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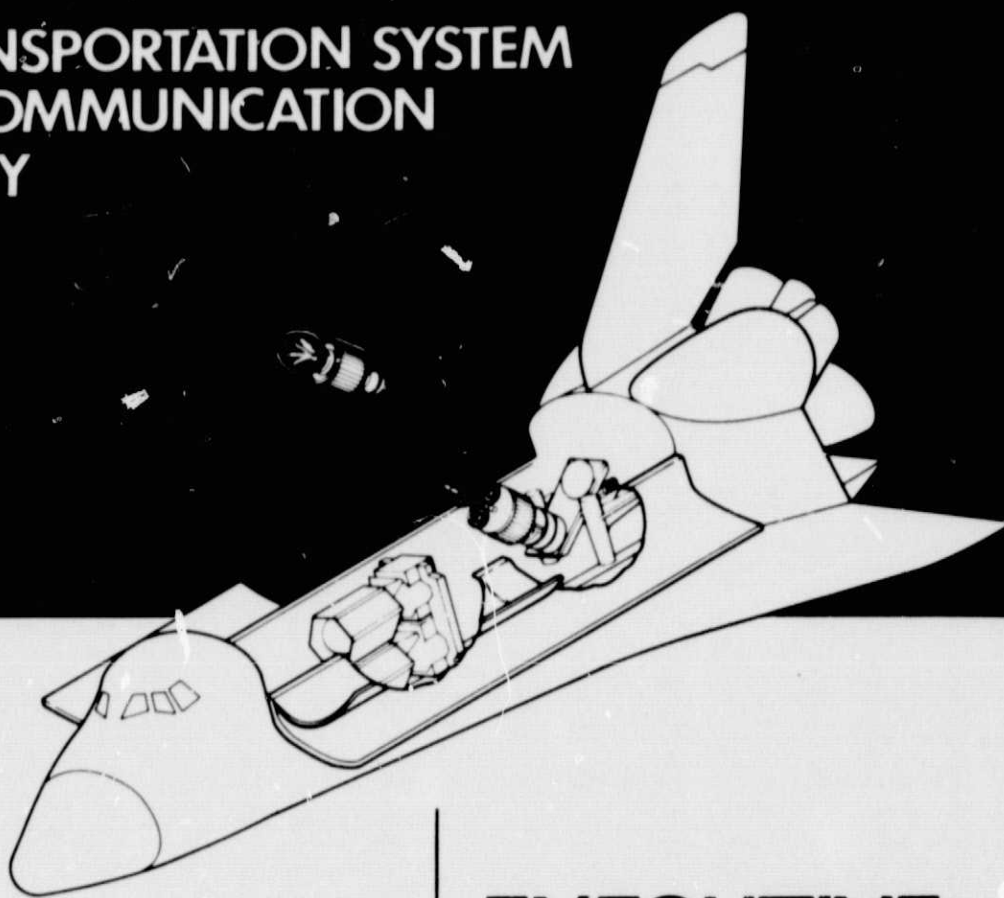
NASA CR-144015

OCTOBER 1975 NASA Contract NAS 8-31435

UTILITY OF SPACE TRANSPORTATION SYSTEM TO SPACE COMMUNICATION COMMUNITY



(NASA-CR-144015) UTILITY OF SPACE TRANSPORTATION SYSTEM TO SPACE COMMUNICATION COMMUNITY: EXECUTIVE SUMMARY (Hughes Aircraft Co.) 22 p HC $3.25 CSCL 17B

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EXECUTIVE SUMMARY

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE AND COMMUNICATIONS GROUP

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INPUT BRANCH

MARSHALL SPACE FLIGHT CENTER

Data Procurement Document No. 514
Data Requirement No. MA-05

Hughes Ref. No. D5221-/
SCG 50313R

OCTOBER 1975 NASA Contract NAS 8-31435

UTILITY OF SPACE TRANSPORTATION SYSTEM TO SPACE COMMUNICATION COMMUNITY

EXECUTIVE SUMMARY

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MARSHALL SPACE FLIGHT CENTER

H.R. Gangl, Jr.
Program Manager, NASA

L.M. Brorstein
Program Manager,
Hughes Aircraft Company
Space and Communications Group

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE AND COMMUNICATIONS GROUP

Hughes Ref. No. D5221

SCG 50313R

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INTRODUCTION

The utility of the Space Transportation System to the space communication community depends on the service and its 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. In this first 10 years, the business has 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, communication satellite systems with expendable launch vehicles continue to offer useful services to users at a competitive price.

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

This executive summary describes the results of a 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 goal of the study was to define a cost competitive STS for geostationary payloads.

The study concludes that the STS will be useful to the space communication community, as well as to other geostationary satellite system users, if NASA adopts the recommendations proposed in this report.

PATTERN STS LAUNCH SEQUENCE AFTER DELTA

The NASA Thor Delta launch vehicle in the 1970 to 1980 time period is employed 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 approximately 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. For a reasonable investment, a Delta-launched satellite system provides a useful service in a competitive market. Furthermore, the proven Delta launch sequence is relevant to a cost competitive STS — orbiter and upper stage — sequence (see Figure 1).

The Thor Delta with strap-on solid rocket motors and first and second stage engines places the Delta second and third stage plus satellite 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, orbiter-altitude circular orbit. With either the Delta or the orbiter, the next step is to prepare for injection of the payload into an elliptical transfer orbit.

The injection into transfer orbit with the Delta is initiated at the first equatorial crossing (the desired perigee) with a short burn of the Delta second stage, then a spinup of the third stage solid rocket motor and satellite with a second stage mounted spin table, separation of the spinning vehicle (third stage and satellite), and firing of the third stage solid rocket motor. The satellite is then separated from the Delta third stage into its transfer orbit with the perigee and apogee on the equator, with a nominal 28^{o} 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 a solid rocket motor upper stage and satellite, ejecting the payload vehicle at the proper time so that the perigee kick motor

