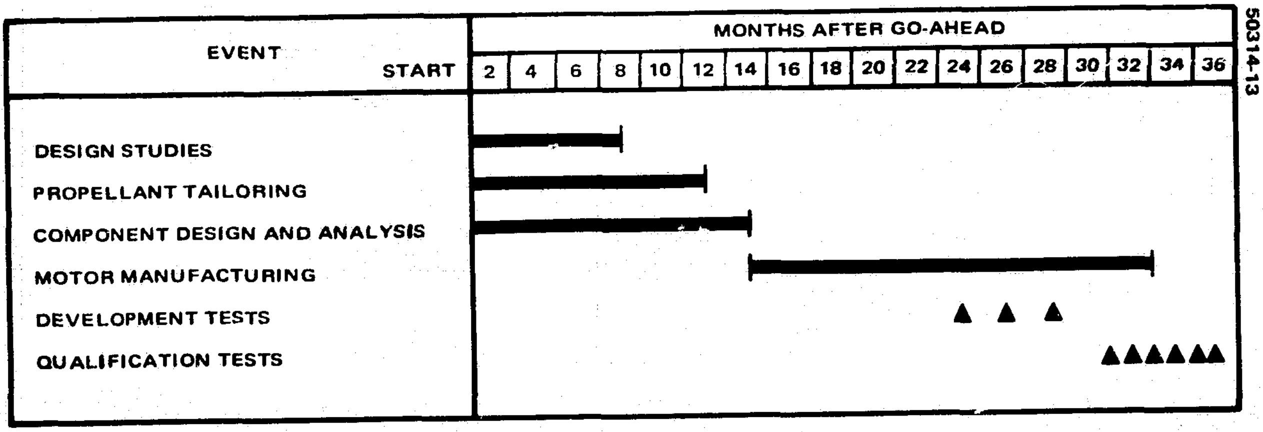

FIGURE 13. TYPICAL DEVELOPMENT PROGRAM

6. PAYLOAD INSTALLATION AND DEPLOYMENT

6.1 BASELINE DEPLOYMENT CONCEPT

The Delta launch sequence has a nominal 40 seconds between the third stage separation and solid motor firing. Firing the PKM only 40 seconds after release from the orbiter is unacceptable because safe separation cannot be reasonably obtained in that time.

A study goal was to achieve safe separation in a reasonable time and without requiring an orbiter maneuver. The assumed safe separation distance was 3000 feet (915 m) based on the USAF IUS (interim upper stage) specification.

The separation velocity (V_{sep}) is constrained by a reasonable separation system design, the payload mass, and reaction forces acting on the orbiter pitch control system. The baseline separation velocities selected are 4 fps (1.2 m/sec) for the Delta-class payloads and 2 fps (0.6 m/sec) for Centaur or larger class payloads.

The deployment concept (Figure 14) is as follows:

1) The orbiter is oriented to the desired attitude depending on the separation velocity the payload requires and tilt table rotation angle.

2) The orbiter payload doors are opened, the payload is rotated on the tilt table 45^{o} to 60^{o} (the exact angle was not determined in the study), and the payload is spun up to the desired spin speed (i.e., the large payloads 30 rpm and the small payloads 30 to 100 rpm). A V_{sep} = 2 fps (0.6 m/sec) will require payload separation 20 minutes before PKM firing; a V_{sep} = 4 fps (1.2 m/sec) will require payload separation 13 minutes before firing.

3) The separation time will be determined so that the payload will be crossing the equator (the desired perigee) at the time of PKM firing.

4) After a safe separation of 3000 feet (915 m), the PKM motor is fired at the point of equatorial crossing in the payload orbit.

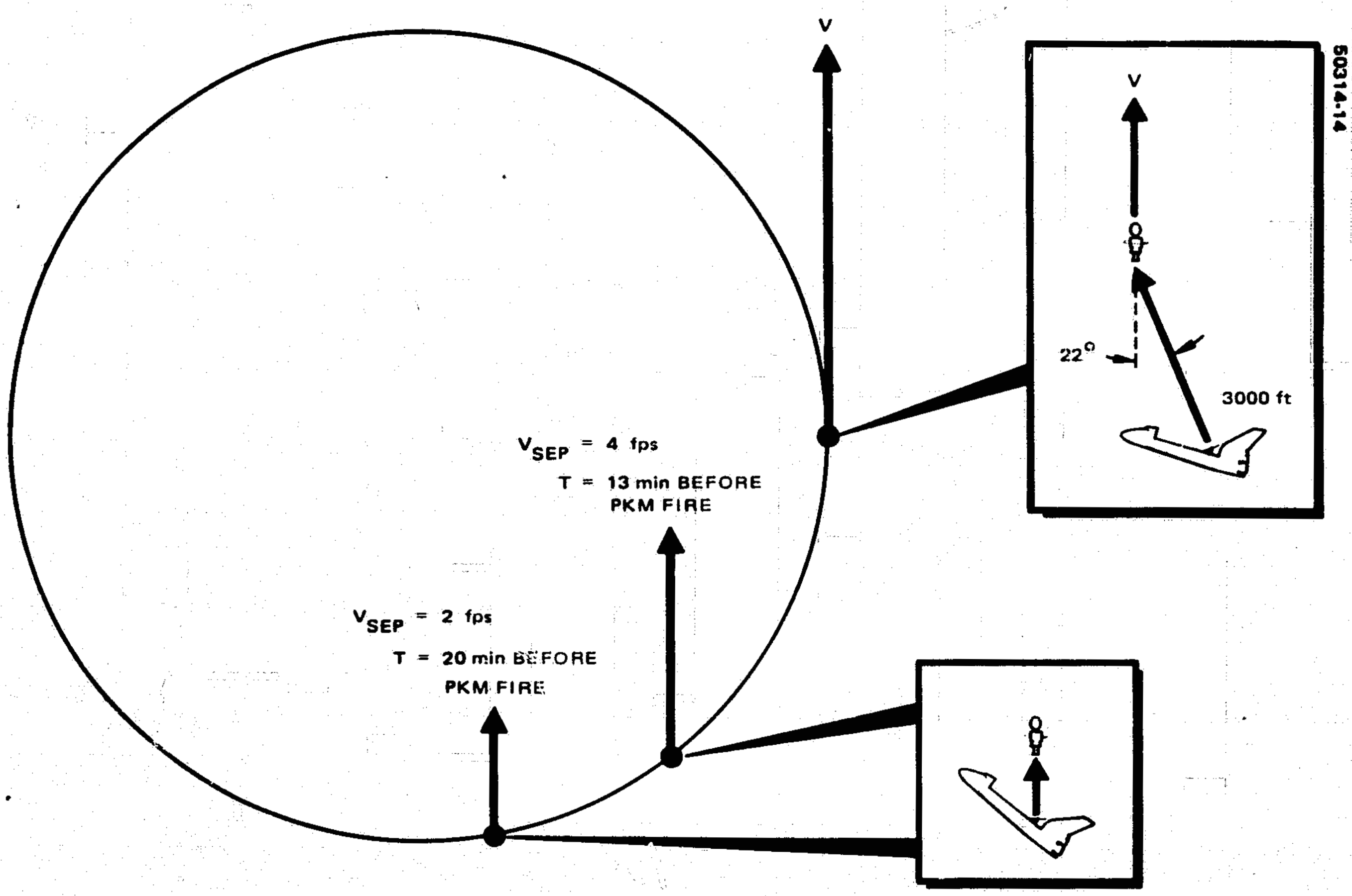

FIGURE 14. BASELINE DEPLOYMENT CONCEPT

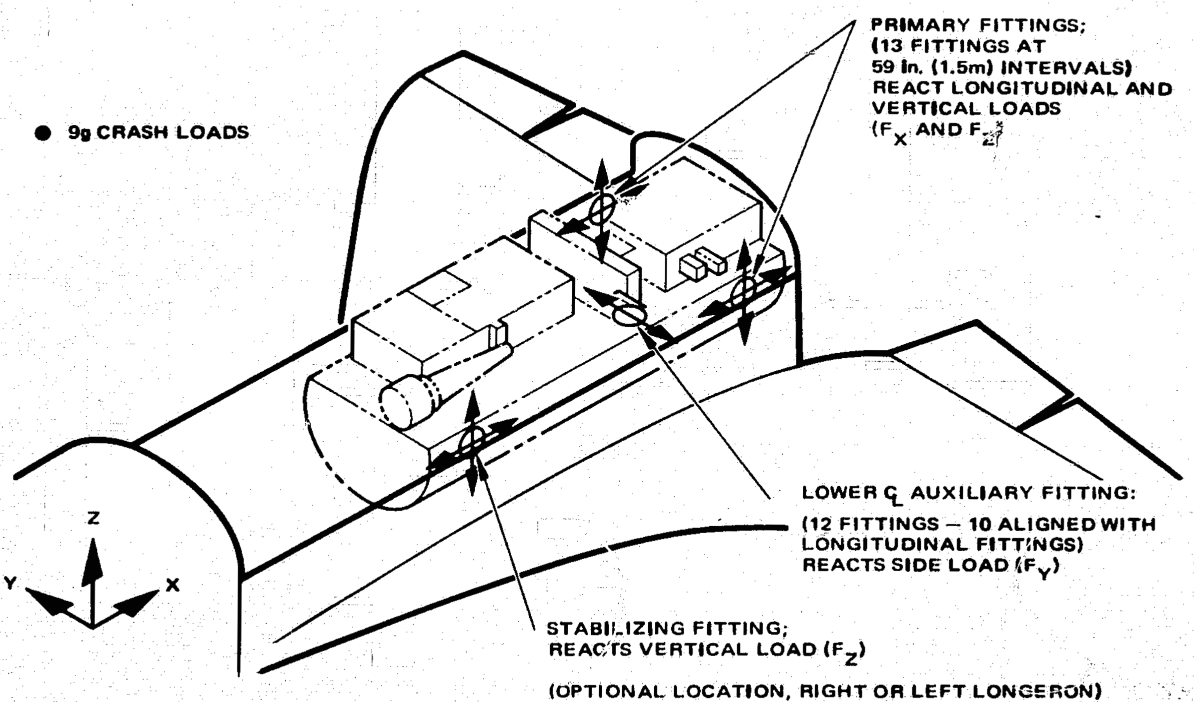

FIGURE 15. PAYLOAD RETENTION SYSTEM

For the two separation velocities, the orbiter will be approximately 20° from the plume center of the solid rocket motor, assuming the orbiter has not made any maneuvers.

The 3000 foot (915 m) safe separation distance and the orbiter being nominally 20° from the solid rocket motor plume were not analyzed in this study. The 3000 feet (915 m) separation was judged to be a safe distance in the event of motor explosion, but a more definitive analysis is required. If further analysis indicates a great distance is required, both orbiter maneuvers or more time before PKM firing are possible, and the baseline concept would remain valid.

6.2 PAYLOAD RETENTION SYSTEM

A four-point retention concept, as shown in Figure 15, provides a statically determinate mounting. The attachment fittings along the longeron react loads in either the ±X and ±Z directions (primary) or the ±Z directions (stabilizing), while the lower keel fittings react loads in the ±Y direction (auxiliary) only. Keel fittings at orbiter Xc stations 715, 951, 1069, and 1181 (18.2, 24.2, 27.2, and 30 m) will react ±X loads in addition to ±Y loads. The stabilizing fittings may be located on either the left or right longeron. The orbiter supplied interface fittings will minimize Y loads in the primary fittings, X and Y loads in the stabilizing fittings, and X and Z loads in the keel fittings. Statically indeterminate payload attachment methods are not precluded, but such methods must be compatible with the structural and mechanical capability of the orbiter attach points for all combinations of deflections and loads.

