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Acronyms

ACCESS	assembly concept for construction of erectable space structure
APS/MT	astronaut positioning system/mobile transporter
CELV	complementary expendable launch vehicle
ESGP	Earth sciences geostationary platform
EVA	extravehicular activity
FF	free flyer
GCTI	global change technology initiative
GEO	geostationary Earth orbit
GHRMR	geostationary high-resolution microwave radiometer
GPS	global positioning system
ID	identification
LaRC	Langley Research Center
LCP	launch configured package
LEO	low Earth orbit
MPESS	mission peculiar equipment support structure
MSC	mobile servicing center
OFA	obstruction free area
OMV	orbital maneuvering vehicle
OTV	orbital transfer vehicle
PSR	precision segmented reflector
S/C	spacecraft
SSF	Space Station *Freedom*
STS	Space Transportation System
STV	space transportation vehicle

Abstract

Large spacecraft, particularly in geostationary Earth orbit (GEO), require special attention to the design challenges of launch vehicle packaging, deployment, and/or on-orbit assembly. Design studies of two different GEO spacecraft required that packaging, deployment, and on-orbit assembly analyses be conducted to establish the viability of these concepts for future NASA missions. This study used these analyses as "strawman" concepts for an investigation of packaging, deployment, and on-orbit assembly techniques. It also revealed generic guidelines for in-space assembly and highlighted the importance of early integration of packaging, deployment, and on-orbit assembly requirements into the spacecraft design. The first study spacecraft was used to study the definition and analyses of on-orbit assembly options for large GEO spacecraft. The second study spacecraft required investigation of the feasibility of deploying large spacecraft at GEO. The second spacecraft was also used to examine the packaging requirements of a deployable spacecraft and the packaging requirements for a hybrid assembled and deployable version of that spacecraft. This investigation was done with attention to minimum volume (and minimum launches) and to the relationship between packaging and spacecraft deployment and final configuration.

1. Introduction

The Langley Research Center (LaRC) is conducting engineering studies to identify and evaluate the role for advanced technologies in proposed future space systems.

The future need for large structures in space, particularly at geostationary orbit, has frequently been highlighted and examined in the literature. (See ref. 1. Other work has been done under contract to NASA by Ford Aerospace Corporation, GE Astro-Space Division, and Lockheed Missiles & Space Company.) Studies concerning in-space construction range from the construction of spacecraft bound for the Moon and Mars (work done under contract to NASA by McDonnell Douglas Space Systems Company) to partial telerobotic or autonomous assembly of the Space Station *Freedom* (SSF). (See ref. 2.) LaRC has, as part of its overall spacecraft technology studies, performed packaging, deployment, and on-orbit assembly studies relating to two large geostationary platform concepts as an adjunct to other studies in progress.

The first concept used an LaRC design of a large second-generation Earth sciences geostationary platform (ESGP) (refs. 3 through 6) as shown in figure 1. The ESGP spacecraft designed by LaRC was based on an earlier concept developed by Ford Aerospace Corporation. Configuration details, subsystem definitions, and large antenna designs necessary to conduct in-depth performance analyses were developed for the ESGP spacecraft. Assembly techniques and options necessary for low Earth orbit (LEO) assembly were studied. Packaging of this spacecraft for delivery to LEO is discussed in section 3.2 and in reference 7.

The second concept stemmed from the Global Change Technology Initiative Architecture Trade Study recently conducted at LaRC. This trade study defined an infrastructure of both LEO and geostationary orbit (GEO) spacecraft. The geostationary Earth science spacecraft from this study (fig. 2) is referred to as the "geostationary high-resolution microwave radiometer" (GHRMR) spacecraft after its principal instrument.

The study described in this paper serves to supplement the previous technology studies and provide a mechanism for examining near-term (defined here as prior to 2010) packaging, deployment, and on-orbit assembly options for this class of large geostationary spacecraft. Specifically, the ESGP spacecraft was used in this study to examine requirements for LEO assembly of large GEO spacecraft, and the GHRMR spacecraft was included to investigate complex packaging and subsequent deployment/assembly activities. This paper addresses the two concepts separately and then presents some overall concluding remarks.

Figure 1. Earth sciences geostationary platform.