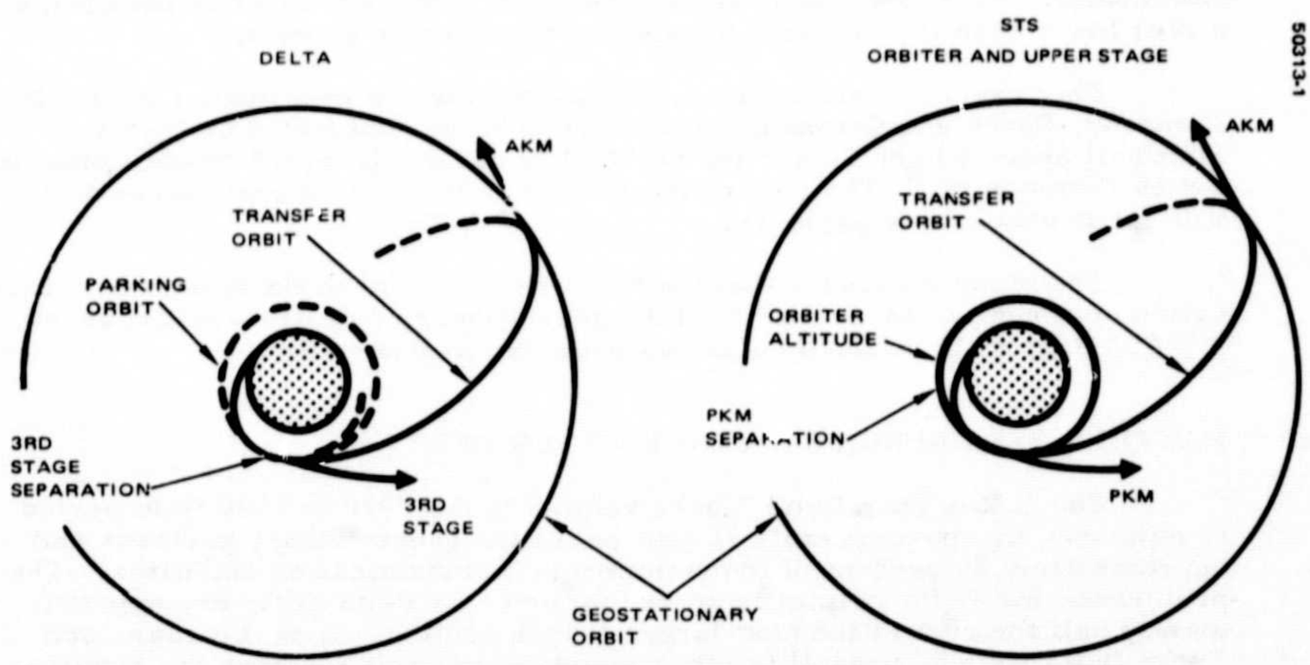


FIGURE 1. PATTERN STS LAUNCH SEQUENCE AFTER DELTA

(PKM) would fire at an equatorial crossing. The satellite 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.

The Atlas-Centaur launch vehicle with a second burn of the Centaur injects a satellite into the same type transfer orbit. After separation, the satellite is spun up by satellite-mounted jets.

All commercial communication satellites launched in the 1970 to 1980 time period are placed in geostationary orbit from a transfer orbit by a satellite-mounted apogee kick motor (AKM). The satellite is oriented while in its transfer orbit and the AKM firing is timed in order to remove the orbit inclination and to circularize the orbit at geostationary orbit altitude.

STS delivery of geostationary satellites can be patterned after the Delta concept, namely using spin stabilized PKM and AKM staging.

INSTALL PAYLOAD IN ORBITER WITH SINGLE CRADLE

The initial consideration for an STS launch sequence patterned after the Delta concept 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 in Figure 2 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.

The baseline is a single-cradle concept with two attachments on each orbiter bay longeron and a single attachment to the orbiter keel. The cradle would be aluminum construction and in accordance with the orbiter requirements as specified in Volume XIV, JSC 07700. 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 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 the spacecraft with AKM are nearly equal in weight, the center of gravity will generally be slightly forward of the PKM and the payload can be approximately balanced on the cradle. The launch loads are transferred to the cradle, thus precluding significant loads into the tilt table mechanism attached to the payload adapter aft interface.

The baseline cradle designed for Delta-sized spacecraft uses the same concept as shown in Figure 3. A significant design feature is the accommodation of two existing Delta-launched spacecraft without spacecraft modification. WESTAR and MARISAT spacecraft are used as examples.

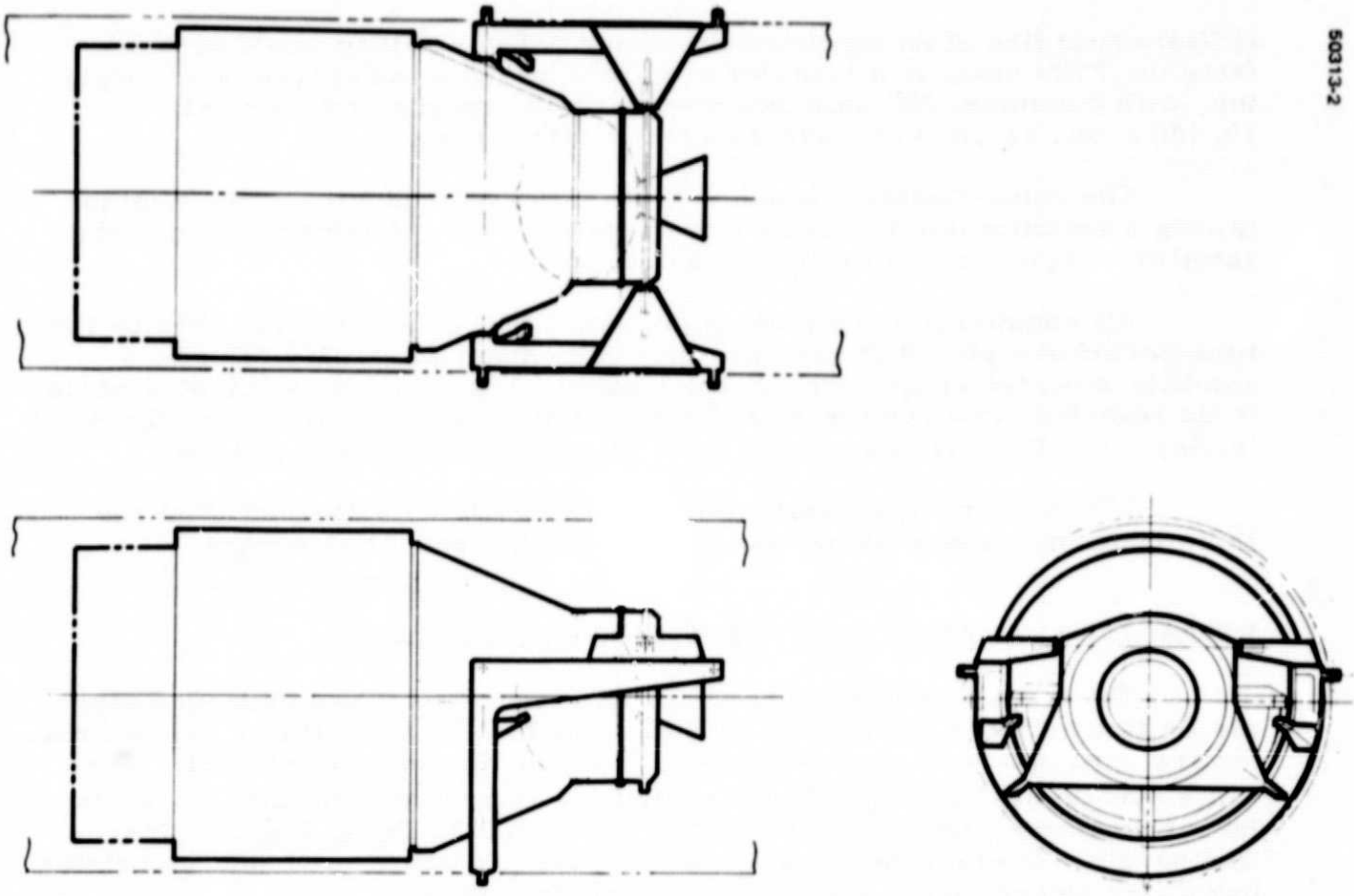


FIGURE 2. SINGLE-CRADLE CONCEPT FOR FULL ORBITER BAY DIAMETER CLASS PAYLOAD

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 with one spacecraft. A side-by-side arrangement violates the orbiter lateral center of gravity requirements for landing if one spacecraft is launched and the other is retained.