Primary longeron fittings occur every 59 inches (1.5 m) on both left and right longerons. Intermediate fittings can be provided as vernier fill bridges.

The installation must be such that structural integrity is maintained in the event of a 9 g crash load.

6.3 CRADLE CONCEPT

The initial consideration for an STS launch sequence patterned after Delta is installation of the payload in the STS orbiter payload bay. Several concepts were evaluated and a baseline design was selected. The large hypothetical payload shown is a valid indication the Delta launch sequence pattern for the STS is not limited to Delta-sized payloads. The baseline cradle concept is shown with the payload stowed and the orbiter bay doors closed, i.e., the STS launch configuration (Figure 16).

The baseline is a single-cradle concept with two attachments on each orbiter bay longeron and a single attachment to the orbiter keel. The advantage of a single cradle is that the payload attachment has a statically determinant load path (three-point attachment) that prevents loads being induced

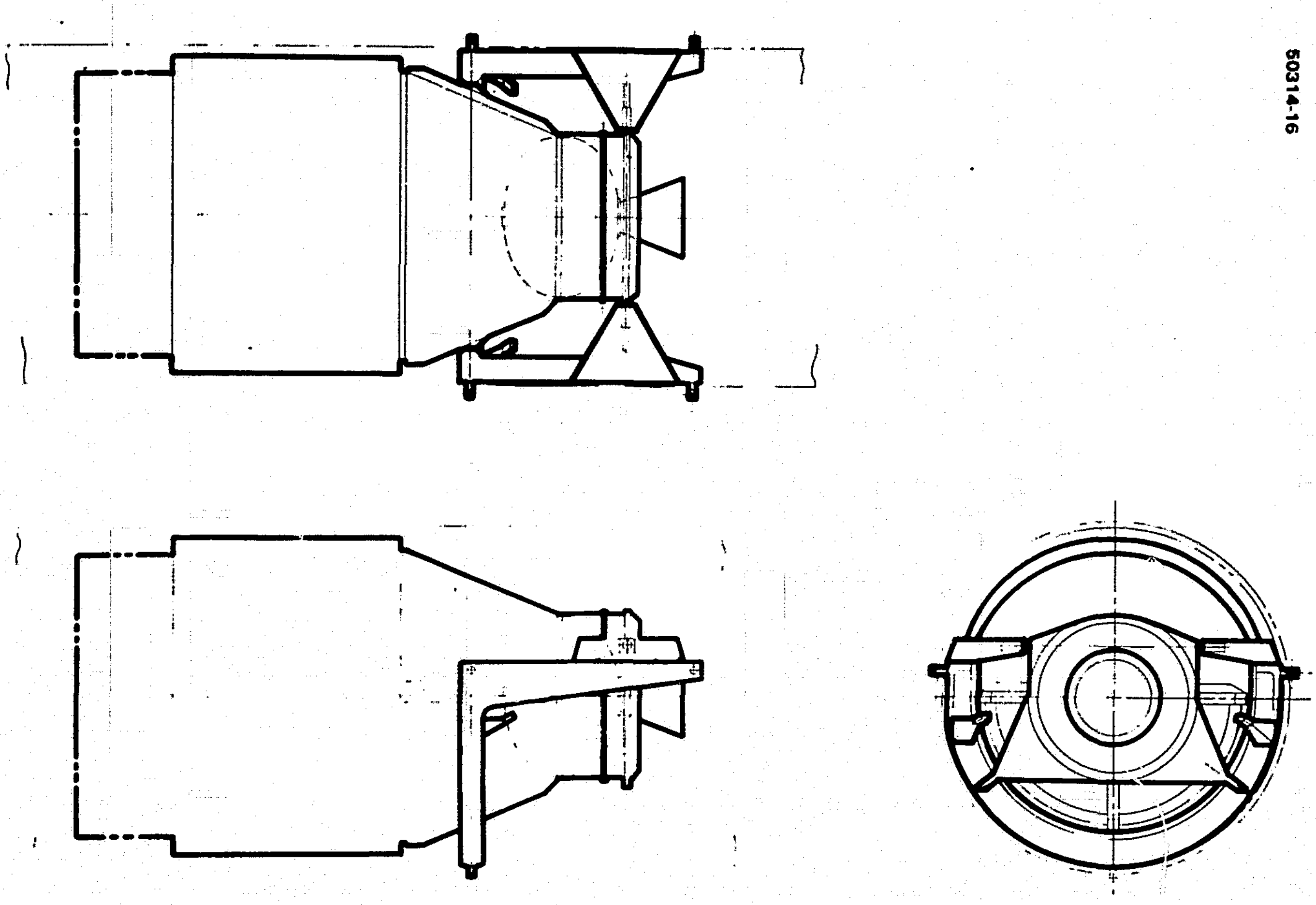

FIGURE 16. CRADLE CONCEPT

in the payload by orbiter distortions. Furthermore, since structural coupling occurs only between the cradle and the orbiter, an orbiter to payload coupled analysis will not be required for different payloads. Such an analysis would be required for each new payload with a dual-cradle concept.

The loads into the cradle are minimized by making the attachment to the payload adapter close to the payload center of gravity. Since the PKM and its payload are nearly equal in weight, the center of gravity will generally be slightly forward of the PKM. The launch loads are transferred to the cradle, thus precluding significant loads into the tilt table mechanism described later. In the unlikely event of an on-orbit emergency, the cradle attachments to the orbiter could be designed for emergency release and the entire cradle and payload could be discarded by the remote manipulator system (RMS).

The aluminum cradle will be designed to orbiter requirements as specified in Volume XIV, JSC 07700.

6.4 TILT TABLE DEPLOYED

After the orbiter altitude is achieved and the payload bay doors are open, the payload will be released from the cradle latches. Because of reliability and ability to relatch considerations, the baseline design incorporates electrical latches (defined as orbiter standard latches, page 7-4, Vol. XIV, JSC 07700).

The large hypothetical payload is shown in Figure 17 with the tilt table deployed and locked, and the spacecraft ready for spinup with a spin mechanism mounted on the tilt table. The tilt table is driven by redundant electric motors and the tilt table rotation takes several minutes to avoid disturbances to the orbiter control system. The tilt table locks into position for precise orientation and stability during payload release.

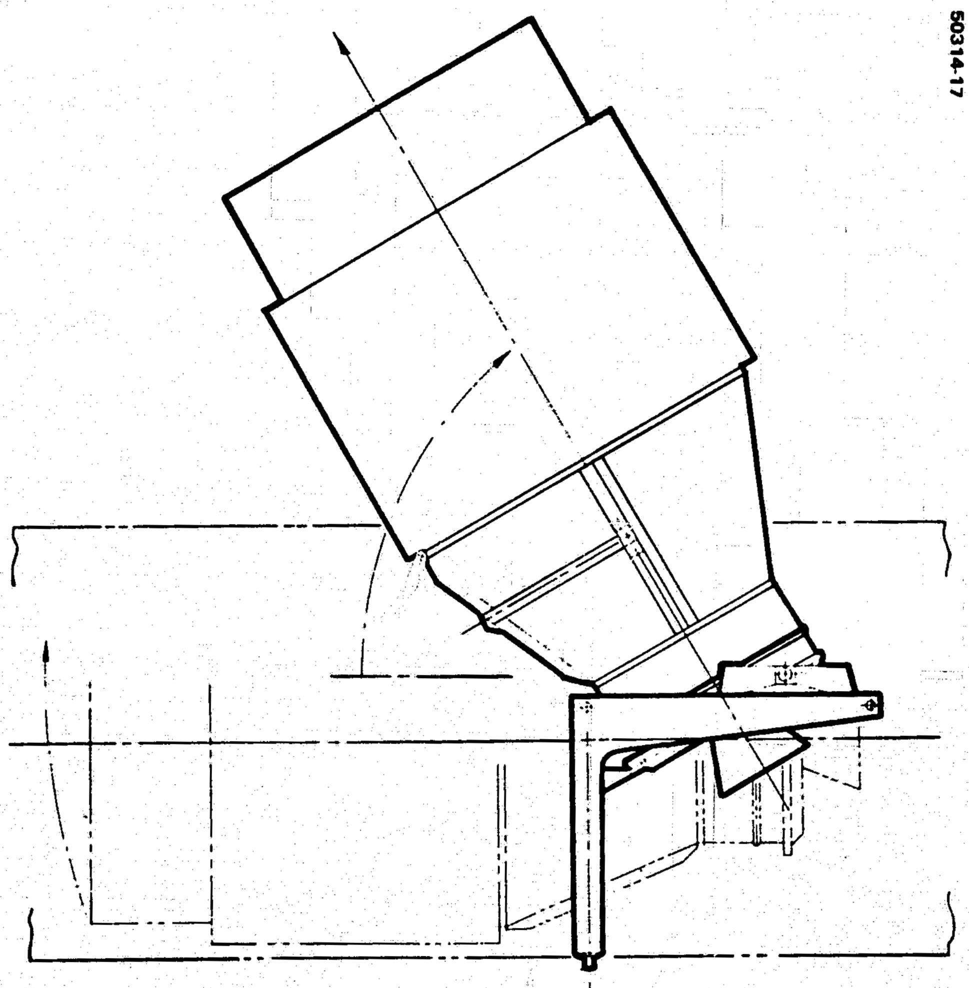

FIGURE 17. TILT TABLE DEPLOYED

The desired payload spin speed will be nominally 30 rpm due to accuracy and stability considerations. The payload is separated by releasing a Marmon clamp. A device is required to retain the Marmon clamp. When the clamp is removed the separation springs are able to accelerate the payload axially away from the orbiter. For a tilt angle of 45° to 60° and the characteristics of the selected springs as shown in Table 8, the separation velocities selected in preliminary analysis indicate no problem of clearance with the shuttle. The velocities imparted by these springs and the pitch reaction induced in the shuttle are given in Table 9.

Normal spring operation will produce a nutation of the spinning payload of no more than 0.6°. Total failure of a spring would produce a nutation angle of less than 10° and no lateral velocity. This event would not lead to recontact for a single payload but could be a problem for the dual Delta-class launch. Lateral spring forces are very small. This type of spring configuration has been used in many launches without abnormality. Guides to ensure axial separation are unnecessary and represent a more serious failure mode than the one they would attempt to avoid.

TABLE 8. SPRING CHARACTERISTICS

Free length	12 in. (30.5 cm)
Compressed length	5.5 in. (14 cm)
Diameter	4 in. (10.2 cm)
Thickness	0.375 in. (1 cm)
Steel	17-7 PH
Coils	9

TABLE 9. SEPARATION PARAMETERS – 16 SPRINGS

Synchronous Payload Weight, lb (kg)	Shuttle Payload Weight, lb (kg)	Separation Velocity, fps (m/sec)	Shuttle Pitch Rate, deg/sec
1,000 (455)	5,700 (2,590)	4.5 (1.4)	0.2
2,000 (909)	13,000 (5,909)	3.3 (1.0)	0.3
6,000 (2,727)	30,000 (13,636)	2 (0.6)	1.2

6.5 DELTA-CLASS CRADLE CONCEPT

A Delta-sized payload cradle concept was also considered and a baseline design selected. The design features as shown in Figure 18 are the same as the cradle for the large hypothetical spacecraft. The significant difference is the ability to support two spacecraft.