2. Design Processes

As with most engineering design processes, early incorporation of significant requirements enhances the probability of overall success. Similarly, a design for spacecraft requiring on-orbit assembly must incorporate special requirements at the beginning of the design process. In order that the spacecraft fulfill its on-orbit mission, the designer must carefully attend to the requirements of in-space assembly as they pertain to ease and reliability of construction, time and cost of assembly operations, and the safety of the spacecraft and its assemblers.

GHRMR option	GHRMR-a	GHRMR-b
GHRMR mass, kg	2417	1947
Other bay-loaded mass, kg	732	732
Other S/C mass, kg	3010	2754
Total mass, kg	6159	5433

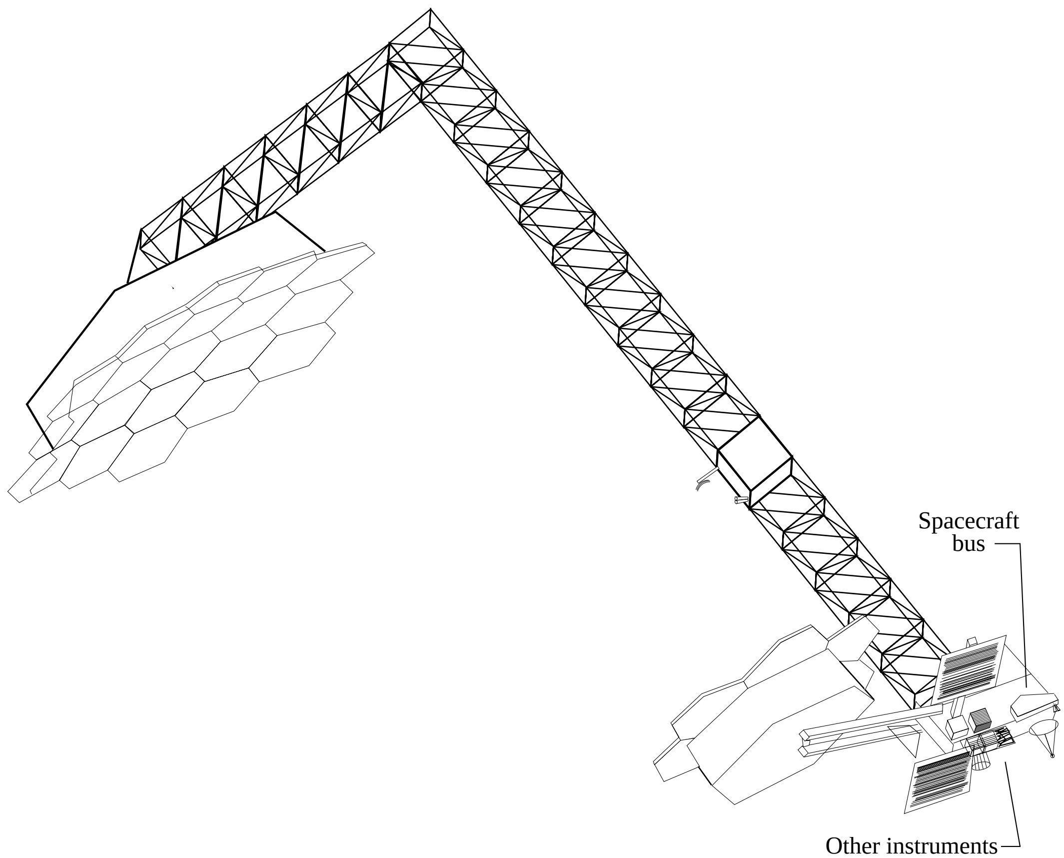


Figure 2. GHRMR spacecraft concept.

A process for synthesizing the design of a large geostationary spacecraft that requires in-space assembly is postulated and shown in figure 3. The normal or usual design process is indicated by the solid lines. A new group of design flows which are necessary for on-orbit spacecraft assembly are indicated by the dashed lines.

The normal inputs to a spacecraft design are shown as mission requirements, instrument requirements, spacecraft subsystem hardware needs, checkout requirements, and launch vehicle constraints. Additional items to be considered for on-orbit assembly include new techniques and hardware specifically for in-space construction, such as EVA techniques, assembly-specific jigs and tooling, and unique deployment schemes. For this new group of design flows, the ESGP and the GHRMR spacecraft provide "strawman" vehicles with a wide-ranging technology representation, which provide an elementary understanding of considerations pertinent to on-orbit assembly/deployment requirements.