TILT AND SPIN PAYLOADS IN ORBITER BAY

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

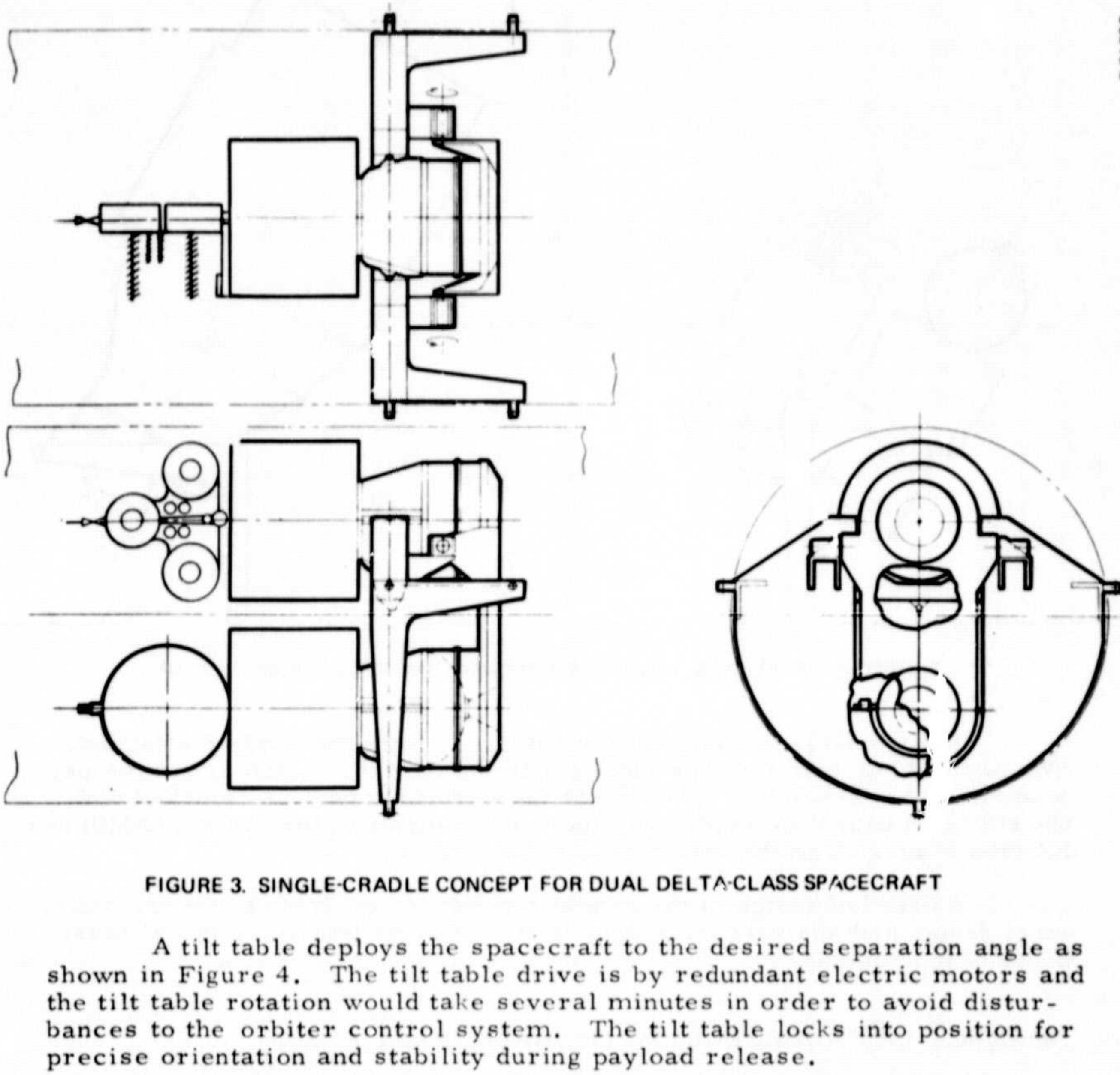


FIGURE 3. SINGLE-CRADLE CONCEPT FOR DUAL DELTA-CLASS SPACECRAFT

A tilt table deploys the spacecraft to the desired separation angle as shown in Figure 4. The tilt table drive is by redundant electric motors and the tilt table rotation would take several minutes in order to avoid disturbances to the orbiter control system. The tilt table locks into position for precise orientation and stability during payload release.

A spin mechanism mounted on the tilt table spins the payload to the desired speed, nominally 30 rpm, determined by spacecraft stability considerations.

The Delta-class spacecraft are extended on a common tilt table, 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. The payload is axially separated, similar to Delta separation, by a set of conventional separation springs. The tilt platform is designed with sufficient stiffness to hold attitude errors during separation to less than 0.6°.

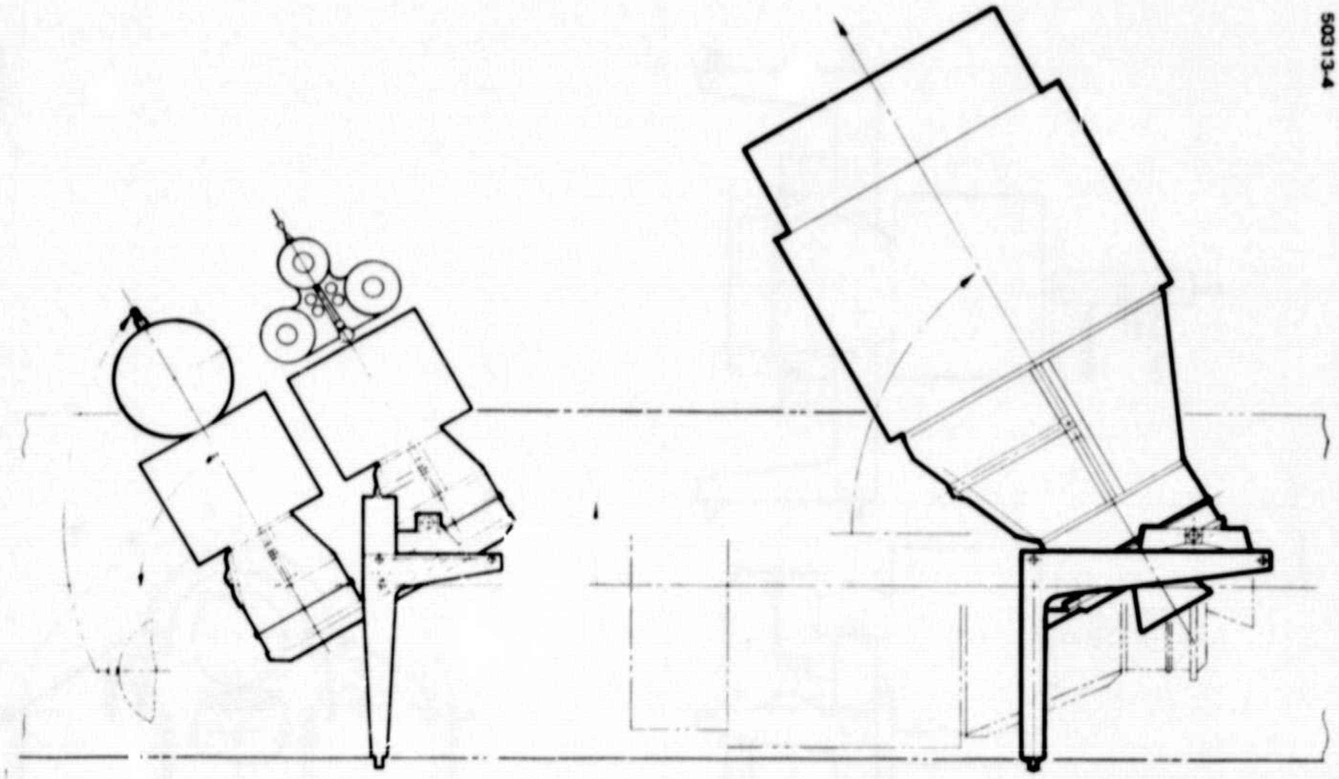


FIGURE 4. PAYLOADS TITLTED TO SEPARATION ANGLE AND SPUN UP

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

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

CORRECT LAUNCH ERRORS WITH SPACECRAFT FUEL

Since the PKM/AKM concept uses unguided stages, errors are introduced at each staging and the residual errors must be corrected when the spacecraft reaches geostationary orbit. All geostationary spacecraft have reaction control systems (RCS) to make on-orbit corrections; thus the cost of launching errors can be directly translated into spacecraft RCS fuel required to correct the residual errors.