A cradle design refinement, which was not attempted in the limited time of this study, would be a common cradle for Delta-class, Centaur-class, and full orbiter diameter class payloads. This baseline single-cradle concept has the virtue of making a common design for different payload sizes a reasonable consideration.

The over and under arrangement for the Delta-class payloads was selected because the orbiter center of gravity landing requirements are satisfied even if only one spacecraft is launched. A side-by-side arrangement violates the lateral center of gravity requirements for landing if one spacecraft is launched and the other is retained.

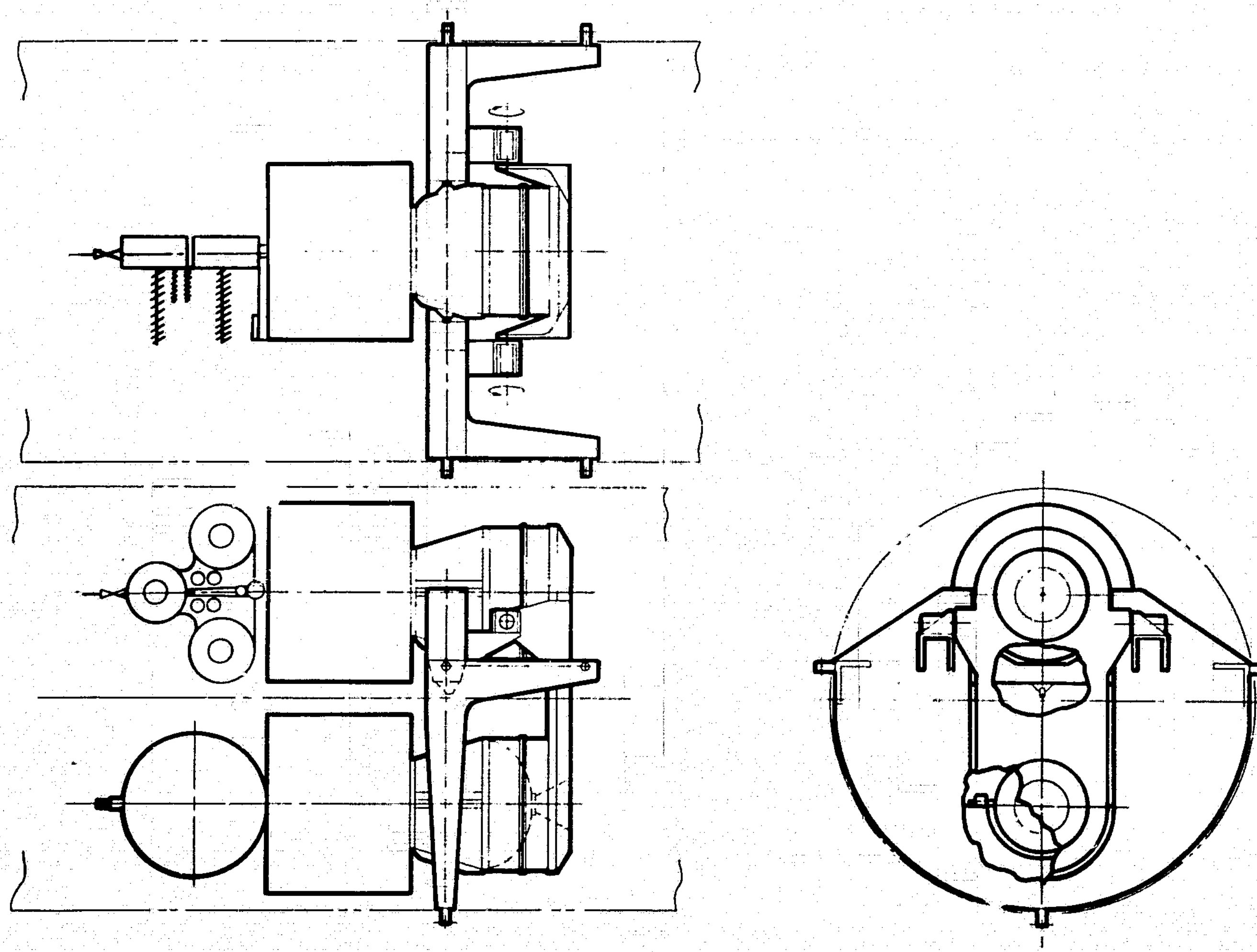

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FIGURE 18. DELTA-CLASS CRADLE CONCEPT

The cradle mounting of the payload and the holddown technique are the same as for the single payload. The lower payload as well as the upper payload is held down by cradle-mounted electric latches which fasten over pins that rest on the cradle. A bridge is provided across the cradle to accept the lower pin of the upper payload. The stiffness of the strongback must be adequate to prevent bending under the unbalanced separation impulse. The weight of the cradle and tilt table is estimated at 700 pounds (318 kg).

A significant feature of this design concept is the accommodation of existing Delta-launched spacecraft without modification. WESTAR and MARISAT spacecraft are examples.

6.6 DELTA-CLASS TILT TABLE

The Delta-class payloads are extended on a common tilt table (Figure 19), but each payload is individually spun up and separated. The spin speed could vary from a nominal 30 to 100 rpm (Delta spin is a nominal 100 rpm), depending on payload requirements.

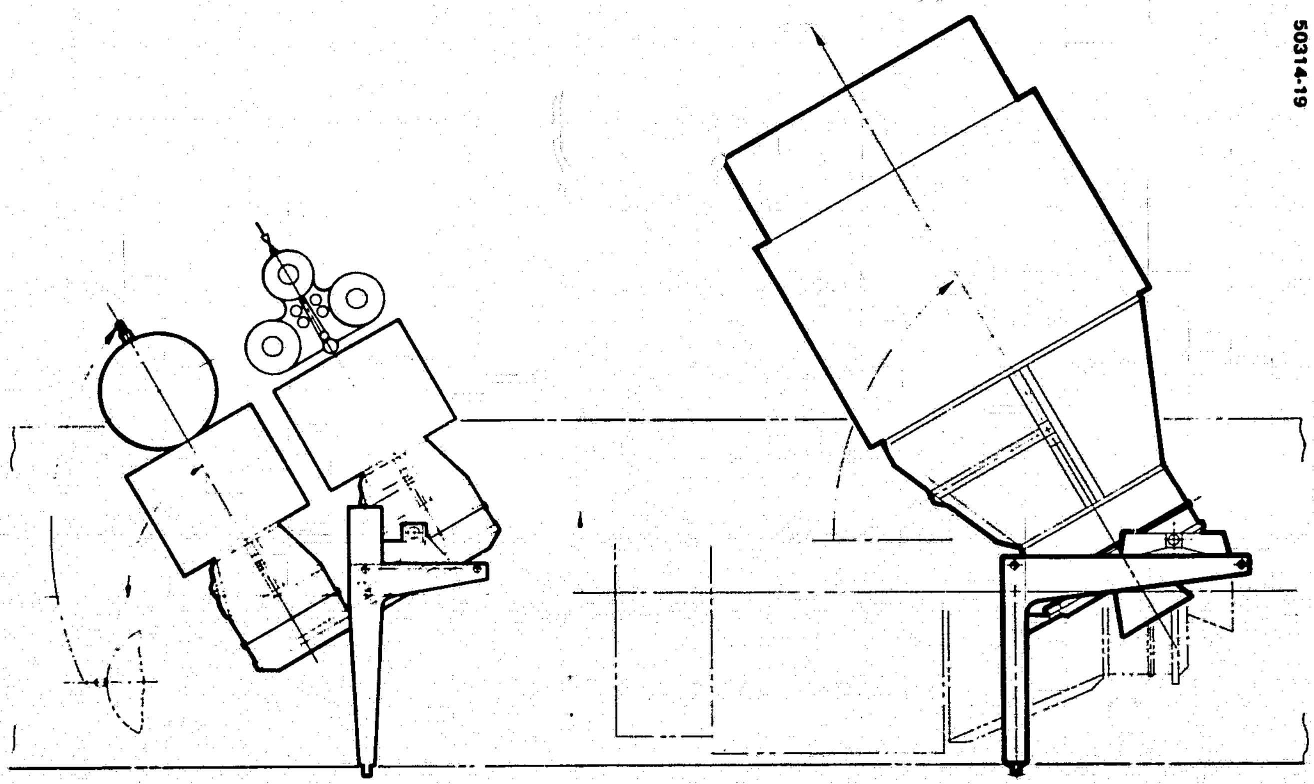


FIGURE 19. DELTA-CLASS TILT TABLE DEPLOYED

The tilt table is designed for retraction with either a single or dual payload attached. The payload bay doors can be closed with the tilt table rotated, but both payloads must be launched. In the event either payload cannot be launched and the tilt table cannot be retracted, the remote manipulator system can be used to discharge the entire cradle and payloads.

A preliminary analysis indicates that collision between the two payloads due to shuttle pitch rate induced by the separation impulse of the first payload to the separated is not a serious threat.

A detailed design of the cradle concepts presented is required; however, sufficient design and analysis were done in the study to determine initial feasibility of the concept.

7. GUIDANCE CONSIDERATIONS

7.1 BASELINE ERROR BUDGET

There are four basic sources of errors in the parameters of the initial geosynchronous orbit: PKM attitude errors, PKM velocity increment errors, AKM attitude errors, and AKM velocity errors. (Shuttle navigation errors will not significantly affect the synchronous orbit.) The significance of the orbital errors caused by these four factors is that RCS fuel must be provided in the spacecraft to correct the errors.

Of the four error sources, only PKM attitude error can be affected by the design of the perigee stage and deployment systems. Accuracy of the PKM velocity increment and the AKM velocity increment is limited by solid rocket motor variations. The AKM pointing error is a function of the spacecraft transfer orbit attitude determination and precession accuracy, although spin speed is also a factor. The elements of the PKM attitude error are listed in Table 10.

TABLE 10. BASELINE ERROR BUDGET

Error Source	Error, 3σ		Payload Weight, Penalty, %	Delta Launch, %
PKM Attitude Errors:				
Orbiter attitude control	0.5 deg			
Orbiter thermal distortion	2 deg	Algebraic sum		
		5.1 deg	2.3	~1.0
PKM thrust unbalance	2 deg	RSS		
		2.8 deg	1.0	
Deployment error	0.6 deg			
PKM velocity increment	25 fps		1.7	25 fps
AKM velocity increment	15 fps		0.2	15 fps
AKM pointing error	0.75 deg		1.6	0.75 deg
Total worst case			6.7	3.5
Total rss			2.6	2.4

The 0.5° orbiter attitude control error is made up of inertial measurement unit (IMU) errors and control jitter. The orbiter thermal distortion error will probably depend on the location of the payload in the bay and the extent to which this error will be predicted and compensated. Although the effect of the orbiter thermal distortion error can be removed by installing a celestial reference in the payload or spin table, this extra complexity is not considered worth the accuracy improvement considering the modest increase in fuel required for correction of this error.

The PKM ignition transient error is due to tipoff of the payload angular momentum vector by the thrust offset of the PKM.

The only error source that is subject to design is the deployment error. This is basically the tipoff error resulting from unbalance and lateral forces of the separation springs. The Atlas-Centaur separation is a nonspinning separation and rates of no more than 0.1 deg/sec have been observed on a number of launches. The tipoff on Delta separations cannot be observed because of the 100 rpm spin rate. An additional source of attitude error is the angular reaction of the orbiter to the separation impulse. It can be shown that this error is negligible for the smaller payloads. Calculation of the magnitude of this error when large payloads are separated requires a more complete simulation than time permitted in this study.