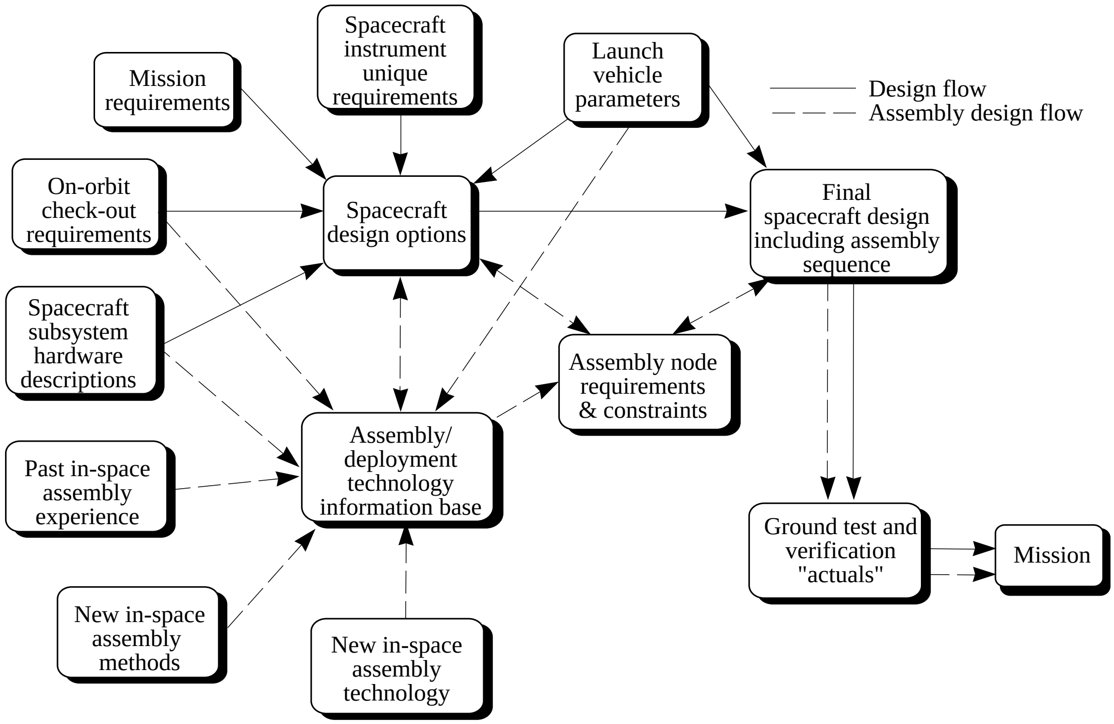


Figure 3. Spacecraft design flows and spacecraft assembly design flows.

3. ESGP Spacecraft

The ESGP spacecraft as shown in figure 4, with part identification (ID) numbers, is a large Earth science geostationary platform concept. It may be delivered to low Earth orbit by either the Space Transportation System (STS) orbiter or the Titan 4 complementary expendable launch vehicle (CELV). (See ref. 7.) After assembly in LEO, it is transferred to GEO by a space transportation vehicle (STV).

3.1. ESGP Spacecraft Description

The overall dimensions of the ESGP spacecraft are approximately 40 by 40 by 10 m. It uses a 3-m box truss structure (ID 1, fig. 4), which acts as a strongback for all spacecraft subsystems and for the payload complement of 18 instruments including two large microwave radiometer antennas (15- and 7.5-m-diameter solid reflectors). This erectable truss provides a rigid and stable base to which the various components of the spacecraft are connected. The individual truss element design (unpublished data from Ford Aerospace Corporation) is derived from a quick-connect type used on the ACCESS experiment conducted on STS mission 61-B. Truss elements are composed of graphite-epoxy tubes of approximately 5 cm diameter.

Attached to the second full bay (from left to right) of the truss is a stack consisting of the payload module, the housekeeping module, and the orbital transfer vehicle (OTV) thrust interface tube (ID 2). At the top of the stack, the payload module contains additional science instruments. Connecting the payload module to the truss is the housekeeping module which contains the major subsystems of the spacecraft such as power and attitude control. Both modules are of aluminum honeycomb construction. The OTV interface tube attaches to the bottom of the housekeeping module and extends through the second bay of the truss to a standard OTV docking ring.