Spacecraft RCS fuel required to correct PKM attitude errors are plotted as a function of PKM pointing error in Figure 5. The 3σ pointing errors for the Delta third stage are shown for comparison.

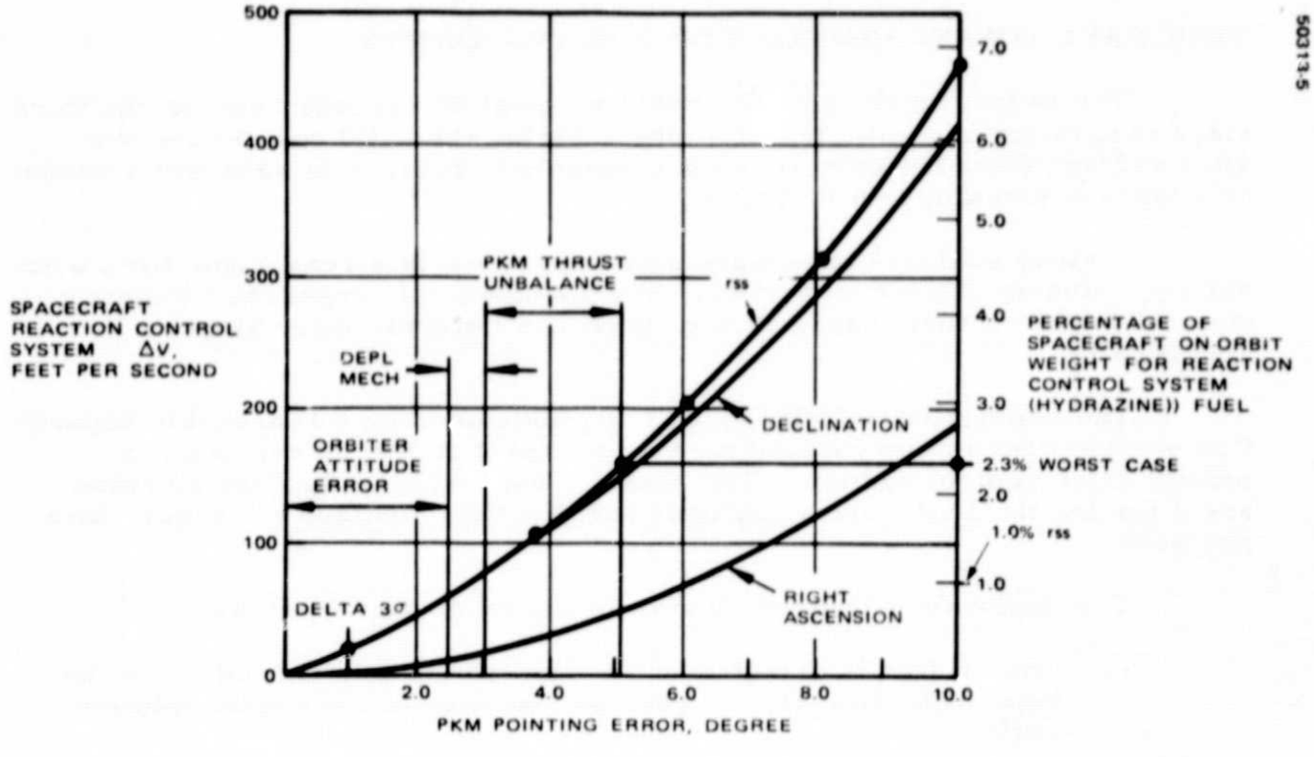


FIGURE 5. SPACECRAFT FUEL REQUIRED FOR CORRECTING PKM POINTING ERRORS

The specifications for orbiter attitude error are 0.5° attitude determination error maximum and 2° orbiter bay structural deformation error maximum. The error due to the deployment mechanism is estimated to be 0.6°. The error for PKM thrust unbalance resulting from mechanical misalignment of the motor to the vehicle and misalignment between the thrust vector and the motor case is 2.0°. The algebraic sum of these errors results in a 2.3 percent of the on-orbit spacecraft weight for additional RCS fuel in the worst case. In fact, these errors should be root sum squared (rss) and the additional RCS fuel percentage decreases to 1 percent. The actual orbiter attitude errors, the deployment mechanism errors, and PKM thrust unbalance 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 limits useful spacecraft life. The RCS fuel contingency required for PKM/AKM launch from the STS orbiter requires refinement as actual test data become available. It is significant, however, that the maximum error assumptions do indicate a tolerable RCS fuel penalty even though RCS fuel is a very valuable commodity in long life spacecraft.

TIME PAYLOAD SEPARATION FOR ORBITER SAFETY

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 without requiring an orbiter maneuver. The assumed safe separation distance was 3000 (915 m) feet, 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 for the Delta-class payloads and 2 fps for Centaur or larger class payloads.

The deployment concept, shown in Figure 6, 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) Orbiter payload doors are opened, the payload is rotated on the tilt table 45° to 60° (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.

3) Separation time will be determined so that the payload will be crossing the equator (the desired perigee) at the time of PKM firing. A V_{sep} = 2 fps will require payload separation 20 minutes before PKM firing; V_{sep} = 4 fps will require payload separation 13 minutes before 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. 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 (915 m) foot safe separation distance and the orbiter being nominally 20° from the solid rocket motor plume require additional analysis. The 3000 foot (915 m) separation was judged a safe distance in the event of motor explosion, but a more definitive analysis is required. If further analysis indicates a greater distance is required, both orbiter maneuvers or more time before PKM firing are possible; however, the baseline concept would remain valid.