7.2 DEPENDENCE OF POINTING ERROR ON SPIN SPEED

As shown in Figure 20, the PKM pointing error decreases as spin speed is increased. A spin speed of 30 rpm, which appears to be the minimum reasonable value, was assumed for the error budget presented previously in order to provide an upper bound on pointing error. The optimum spin speed will depend on the specific payload being launched. Many pay-

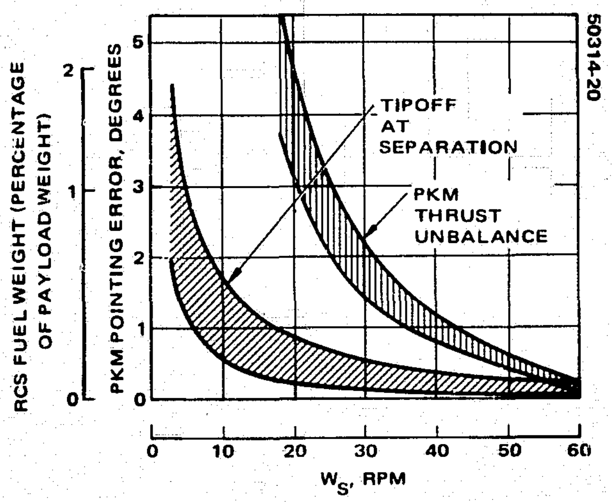


FIGURE 20. DEPENDENCE OF POINTING ERROR ON SPIN SPEED

loads will use a higher spin speed to minimize the fuel required to correct orbit insertion errors and, in some cases, because higher spin speeds are required for on-orbit operation. The use of a high spin speed does not appear to be a safety threat or operational inconvenience to the shuttle orbiter. Spinup can be accomplished over a sufficiently long time that reaction torques are not significant. A potential reason for not increasing spin speed is the effect on payload nutation before PKM firing. At lower spin speeds, the payload has a longer dedamping time constant and the nutation angle builds up more slowly, possibly eliminating the need for perigee stage ANC. Another potentional reason to limit spin speed is the need of three-axis satellites to be despun before beginning on-orbit operations.

7.3 SPACECRAFT RCS FUEL REQUIRED TO CORRECT FOR PKM ATTITUDE ERRORS

Since the PKM/AKM concept uses unguided stages, the errors of each stage must be corrected when the spacecraft reaches its orbit after AKM firing. All geostationary spacecraft have on-orbit control systems to make on-orbit corrections; thus, the cost of launching errors can be directly translated into spacecraft RCS fuel required to correct these errors.

The spacecraft RCS fuel required to correct perigee kick motor attitude errors is plotted in Figure 21 as a function of PKM pointing error. This is an area addressed in some detail in the study. The Delta 3σ errors for the third stage firing are shown, whereas in actual practice the Delta pointing errors have been undetectable for the launches for which data were available. The specification for the orbiter is 0.5° attitude control error maximum and 2° orbiter bay structural deformation error maximum. The error for the baseline deployment mechanism is estimated to be 0.6°. The error for the misalignment of the PKM thrust vector resulting from mechanical alignment of the motor to the vehicle and misalignment between the thrust vector and the motor is 2°. The algebraic sum of these errors results in 2.3 percent of the on-orbit spacecraft weight for additional RCS fuel requirement in the worst case. In fact, these errors should be root sum squared (rss) instead of added algebraically. The actual orbiter errors, the deployment mechanism errors, and PKM thrust misalignment errors will only be known with reasonable certainty after the hardware is built and tested. The assumed maximum errors and the algebraic adding of the errors are conservative maximum error estimates.

For reference, the PKM velocity errors and the AKM pointing and velocity errors require provisions for 3.4 percent of the spacecraft on-orbit weight in RCS fuel if the PKM pointing is perfect. This is typical of RCS fuel contingency used in geostationary spacecraft now.

RCS fuel is important in long life commercial communication spacecraft because the amount of RCS fuel sets a limit for useful spacecraft life. The RCS fuel contingency required for PKM/AKM launch from the STS requires refinement as actual test data become available. It is significant, however, that the maximum error assumptions do indicate acceptable performance for launch accuracy.

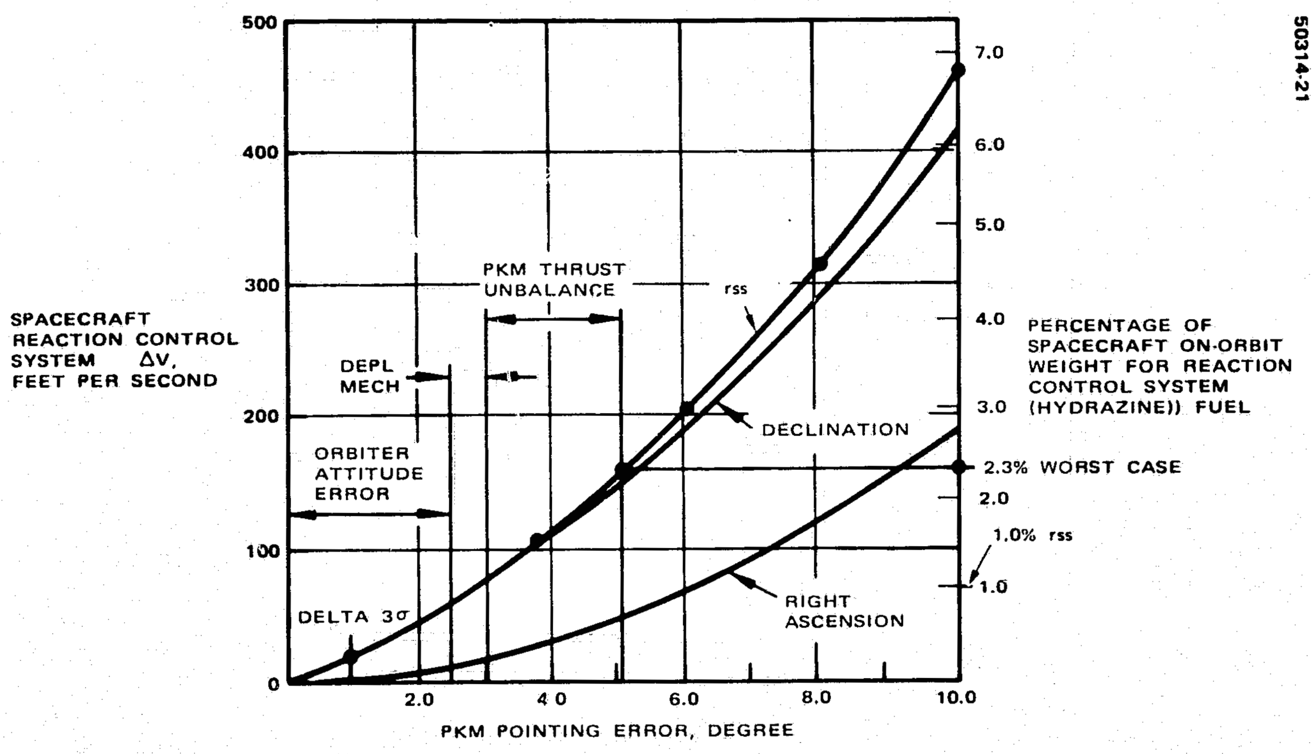


FIGURE 21. SPACECRAFT RCS FUEL REQUIRED TO CORRECT FOR PKM ATTITUDE ERRORS

8. DEPLOYMENT OPTIONS

8.1 ALTERNATE TECHNIQUES

Although the baseline approach to deployment appears optimum in the sense that its elements and the associated dynamics are well understood and it places the main deployment burden in the orbiter where the hardware is reusable and weight is less critical, other options that may simplify the required hardware and operation are worth considering. The rather noncritical nature of pointing accuracy indicated in Section 7 permits consideration of a variety of deployment options (see Figure 22).

The baseline can be modified by eliminating the spin table and providing the payload with the capability for a free body spinup using cold gas jets or hydrazine thrusters. This approach avoids the existence of a rapidly spinning heavy body attached to the orbiter and eliminates a fairly heavy mechanism that must be despun after payload separation; however, this approach requires additional expendable hardware on the payload. Also, the spinup must take place a very short time after separation to avoid degrading the pointing accuracy. Although this approach appears acceptable, the cost of providing the expendable spinup system on each of the many payloads exceeds the savings which results from elimination of the spin table.

Use of the remote manipulator system (RMS) was eliminated because of the poor pointing accuracy that can be expected with this relatively flexible device. To maintain reasonable accuracy, an attitude reference must be provided. The simplest attitude reference would be a momentum wheel spun up in the payload before separation. This device would reduce the resulting attitude errors, but the payload would need to be lifted nearly vertically until release to avoid large nutation angles. This would be a complex operation. Also, the use of the RMS imposes significant constraints on the location of the payload cg in the shuttle payload bay.

The vertical impulse ejection and lift arm methods of deployment are discussed in this section.