Figure 4. ESGP spacecraft.

The 15-m-diameter microwave radiometer reflector (ID 3) and the 7.5-m-diameter microwave radiometer reflector (ID 4) are the largest instruments on the platform and are the primary platform configuration drivers. Each uses offset-fed Cassegrain-type geometry, which makes use of folded optics to enhance the scanning performance of the radiometer. In this configuration, the large primary reflector focuses the radiation upon the smaller subreflector which, in turn, focuses the radiation upon the feed array. The scanning of the radiometers is accomplished by pivoting their subreflectors through the use of electromechanical actuators. The subreflectors are mounted at the top of the subreflector masts (ID's 5 and 6). These masts are deployable and are of a Minimast design. (See ref. 8.) This type of boom requires the use of a deployment canister mechanism which extends the triangular-cross-section boom two bays at a time. The 7.5-m antenna is similar in concept to the precision segmented reflector (PSR) being developed at the Jet Propulsion Laboratory. (See ref. 9.) It consists of 12 precision surface hexagonal-shaped reflector segments mounted atop a strongback truss structure. The 15-m antenna strongback is similar in concept to a General Dynamics deployable concept (ref. 10) and includes a deployable reflector membrane. The undeployed 15-m antenna is represented in figure 4 by the cylinder (ID 7) at the left end of the spacecraft.

An adjustable solar sail (ID 8) is attached to the right end of the box truss structure to reduce spacecraft control torque requirements.

The foldable solar arrays (ID 9) are attached to the housekeeping module by deployable booms (ID 10) which place them at sufficient distance from the spacecraft so that they are not shadowed by the 15-m radiometer reflector. These deployable booms are 0.5 m in diameter with flexible longerons and are deployed from the housekeeping module by electric motors. Each solar panel boom rotates to maintain solar pointing.

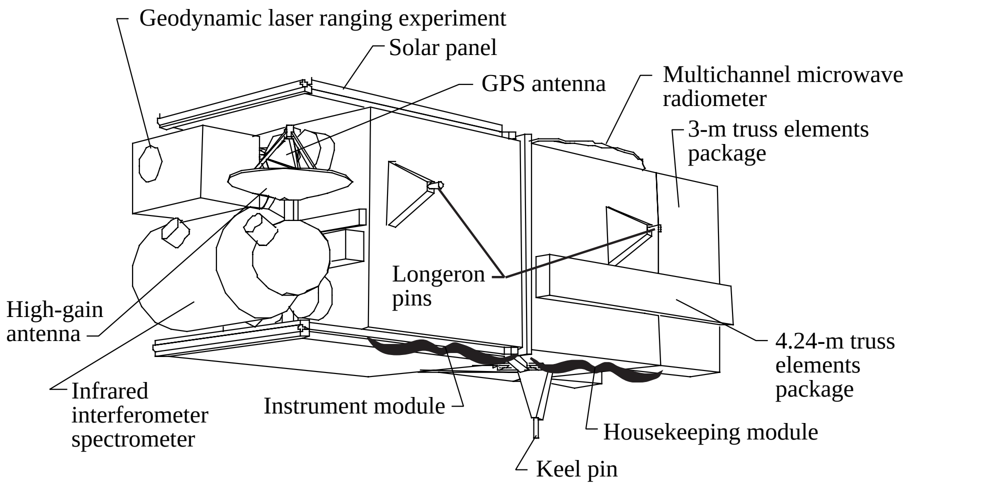
Attached to the third, fourth, fifth, and sixth bays of the box truss, between the housekeeping module and the 7.5-m reflector, are other elements of the scientific instrumentation complement (ID 11) of the spacecraft.