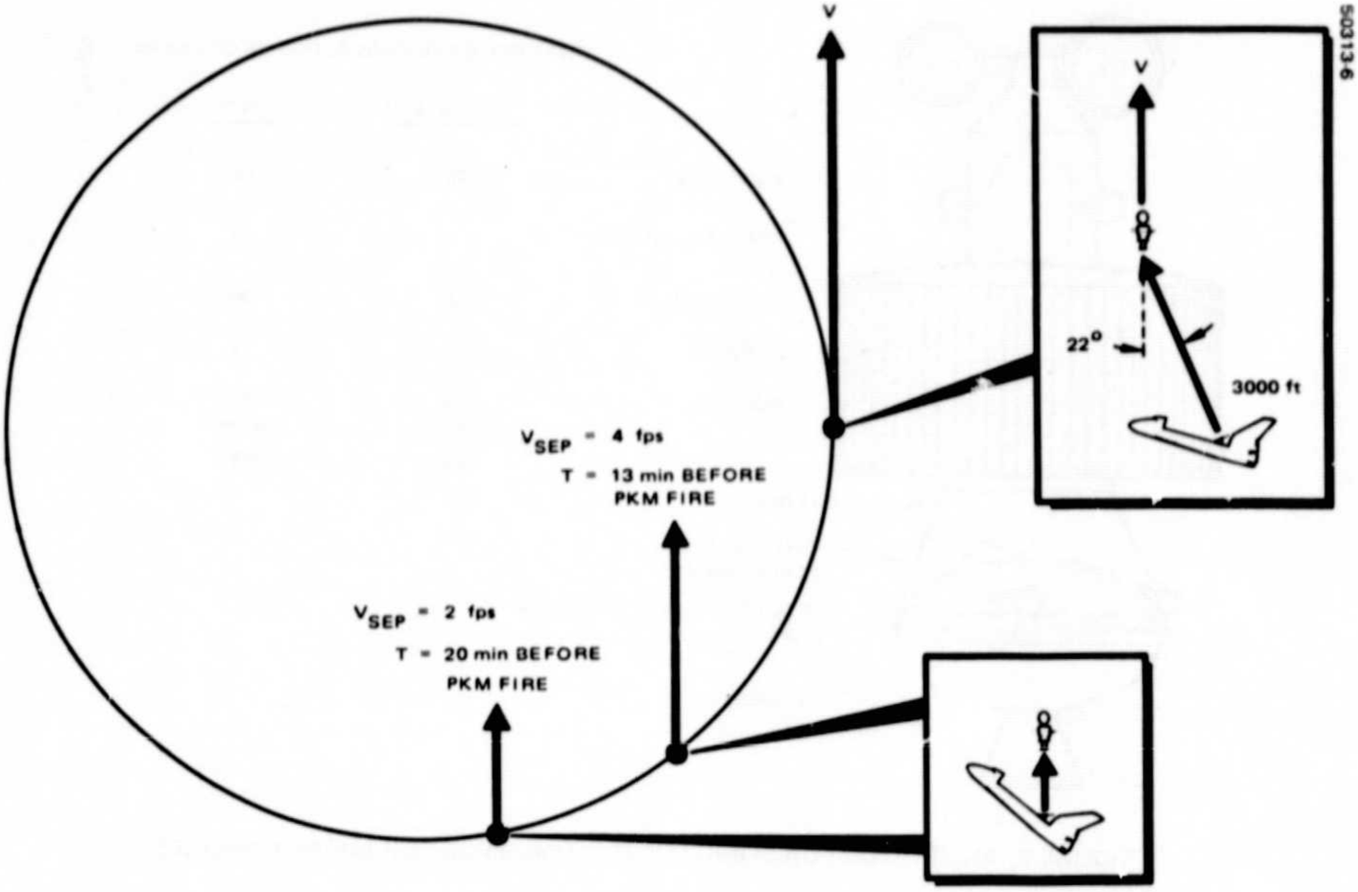
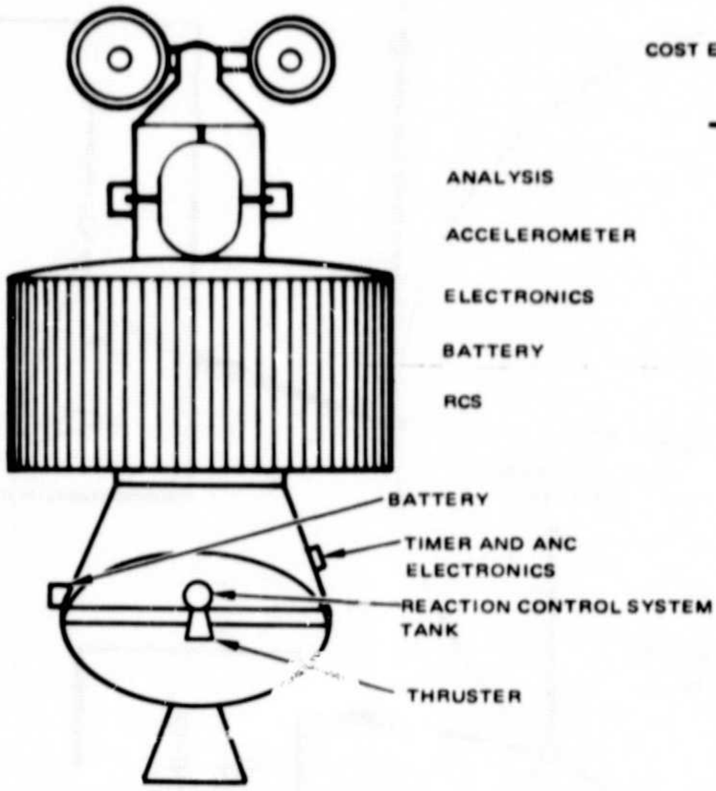


FIGURE 6. PAYLOAD DEPLOYMENT CONCEPT

STABILIZE STAGE AND FIRE PKM WITH TIMER

The spacecraft with the PKM attached is most likely to be unstable about its spin axis. (The inertia about the spin axis would be less than the inertia about its transverse axis.) With the Delta third stage, this instability is tolerated because of the short 40 second flight time before PKM firing, whereas the baseline STS deployment concept previously described requires 13 to 20 minutes flight time. An active nutation control (ANC) system will be required with STS orbiter deployment because of the long period from payload separation until PKM firing.

Active nutation control systems are commonly employed on geostationary spacecraft that are unstable in transfer orbit before AKM firing, and this spacecraft system may be satisfactory for ANC during the PKM phase. An autonomous ANC system using cold gas thrusters and accelerometer sensors can also be mounted on the PKM stage (see Figure 7). The required electronics and battery power supply for the ANC could be combined with the equipment required for PKM firing.



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COST ESTIMATE THOUSANDS OF DOLLARS

	RDT & E	UNIT
ANALYSIS	25	10
ACCELEROMETER	–	3
ELECTRONICS	250	30
BATTERY	–	1
RCS	270	75
	545	119

FIGURE 7. ANC SYSTEM FOR STABILITY AND TIMER FOR PKM FIRING COMBINED

The baseline design selected for motor firing is a simple, highly reliable timer. A radio command link from the orbiter to the payload was evaluated and determined more dangerous because the orbiter crew could only control the time of firing. The orbiter crew would not have a method for determining this function except by a timer. Placing the timer on the PKM stage is a more reliable system than depending on a radio link. The timer would only be activated after the release forces had persisted for a designed period of time after separation from the orbiter tilt table.

A budgetary cost for the ANC and timer hardware is shown in Figure 7 where the timer costs are included in the cost of the electronics. The baseline assumption was to make this system self-contained on the PKM stage; therefore, a battery was also included. The RCS system selected was a cold gas system.

A detailed design of the system is required and further tradeoffs could be performed, e.g., cold gas versus hydrazine RCS system, the addition of a spinup or spindown capability with a minor addition to the RCS system, the safe and arm considerations for the timer, etc.

MATCH PKM SOLID ROCKET MOTOR TO PAYLOAD

Solid rocket motor technology for space-qualified motors was surveyed and the results indicate the currently used motors can only be marginally improved with known advances in technology. Major technology advancement with factors of more than 50 percent improvement in performance, weight, volume, etc. is not anticipated in the next 5 years.