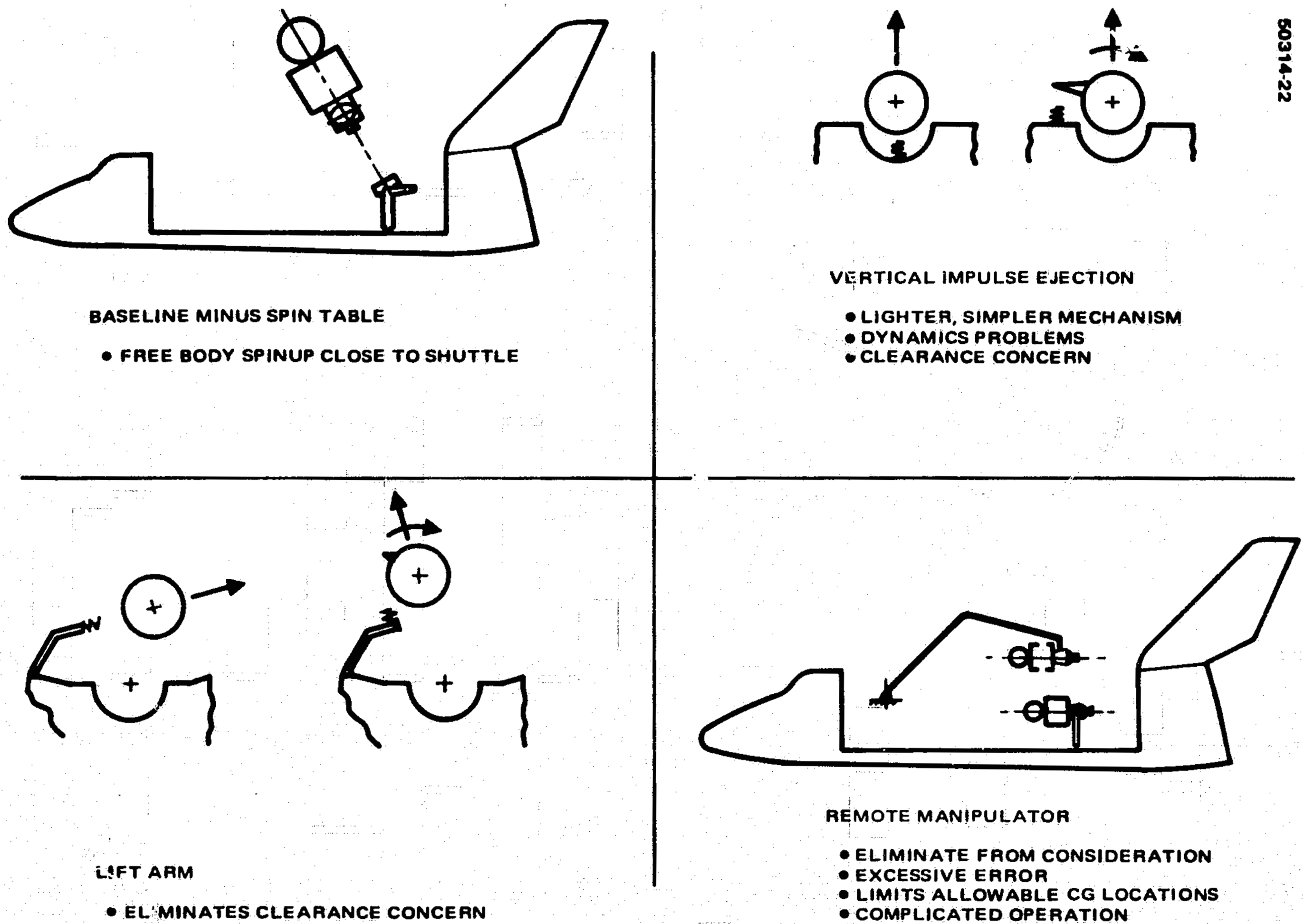


FIGURE 22. DEPLOYMENT OPTIONS

8.2 VERTICAL IMPULSE EJECTION

An approach that employs a simpler, lighter, less expensive deployment mechanism is illustrated in Figure 23. The payload is mounted on the cradle in the same manner as in the baseline concept. A spring or other impulsive device applies an impulse vertically through the spacecraft center of gravity. The payload then rises slowly out of the bay. A velocity of 0.2 fps (0.6 m/sec) was used in the study to avoid the safety implications of a high velocity separation. After clearing the shuttle bay, the payload is spun up by its spin jets. As shown in Figure 23, the spring can be mounted to provide an impulse through the nominal payload cg or it can be displaced laterally to provide spinup torque along with the vertical translation impulse. The spinup torque provides a small amount of angular momentum to stabilize the payload attitude until free body spinup is performed. The 0.2 fps (0.6 m/sec) separation velocity is accomplished by about 3.5 rpm payload spin rate for the offset spring case.

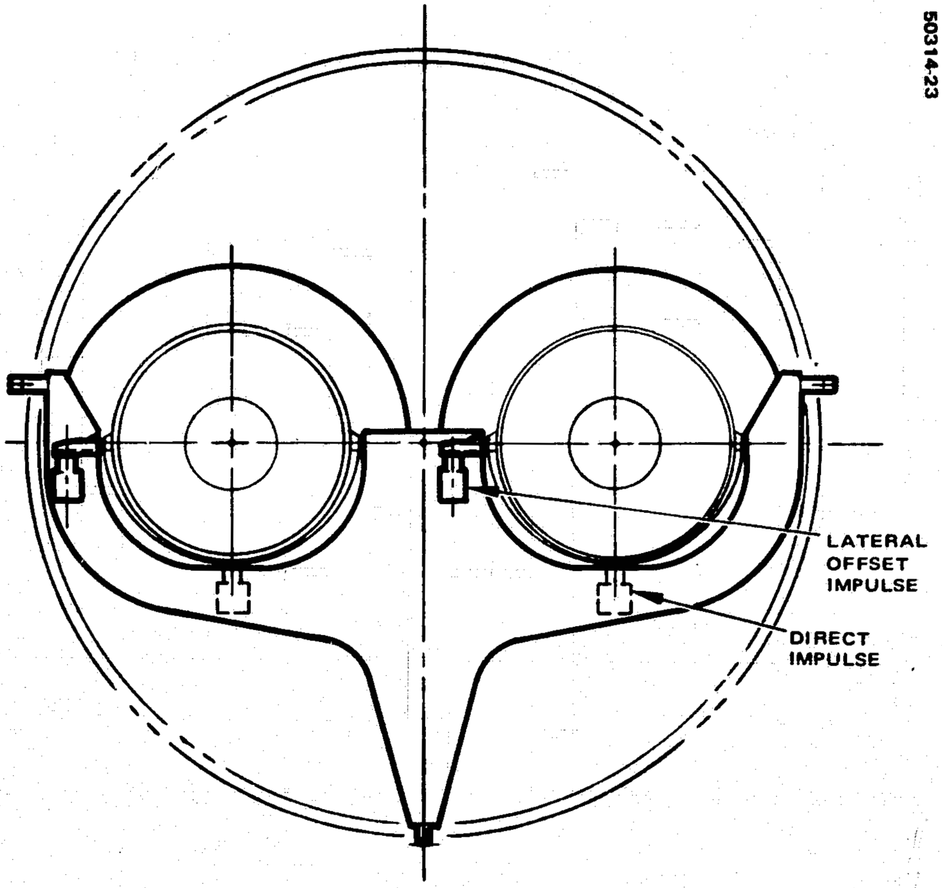


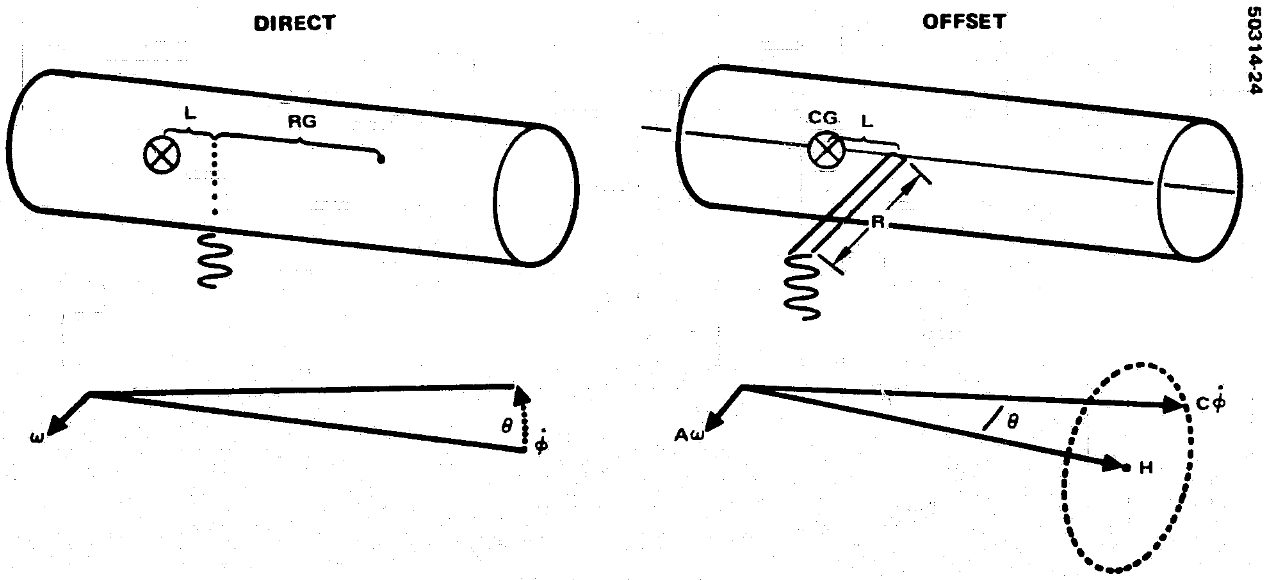
FIGURE 23. DUAL DELTA-CLASS VERTICAL IMPULSE INJECTION

Because the payload is being ejected directly from the bay, the payload dynamics become a significant safety factor. Preliminary analysis indicates that these two approaches pose no threat of recontact; however, they do not have the extensive flight background that the baseline deployment scheme possesses. Also, because of low separation velocity, a shuttle maneuver is required to acquire adequate separation for PKM firing.

Although an expendable spinup capability is required on the payload, it is likely that the offset spring approach can eliminate perigee active nutation control. The low spin rate of 3.5 rpm is accompanied by a long dedamping time constant so that nutation builds slowly. The acquisition of the full spin rate by free body spinup can be delayed until just prior to PKM firing.

8.3 VERTICAL EJECTION ANGULAR DYNAMICS

Figure 24 illustrates the angular dynamics of the two vertical ejection techniques. The direct spring pushing through a position axially displaced from the spacecraft cg imparts to the payload an angular rate about the orbiter pitch axis. The pitch rate corresponding to the parameters in the



WORST CASE CONFIGURATION

L = ±0.5 In. (1.3 cm)

R = 3 ft (0.92 m)

R_G = 3 ft (0.91 m) = RADIUS OF GYRATION ABOUT PITCH AXIS

$$\theta = wT = \frac{L}{R_G^2} \; H\,sep$$

$$= 0.005 \; H\,sep$$

• FOR $\theta = 0.77°$ H sep = 3 ft (0.92 m)

$$\bullet\, \theta = \frac{L}{R} = \quad \bullet\, \theta = \frac{L}{R} = 0.77 \text{ deg}$$

FIGURE 24. SPRING ANGULAR DYNAMICS

figure is about 0.06 deg/sec, which presents no threat of recontact with the orbiter. The payload pitch error will grow linearly with time until the spinup jets establish an angular momentum vector. The resulting error will depend on the time between separation and spinup. Because the pitch torque impulse imparted by the separation process will be proportional to the separation velocity, the error at spinup will be a function of the separation distance required at spinup.

In the case of the offset spring, the payload will acquire angular momentum from the separation process. The effect of an axial cg offset will be to produce a nutation of the payload as shown. The nutation angle, θ, will depend only on the ratio of the axial cg offset, L, to the intentional lateral offset, R. For a 0.5 inch (1.3 cm) cg offset the nutation angle is 0.77°. The payload spin axis will then nutate about the angular momentum vector, $\hat{H}$. Clearance between the payload and the orbiter should not be a problem because the nutation period is at least 90 seconds for the configuration considered and pitch rate is less than 0.1 deg/sec.

For the direct impulse to achieve the 0.77° error associated with the offset impulse, it would be necessary to commence spinup by the time the payload has moved 3 feet (0.92 m) from the stowed position. Because activation of the spin jets in such close proximity to the orbiter is undesirable from a safety point of view, the offset impulse approach is preferred. The effect on these dynamics of the liquid fuel in the spacecraft RCS, and perhaps in the AKM, is discussed in the next subsection.