The PKM stage must provide a nominal $\Delta V = 8000$ fps. The size of the payload will determine the size of the perigee motor (see Figure 8). The Delta 2914 launched payloads are nominally 750 (341 kg) pounds in geostationary orbit and the STS can launch these payloads with the Delta third stage motor, the TE-364-4, as 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, (e.g., 400 n.mi. (760 kg)) at the expense of 20,000 pounds (9,091 kg) of additional orbiter fuel. The 750 to 820 pound (341 to 373 kg) payload cases, therefore, do not require a PKM development for the TE-364-4 is adequate.

GEOSTATIONARY ORBIT PAYLOAD POUND	PKM WEIGHT, POUND	SHUTTLE PAYLOAD, POUND	BASIS
750	2500	5000	TE 364-4
750			2914 DELTA
820	2500	5000 (+20000 ORBITER FUEL)	TE 364-4
1000			3914 DELTA
1000	3400	5700	PKM DEVELOPMENT
2100			CENTAUR
2160	8050	13,200	MINUTEMAN III
3200/1600			TITAN IIIC
6150	18,800	32,500	1/2 ORBITER PKM DEVELOPMENT

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FIGURE 8. MATCH PKM SOLID ROCKET MOTOR TO PAYLOAD

The Delta 3914 capability matches a nominal 1000 (455 kg) pound spacecraft in geostationary orbit which exceeds the 400 n. mi. (760 km) orbiter plus the TE-364-4 capability. A new PKM with a weight of 3400 pounds (1545 kg) must be developed to match this requirement.

The Atlas-Centaur has a 2100 pound (955 kg) spacecraft in geostationary orbit launching capability and the existing, space-qualified Minuteman III third stage solid rocket motor can provide this capability.

The Titan IIIC can launch a 3200 pound (1455 kg) payload or two 1600 (727 kg) pound payloads into geostationary orbit. The Minuteman III derived PKM would accommodate the dual Titan payloads if they were launched one at a time. A new PKM could be developed for the 3200 pound (1459 kg) class payloads or a PKM could be developed with one-half STS orbiter payload bay capability. This new PKM, weighing 18,000 pounds (8549 kg), provides a capability for a 6150 pound (2795 kg) synchronous orbit spacecraft.

The development cost of the new PKM solid rocket motors range from $4 million for the 3400 pound (1545 kg) Delta 3914 class to $7 million for the 18,800 pound (8545 kg) one-half orbiter payload class.

ACCOMMODATE PKM/AKM COMBINATIONS

The study focused 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: 1) the type of AKM (liquid versus solid) and 2) its attachment to the spacecraft (integral and nonintegral). (See Figure 9.)

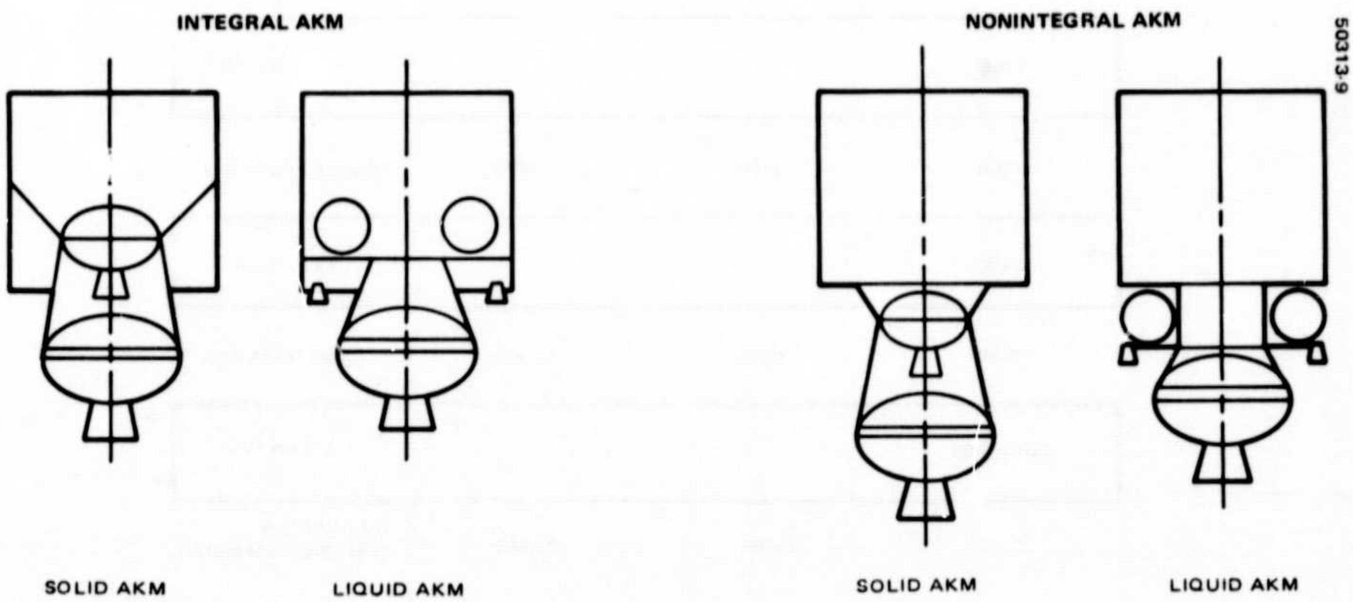


FIGURE 9. PKM/AKM COMBINATIONS

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, for 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, both of which are nonintegral. This design, however, has the feature that the combined PKM/AKM stage could be 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 or larger.

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 with the PKM/AKM offers the payload supplier a variety of options that the current expendable launch vehicles do not have and that would also be constrained by a government-furnished upper stage. A useful service would be for NASA to accommodate a variety of payload propulsion configurations supplied by the payload suppliers and the PKM/AKM concept permits this desired service flexibility.

ACCOMMODATE USER SPACECRAFT

All three-axis or spin stabilized spacecraft designed for launch by Delta or Centaur are accommodated on a spinning PKM stage without spacecraft modifications or cost impact. The list of geostationary three-axis spacecraft launched or to be launched on Delta with a spinning PKM in the 1970 to 1980 period includes Symphonie, CTS, RCA DOMSAT, OTS, JBS, and IUE. The three-axis stabilized FLTSATCOM launched on Centaur must also be spin stabilized in its transfer orbit. This is an important consideration if the STS is to accommodate spacecraft in the 1980 to 1990 period, particularly during transition.

All existing spacecraft would require expensive modifications to be compatible with a three-axis stabilized upper stage. On the other hand, potential cost savings that might be realized for new spacecraft design with a three-axis Tug delivery system were evaluated in the study. A three-axis spacecraft would not require: spinning sun and earth sensors, an RCS capability for spacecraft precession in the transfer orbit or for spacecraft spindown after AKM firing, and a spacecraft spin balanced design or testing. The spinner and three-axis spacecraft would not require stability analysis or ANC in the transfer orbit, and the AKM would not be required. The spinner would require a spinup system, but this was assumed to be a part of the spinner RCS system as it is with Centaur launches. A spinup system must be added to Delta-class spacecraft launched by Tug. The maximum saving possible for three-axis spacecraft is approximately $400,000 per spacecraft where spacecraft cost is a minimum of $10 million. This small savings raises the question whether they would be realized in an actual program particularly when additional costs are required to three-axis stabilize the Tug stage.

The primary and probably most important impact on spacecraft when considering STS launch will be the orbiter payload bay environment during launch. This was not considered in the study and such analysis is required as soon as reasonable test data are available.

ADOPT POLICY OF USER PROVIDED UPPER STAGE

When the user provides the upper stage, he must consider the added cost. The total cost for geostationary payload delivery by the PKM/AKM technique is estimated in terms of existing spacecraft that only require a PKM stage and new spacecraft that require both a PKM and AKM.

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 would only require development of the stage structure and support; hence, an RDT&E cost of approximately $800,000 and a unit cost of $490,000. The Centaur-class spacecraft using the Minuteman III PKM would be approximately the same. If a new solid rocket motor were developed for a Delta 3914 class, the RDT&E costs would be $4.8 million and the unit cost would be $700,000. The estimated RDT&E cost for the largest PKM stage compatible with one-half the STS orbiter bay would be approximately $8 million and the unit cost $1.05 million. (These costs are given as the three columns in Table 1).

TABLE 1. 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 three-axis 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 the cradle and tilt table can be reused many times and its design should consider this reuse philosophy. The charges per flight for 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.

ESTABLISH PAYLOAD CAPTURE PLAN

A comparison of all geostationary spacecraft launched or under construction for launch in the 1970 to 1980 period versus all the geostationary spacecraft in the NASA STS Payload Data and Analysis (SPDA) document for 1979 to 1991 revealed the dominance of reimbursable launches and Delta-class payloads (see Table 2).

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 is important in light of the large number of reimbursable launches expected in the 1979 to 1991 period.

TABLE 2. 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
Grand total			1970 to 1980			1979 to 1981	
	Reimbursable		56			89	
	NASA		5			29	

Forty-two Delta launches are firmly planned in 1970 to 1980, while one Titan and 18 Centaur are planned for the same period neglecting DoD launches. The SPDA has 44 Delta launches in the reimbursable category. i.e., either commercial, other government (NOAA), or foreign. The 18 Centaur launches in 1970 to 1980 are all reimbursable, whereas the SPDA assumes a major increase in NASA development of spacecraft in this class. The data are somewhat skewed in this area and in the Titan category because the SPDA assumes Titan-class spacecraft for Intelsat and COMSTAR type users, hence the 39 reimbursables in the SPDA Titan category. The growth from 18 Centaur-class spacecraft in 1970 to 1980 to 45 (sum of Centaur and Titan categories) is suspect, but the 45 Centaur/Titan-class versus 44 for Delta indicates at least 50 percent will be Delta-class in accordance with NASA planning.

The reimbursable versus NASA payloads in the 1970 to 1980 period (56 versus 5) drops in the NASA planning data to 89 versus 29 (a factor of 11 to a factor of 3). The large number of reimbursable launches in either case means the STS must provide a useful service at a competitive cost.

ESTABLISH COMPETITIVE COST WITH MULTIPLE PAYLOADS

Projections of Delta and Centaur cost for the 1980 to 1990 period are plotted in Figure 10 with an assumed inflation rate of 5 percent per year. The projected reimbursable costs are based on

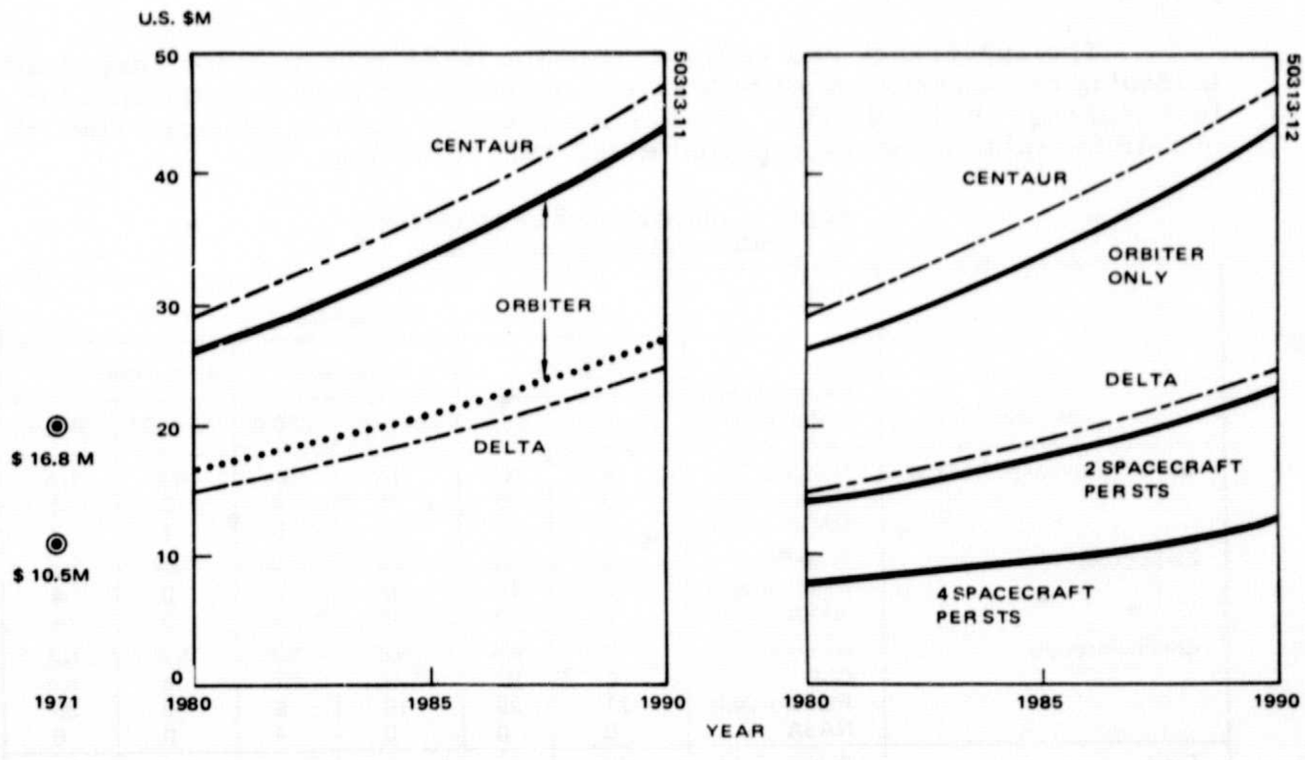


FIGURE 10. PROJECTED REIMBURSABLE COST (5% INFLATION, 60% OF DIRECT COST FOR STS ORBITER INDIRECT COST, UPPER STAGE $1 MILLION TOTAL PER SPACECRAFT)

1) Delta, $12.9 million for a 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's use of expendable launch vehicles. An additional factor must be added to arrive at an equivalent reimbursable cost and the expendable vehicles ranged from 60 to 100 percent of the direct cost. A 60 percent of direct cost factor was assumed for the orbiter flight cost (i.e., $18.6 million in 1971 dollars).

The plot of the orbiter cost per flight, assuming 5 percent inflation and the 60 percent factor, shows the orbiter cost is nearly twice the cost of Delta and approaches the cost of Centaur. If this proves true, the Delta costs could double and still be lower than the STS costs (orbiter plus upper stage), or the Delta users could transfer to a competitor.