8.4 EFFECT OF RANDOM ORIENTATION OF FUEL

Figure 25 illustrates the potential effect of a random orientation of the spacecraft liquid propellant on the offset spring deployment dynamics. If the fuel has migrated in the zero g environment to the top of the tank, a cg offset will result because the weight of the fuel will not be part of the system until the spacecraft propellant tank rises to meet the fuel. This cg offset will induce nutation as shown. For the range of configurations studied, the nutation angle could be as large as 3° for spacecraft with solid AKMs and 15° if a liquid apogee motor is used. When the propellant tank reaches the fuel, the torque impulse restores the angular momentum to the correct position. By this time the spin axis has rotated through a small angle and a new, smaller nutation angle is established. This analysis is simplified and does not account for the effect of the spin rotation on spacecraft propellant location. Although the random fuel orientation problem makes use of vertical impulse

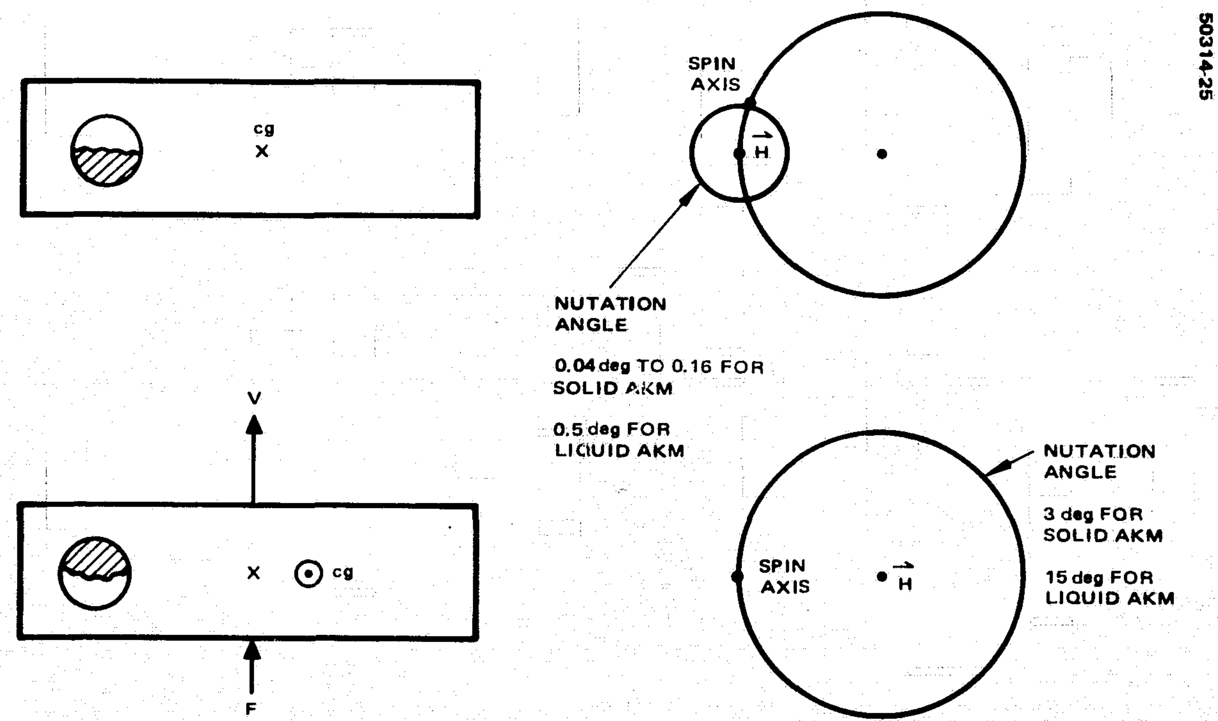


FIGURE 25. EFFECT OF RANDOM ORIENTATION OF FUEL

ejection questionable from a clearance point of view, it is likely that the fuel problem can be eliminated if the shuttle orbiter applies a vertical acceleration shortly before separation. There is no evidence of random fuel orientation in Atlas-Centaur payload separations.

8.5 LIFT ARM ALTERNATE

In the event of insufficient confidence in the clearance associated with the vertical impulse approach, the clearance problem can be eliminated by the use of an arm to lift the payload out of its stowed position before ejection as shown in Figure 26. The payload is then ejected by a direct or offset spring as described in the previous sections. The dynamics are essentially unchanged; however, because the payload is clear of the orbiter the threat of recontact is essentially eliminated. The lift arm is suitable to both small and large payloads, although the geometry of the arms for a dual launch is more complicated.

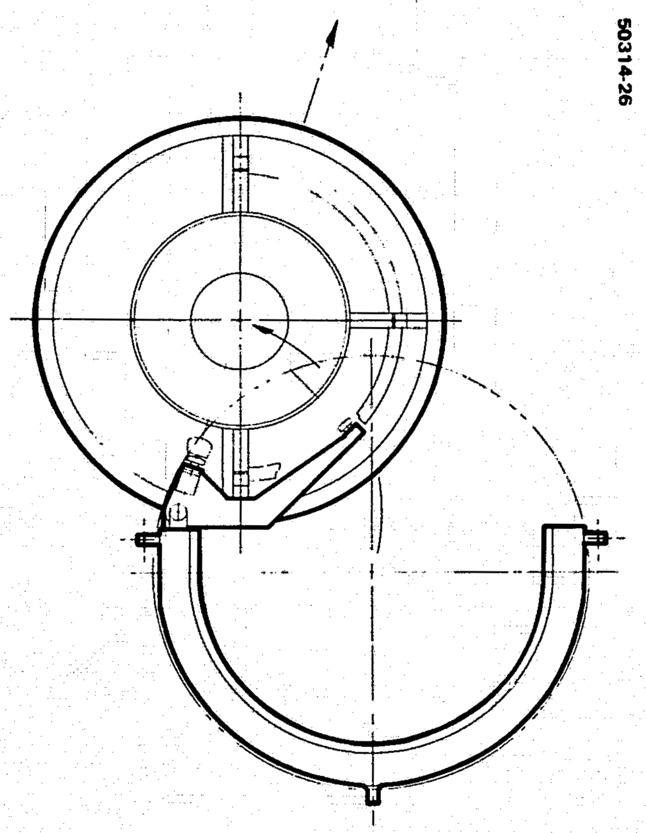


FIGURE 26. SHUTTLESAT CLASS LIFT ARM DEPLOYED

9. SHUTTLE MULTIPLE LAUNCH CAPABILITY

9.1 MULTIPLE LAUNCH IN SHUTTLE PAYLOAD BAY (Figure 27)

The division of the orbiter bay in halves and quarters shows the respective payload maximum weights would be 6150 pounds (2792 kg) and 2850 pounds (1294 kg). The volume available including the upper stages with each space would be 15 feet (4.56 m) in diameter by 30 feet (9.15 m) in length. The volume availability needs more detailed study after the cradle design is made. The current 8 foot (2.44 m) diameter Delta shroud and the 10 foot (3.05 m) diameter Centaur shroud limited spacecraft should be readily accommodated in the orbiter.

The actual mix of payloads would also depend on satisfying the orbiter cg constraints. Since volume constraints are most likely to be dominant considerations, the available weight capability could be used for ballasting and would permit numerous valid payload combinations.

A mix of heavier low orbit spacecraft and synchronous orbit spacecraft in a single orbiter flight might result in a more optimum use of the orbiter weight capability. This consideration requires further study.

The PKM/AKM concept where each payload includes its own upper stage means NASA would have the flexibility to mix payloads and utilize the orbiter capability in a more optimum mode.

9.2 SHUTTLE ORBITER CAPABILITY

Table 11 gives the parameters of the shuttle orbiter capability that affect the payload configurations carried on a shuttle mission. The volume parameters are by far the most constraining of the parameters for the types of payloads that are the subject of this study.

9.3 PAYLOAD CG LIMITS (Figure 28)

An additional significant constraint on payload installation is the constraint on total shuttle payload cg constraint, particularly the axial constraint. While the later cg constraint is very tight, it is not particularly

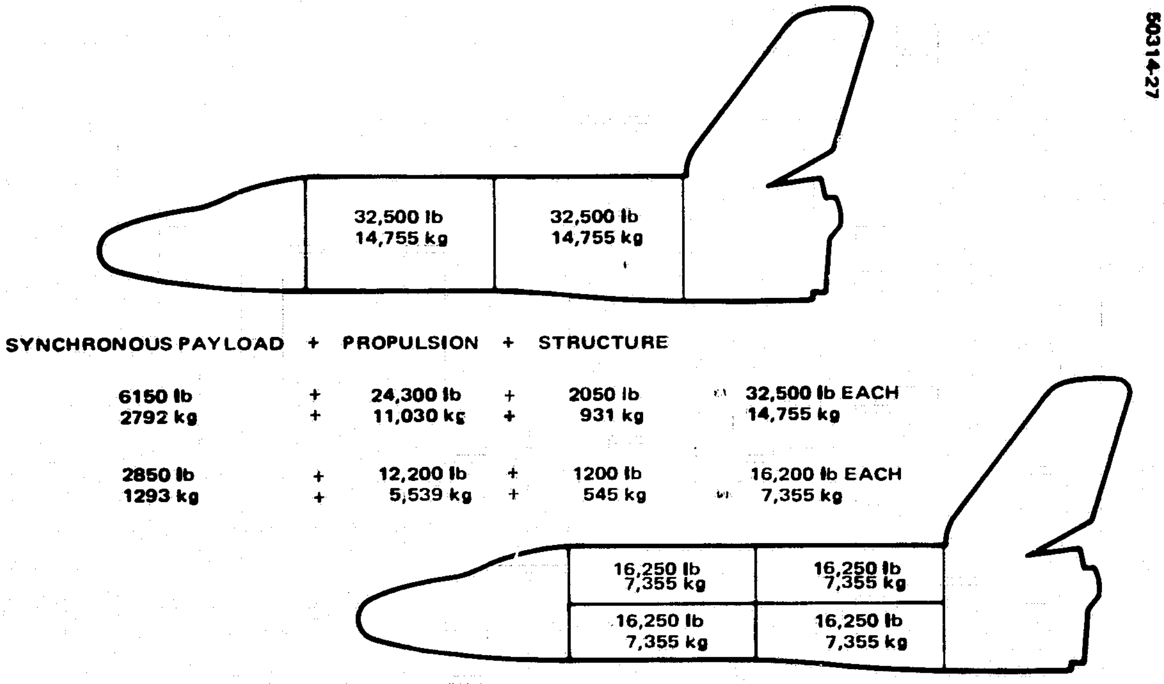


FIGURE 27. MULTIPLE LAUNCH CAPABILITY

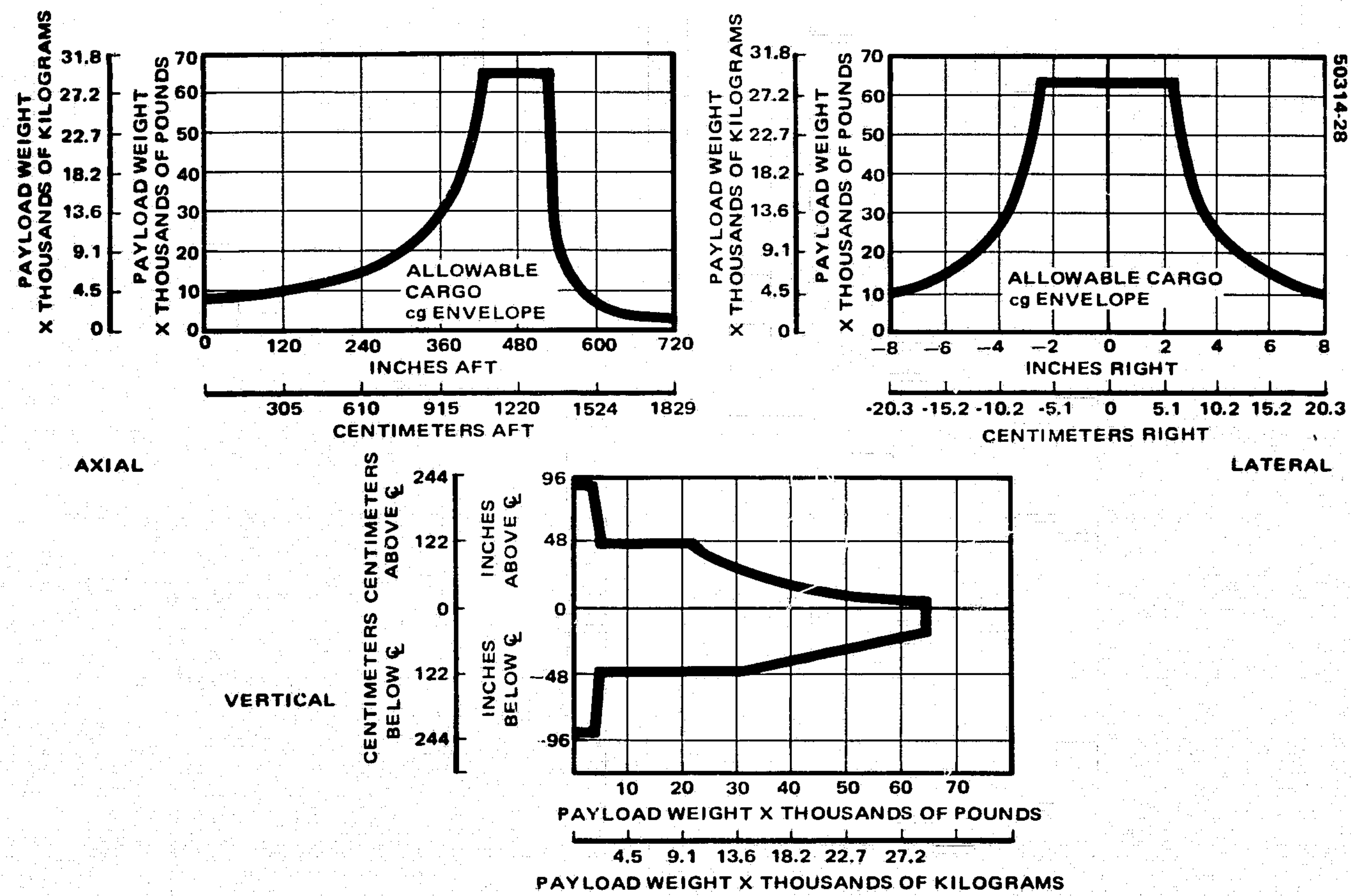


FIGURE 28. PAYLOAD CG LIMITATIONS

TABLE 11. SHUTTLE ORBITER CAPABILITY

Parking orbit injection capability	65,000 lb (29,500 kg) in 160 n.mi. (296 km) circular orbit; i = 28 deg
Orbiter bay volume cylinder	15 by 60 ft (4.58 by 18.3 m)
Heat rejection and electrical power	7 kW 21,500 Btu/hr

difficult to balance the payload laterally. The primary impact of the lateral constraint is to make it unacceptable for the shuttle to land with one of a pair of side-by-side payloads.

9.4 EFFECT OF CG CONSTRAINTS

Figure 29 illustrates the effect of cg constraints on the installation of two half-shuttle payloads. For the particular spacecraft configuration used as a model for this analysis, the composite cg was about 1 foot (0.3 m) outside the allowable cg range. There are three ways to bring the center of gravity into the acceptable region:

1) Reduce the weight of the payloads 7500 pounds (3409 kg) each.

2) Transfer 1500 pounds (682 kg) from each payload to ballast at the aft end of the payload bay.

3) Reduce the length of the payloads by 28 inches (71 cm) so that forward payload can be moved sufficiently aft to bring composite cg into allowable range.

The cg problem described is peculiar to a particular hypothetical payload and is presented to illustrate a potential problem in the future. None of the specific payloads in the payload model discussed earlier is in this size class. Future half-shuttle payloads may or may not be limited by cg constraints depending on their specific configuration.

Figure 30 shows three installations of a group of four Delta-class satellites that, with their installation hardware, weigh a total of 22,800 pounds (10,350 kg). The curve under each installation drawing shows how much additional payload weight can be installed in the bay as a function of the cg location of the additional payload. In the installation of Figure 30a, only a small part of the available weight capability can be used because the required cg location is occupied by the four Delta-class payloads. For alternate arrangements in Figure 30b and 30c, the available 42,000 pounds (19,182 kg) can be installed without difficulty. In Figure 30b, the cg cannot be forward of the indicated allowable area because the 42,200 pounds (19,182 kg) allowed by the 65,000 pound (29,545 kg) weight limit would then be insufficient to bring the composite cg into the allowed region.

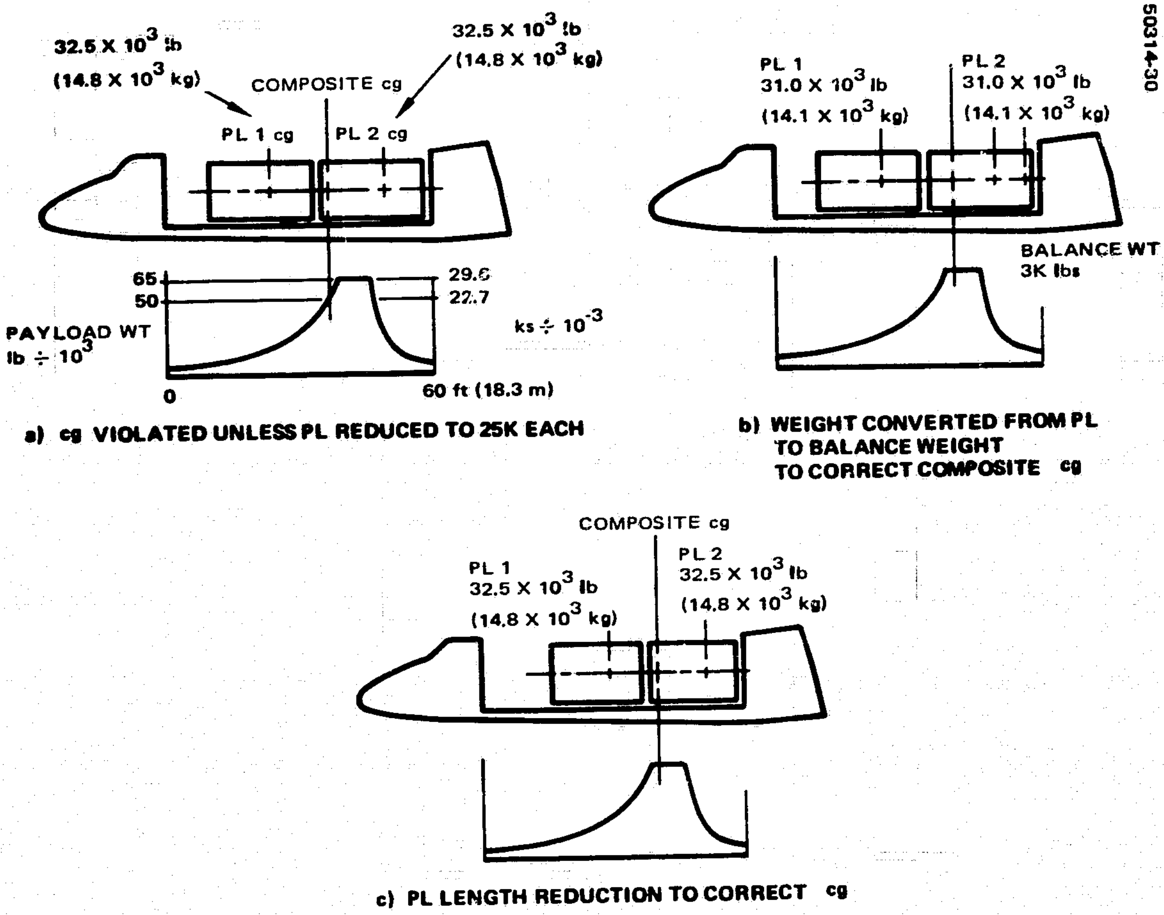
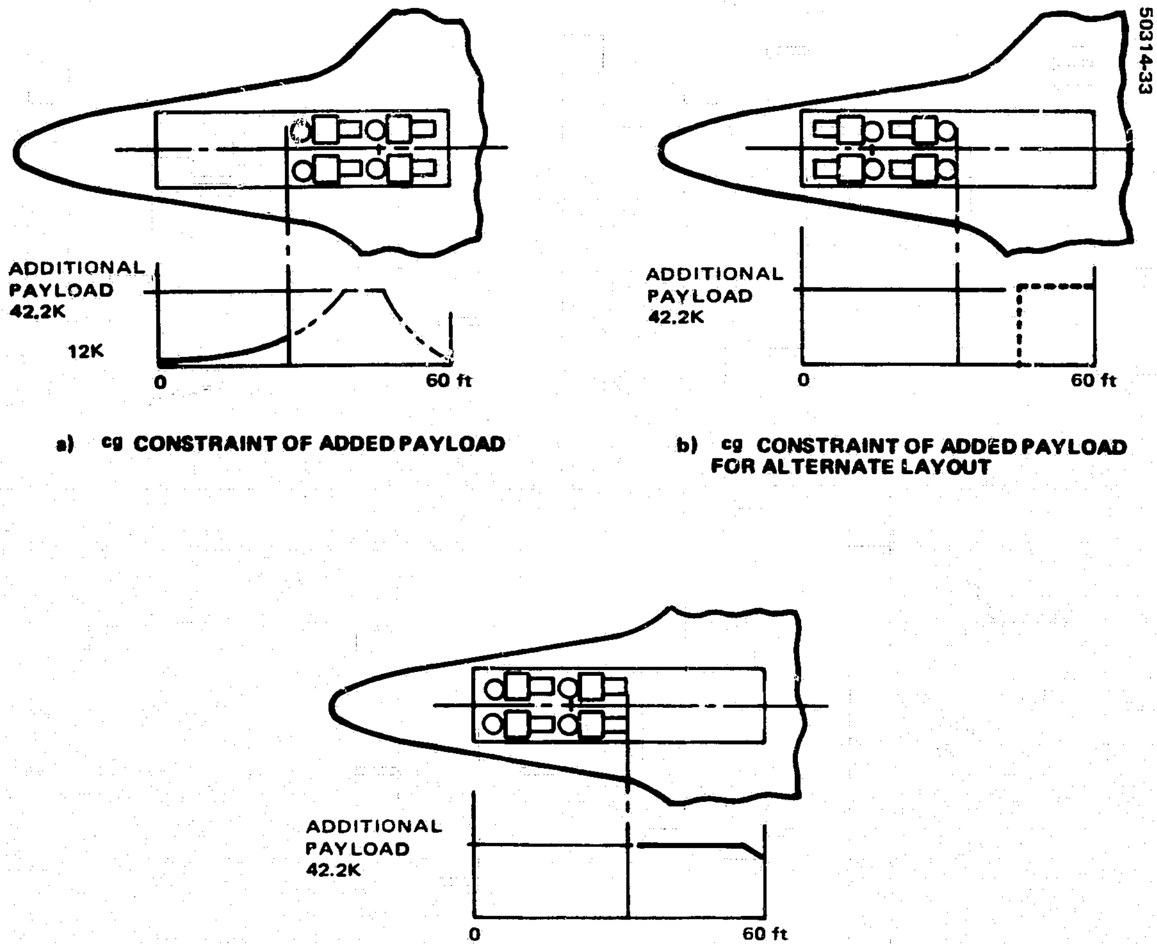


FIGURE 29. EFFECT OF SHUTTLE PAYLOAD CG CONSTRAINT



a) cg CONSTRAINT OF ADDED PAYLOAD

b) cg CONSTRAINT OF ADDED PAYLOAD FOR ALTERNATE LAYOUT

c) cg CONSTRAINT OF ADDED PAYLOAD -- ALTERNATE LAYOUT

FIGURE 30. MULTILAUNCH OF FOUR DELTA-CLASS PLUS ADDITIONAL PAYLOAD

10. SAFETY CONSIDERATIONS

Potential threats to the integrity of the orbiter and safety of its crew are listed in Table 12.

10.1 SHUTTLE ABORT

The requirement that the payload remain integral and attached to the payload bay fittings through a descent and landing can be met by straightforward design techniques. The weight analysis of the installation cradle and perigee stage structure was based on a 9 g crash load. The solid propellant is stable and eliminates the need to vent explosive fuel and oxidizer combinations. The PKM safe and arm device can be designed to be fail-safe, thus inhibiting potential PKM ignition by the 9 g crash load.