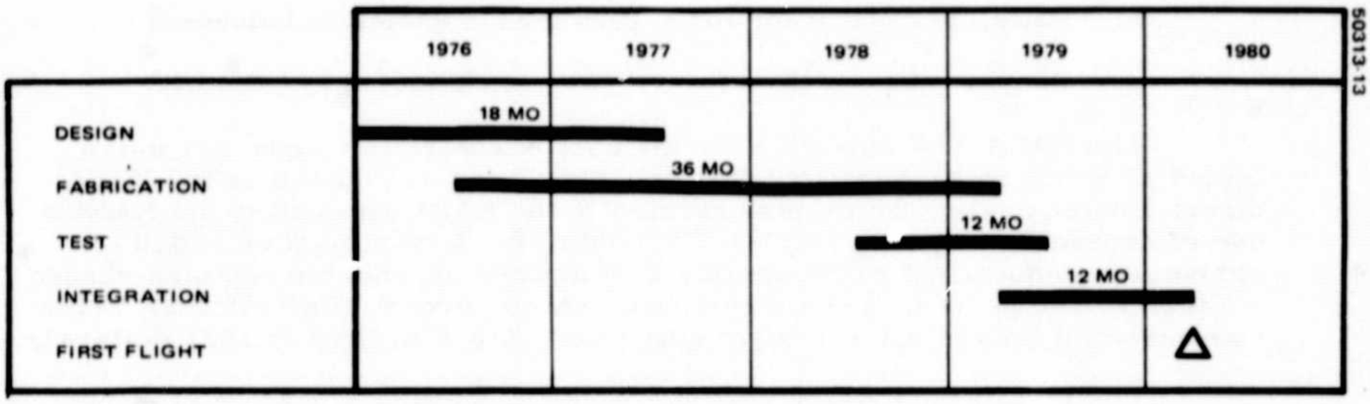
The STS with its large payload capability offers the opportunity for multiple payload launches on a single orbiter flight. Both two-spacecraft and four-spacecraft multiple launch cases are plotted for comparison with the expendable vehicles and the STS orbiter-only case. The multiple launch cases include $1 million cost for the upper stage, which is based on the previously presented data as an upper bound for the PKM stage cost.

The 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 for the STS orbiter cost, and NASA's provision for accommodating multiple users simultaneously.

DEVELOP INITIAL CAPABILITY

A program plan for the PKM/AKM concept, as shown in Figure 11, 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 the Department of Defense 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 Corporation and McDonnell Douglas Corporation are doing now with the Delta 3914, it is highly probable users will develop special PKM/AKM stages matched to their specific needs.



TASKS

ORBITER — PAYLOAD SUPPORT CRADLE
PAYLOAD DEPLOYMENT MECHANISM

PAYLOAD — PKM
STAGE SUPPORT
STAGE STRUCTURE

FIGURE 11. PROGRAM PLAN

The tasks required are design and fabrication of the payload support cradle and payload deployment mechanism for the orbiter and the design and fabrication of the PKM stage including the PKM solid rocket motor, the stage support, and the stage 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.

CAPTURE PAYLOADS WITH COMPETITIVE SERVICE AND COST (Figure 12)

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

1) Lowest nonrecurring cost of any upper stage program

2) Recurring cost totally paid by the user

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

4) Least impact on the STS 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 orbiter hardware, payload stage hardware, and establish a capture plan.

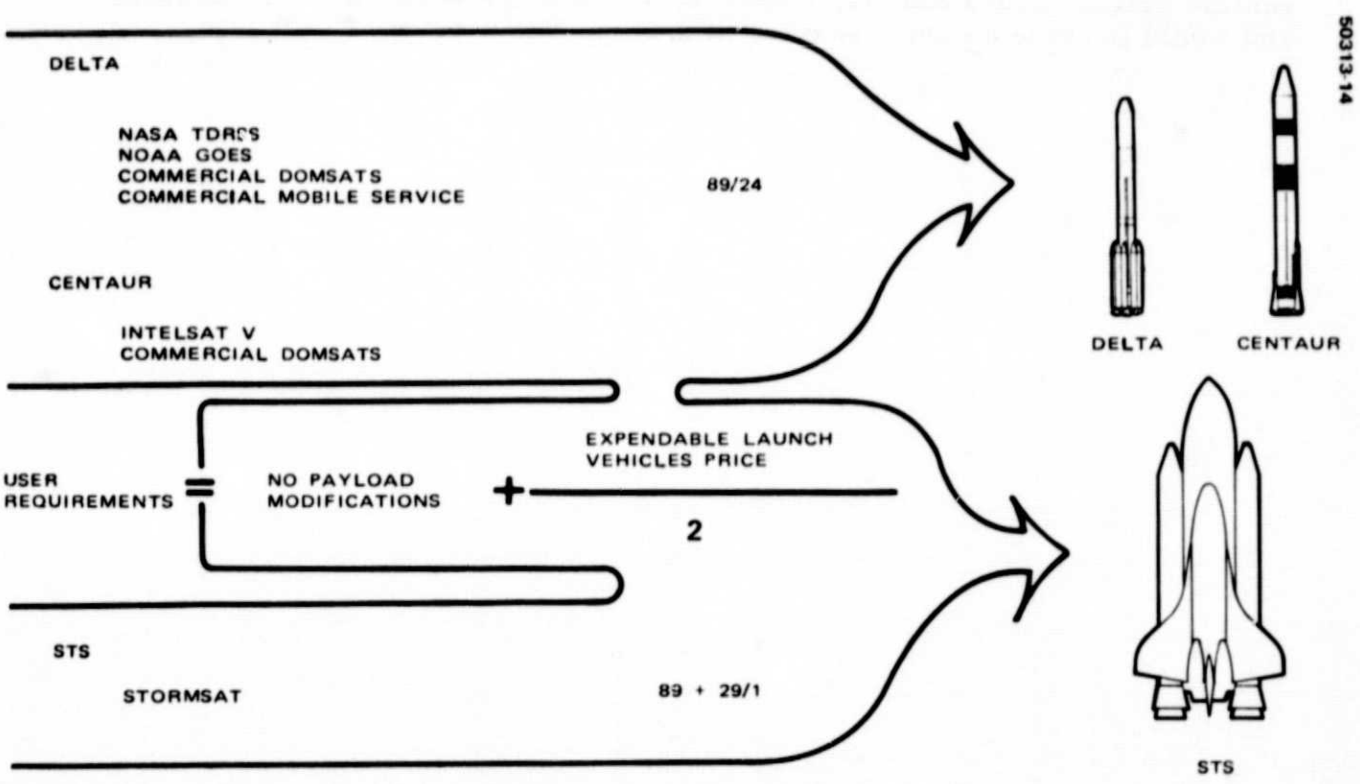


FIGURE 12. PAYLOAD CAPTURE REQUIREMENTS

A proposed capture plan would be to determine which payloads in production 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, COMSTAR, etc., which will need replenishment; the commercial mobile systems such as Aerosat, MARISAT, etc.; and the Intelsat V. 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 the 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.

There are 89 NASA SPDA payloads planned for reimbursable launches, 24 of which are already in the procurement process and designed for either Delta or Centaur launch. Total geostationary payloads in the NASA SPDA are the 89 reimbursable plus 29 NASA; only one, the STORMSAT, is being defined for STS launch. The reimbursable user requirements in order to switch for the expendable vehicles to the STS would be "no spacecraft modifications" and a significantly lower cost, i.e., one-half the expendable launch vehicle price. The PKM/AKM concept would make such an offer possible and would provide a useful service at a competitive price.