10.2 PAYLOAD WEIGHT AND CG DURING LANDING

It is not required that the payloads considered in this study be returned to earth because of spacecraft failure. In fact, no payload checkout is required after shuttle launch. Thus, under normal shuttle performance, the payloads will be ejected in orbit. The payloads could be in the orbiter bay at landing either under the shuttle abort conditions discussed above or if the ejection mechanism failed to eject a payload. The latter condition can be eliminated by design techniques and backup separation devices or removal of the entire payload installation, including the cradle, by the remote manipulator system. The abort situation has been discussed above; however, the possibility exists that an abort might occur after part of the payload complement had been deployed. The cg location of the remaining payload could be unacceptable if provisions are not made for this situation. To prevent a violation of the tight lateral cg constraint, dual launches in a vertical, rather than horizontal, arrangement have been recommended. The vertical cg constraint is easily met with the bottom payload remaining in the bay. The axial cg constraint for partial payload deployment can be accommodated by the way the payloads are arranged in the bay, the first payloads to be deployed being placed forward in the bay; however, this arrangement may result in a descent cg aft of the allowed region.

TABLE 12. SAFETY CONSIDERATIONS

Shuttle Abort
Vehicle designed to meet 9 g crash load
Solid propellant is stable
Payload Weight and cg During Landing
Payload return not required
Backup separation
Payload arrangement to satisfy cg constraints for abort after final deployment
Payload Collision With Shuttle During Deployment
Baseline provides positive separation
Vertical impulse approach required
SRM Explosion at Ignition
10^{-9} probability of particle impact at 3000 ft (915 m)
Contamination of Shuttle by Exhaust
Premature Ignition of PKM or AKM
Fail-safe design

10.3 PAYLOAD COLLISION WITH SHUTTLE DURING DEPLOYMENT

This threat has been minimized by selecting a baseline deployment scheme that raises the payload above the orbiter bay centerline and ejects it at an angle that provides a maximum clearance. Preliminary analysis indicates that recontact is not possible over the potential range of malfunctions of the separation mechanism. This problem, because of its importance, is a subject for more detailed dynamic analysis. Should the alternate approach that ejects the payload directly from the bay be considered, a thorough dynamic analysis would be required.

10.4 SOLID ROCKET MOTOR EXPLOSION AT IGNITION

The separation distance of 3000 feet (915 m), assumed in this study as a requirement for PKM ignition, provides a probability of 10^{-9} that a particle resulting from an explosion at ignition will impact the orbiter.

10.5 SHUTTLE CONTAMINATION BY PKM EXHAUST

It will be difficult for the shuttle orbiter to avoid the PKM exhaust with a 3000 foot (915 m) separation because the payload will move a distance much greater than 3000 feet while the PKM is burning. The effect of PKM exhaust on the shuttle at distances of 3000 feet is a factor to be considered.

10.6 PREMATURE IGNITION OF PKM OR AKM

A fail-safe design of the safe and arm system for the motors is required. Also, provisions must be made to prevent premature arming and firing signals being transmitted to the PKM. Coincidence between two independent timers would be required for an actuation. The spacecraft AKM firing is commanded by the users' ground station over the spacecraft command link. This station is unlikely to have visibility of the spacecraft before injection into transfer orbit; however, the absence of this command must be ensured.

11. TOTAL ESTIMATED COST FOR GEOSTATIONARY PAYLOAD DELIVERY BY PKM/AKM TECHNIQUE

The total cost for geostationary payload delivery by the PKM/AKM technique (Table 13) is estimated in terms of the cost, assuming existing spacecraft are launched and assuming certain costs would be saved if a Tug delivery is available.

Existing payloads (i.e., payloads designed for either Delta or Centaur launch) would require a PKM stage consisting of a PKM solid rocket motor, stage mechanical structure, and stage support electronics. The Delta-sized payloads using TE-364-4 PKM would require development only of the stage structure and support; hence, nonrecurring cost would be approximately $800,000 and a recurring cost $300,000 for support and structure plus $190,000 for the motor. The total recurring cost would be less than $500,000. The Centaur-sized payloads using the Minuteman III would be approximately the same. If a new solid rocket motor were developed for a Delta 3914 class up to a 6000 pound class, the range of costs is shown. The nonrecurring costs would range from $4.8 million to $8 million and the recurring cost from $700,000 to $1.05 million. The estimated recurring cost for the largest PKM stage compatible with the STS would be approximately $1 million using the PKM concept.

TABLE 13. USER COSTS FOR USER PROVIDED UPPER STAGE

	Estimated Cost					
	RDT & E, $M			Unit, $K		
PKM	0 (1)	4.0 (2)	7.0 (3)	190 (1)	400 (2)	550 (3)
Stage structure	0.3	0.3	0.4	150	150	300
Stage support	0.5	0.5	0.6	150	150	200
PKM stage	0.8	4.8	8.0	490	700	1,050
AKM				120	120	250
Spacecraft support (3-axis case)	0.05	0.05	0.06	150	150	200
PKM/AKM total	0.85	4.85	8.06	760	970	1,500

(1) Delta-class using TE-364-4 or Centaur-class using Minuteman III solid rocket motor.
(2) Delta-class using new solid rocket motor.
(3) Half-orbiter payload class with new solid rocket motor.

When the PKM/AKM concept is compared to the Tug concept, the AKM costs must be added to the PKM stage costs because the Tug performs both functions. The comparable costs are a maximum of $8.06 million for the PKM/AKM concept nonrecurring and $1.5 million recurring costs for the 6000 pound (2727 kg) class spacecraft. For a Delta- or Centaur-sized payload, the PKM/AKM costs are significantly smaller.

The cost for the orbiter mounted cradle and tilt table was not estimated. A detail design is required before a reasonable estimate can be made. A significant consideration, however, is that the cradle and tilt table can be reused many times and its design should consider this reuse philosophy. The cost per flight of this facility would amortize the original investment. The cost per flight would be small (for example $100,000) if the purchase cost were as high as $10 million and 100 uses were assumed.

12. PROGRAM PLAN FOR COST COMPETITIVE STS FOR GEOSTATIONARY PAYLOADS

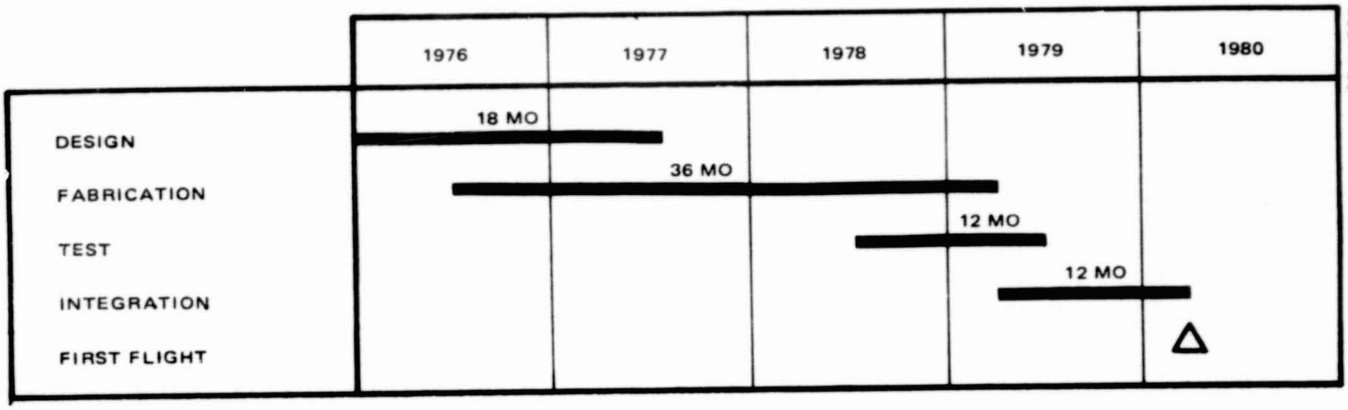
A program plan for the PKM/AKM concept, as shown in Figure 31, was developed with two assumptions. First, the PKM/AKM capability was to be available in the first half of 1980. Second, development of a new PKM stage solid rocket motor may be desired by NASA or DoD in order to capture payloads planned for the Delta 3914 and Titan IIIC Transtage.

Another critical consideration is that it is judged unlikely that a reimbursable user (i.e., commercial company or foreign nation) would make the necessary investment for a PKM/AKM or any upper stage development at this time. NASA or DoD must, therefore, make the initial investment for the required capability and cause the developed stages to be available in the marketplace. In the future, as RCA and McDonnell Douglas are now doing with the Delta 3914, it is highly probable users will develop special PKM/AKM stages matched to their specific needs.

The tasks required are design and fabrication of the payload support structure and payload deployment mechanism for the orbiter and the design and fabrication of the PKM stage including the motor, the RCS, and the structure for payload classes such as Delta, Centaur, etc.

The design issues should be solved in 18 months, the PKM motor development (if it is a new development) could take 36 months, testing could take 12 months, and integration with the orbiter could take 12 months. As the program plan indicates, some overlap is required if the time spans are correct and the desired delivery date is in the first half of 1980. If the assumptions are correct, the program should start in early 1976.

A token program, assuming Delta-class vehicles with TE-364-4-only capability, could be initiated later and this single-point capability could be demonstrated in early 1980. A full service capability for other payload classes would then be developed for the post-1985 period.



TASKS

ORBITER	PAYLOAD SUPPORT CRADLE PAYLOAD DEPLOYMENT MECHANISM
PAYLOAD	PKM STAGE SUPPORT STAGE STRUCTURE

FIGURE 31. PROGRAM PLAN

13. SUMMARY

Results of this study show the PKM/AKM concept provides a cost competitive STS capability for the geostationary payloads. The PKM/AKM concept has:

1) Lowest nonrecurring cost of any upper stage program known

2) Recurring cost totally paid by the user

3) Maximum flexibility in the user's upper stage design

4) Least impact on the orbiter of any upper stage program

NASA must, however, organize the multiple payloads by facilitating and establishing the appropriate management procedures, and most important, price the launch service equitably.

The PKM/AKM concept provides the transition capability from Delta and Centaur to the STS more readily than any other known alternative. NASA can use this feature to capture the large number of reimbursable launches.

NASA, therefore, needs to initiate development of the previously described hardware, payload hardware, and establish a capture plan.

A proposed capture plan would be to determine which payloads in development for either Delta or Centaur launch require launch in the 1980 time period. Some suggested targets are the NASA TDRSS; the NOAA GOES; a large number of commercial domestic satellites such as Anik, WESTAR, RCA, etc.; the Intelsat V (currently in procurement process for 15 spacecraft); the DoD FLTSATCOM, etc., which will need replenishment. These spacecraft can all be launched with their respective launch vehicles or could be moved to the STS if the capability is available and the price is right.

An important factor to all the users is the STS with a PKM/AKM concept can be fully backed up by the existing launch vehicles in the event the STS orbiter is delayed or encounters a long standdown period in the initial phases of its operational employment.

The only NASA spacecraft designed for STS launch to geostationary orbit is the STORMSAT, which is baselined for an interim upper stage or Space Tug launch.

Clearly, the PKM/AKM concept is NASA's best hope for capturing reimbursable geostationary payloads in the 1980s.

14. SUPPORTING RESEARCH AND TECHNOLOGY

A major conclusion of this study is that the technique described in this report for placing satellites in geostationary orbit can be implemented within the current state of the art and that no supporting research and technology effort is required.