3.2. ESGP Assembly Sequence Incorporation

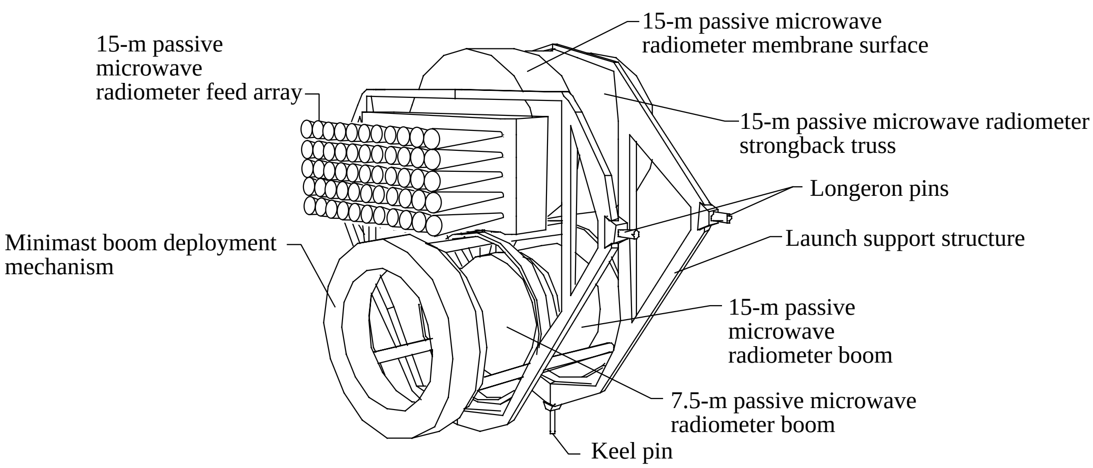
A number of factors influenced the incorporation of the on-orbit assembly sequence into the ESGP spacecraft design. First, other studies relating to this spacecraft were reviewed to acquire spacecraft functional understanding (i.e., the design of the mission, instruments, checkout requirements, launch vehicle constraints, subsystem hardware design). This, in effect, was a review of the spacecraft design process, as shown in figure 3 by the solid lines.

The launch packaging study (ref. 7) identified three viable scenarios for delivery of the ESGP spacecraft to Space Station *Freedom* (SSF) in LEO. Two scenarios use the STS orbiter while the third utilizes the Titan 4 complementary expendable launch vehicle. In the shuttle scenarios, the ESGP components are packaged in three specialized pallets as shown in figure 5. A similar arrangement of three modified payload modules (fig. 6) is integrated into the Titan 4 launch vehicle in a No Upper Stage configuration. The launch configured payload pallets or modules are henceforth referred to as "launch configured packages" (LCP's). Mass and center-of-gravity launch constraints for the three configurations were verified as being met. The Titan 4 scenario and the shuttle single launch scenario both require orbital maneuvering vehicle (OMV) retrieval to SSF for assembly, whereas the shuttle two-flight scenario is launched directly to SSF. This paper uses these launch vehicle assumptions and the launch packaging findings of the launch packaging study (ref. 7) as a starting point for assembly definition.

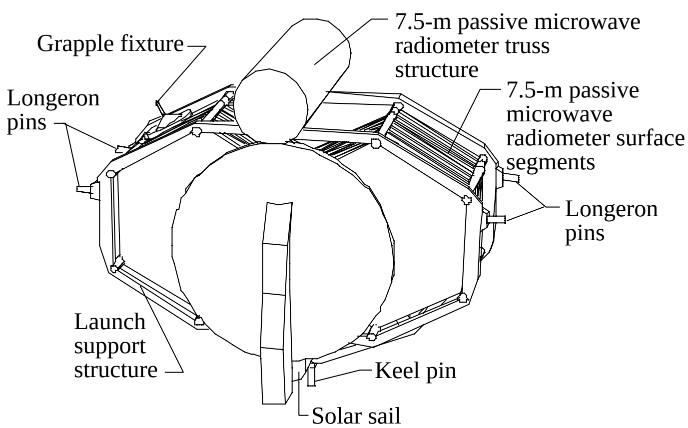
The design flows represented by the dashed lines in figure 3 were next undertaken to define a strawman example of an assembly sequence. An investigation of previous in-space assembly experience (ref. 11), new in-space assembly methods (ref. 12), and new in-space assembly technology (ref. 13), along with previously gathered spacecraft hardware and design data, became the basis for an assembly/deployment technology information base. This information base, along with the spacecraft design options chosen, allowed the definition of an assembly node (an assembly place and process) for the ESGP spacecraft. The specific design form of the assembly node can have a number of acceptable solutions and is normally based on the engineering criteria of simplicity, safety, cost, and so forth, and their respective priorities.



(a) Pallet 1.



(b) Pallet 2.



(c) Pallet 3.

Figure 5. ESGP cargo pallets for STS orbiter.

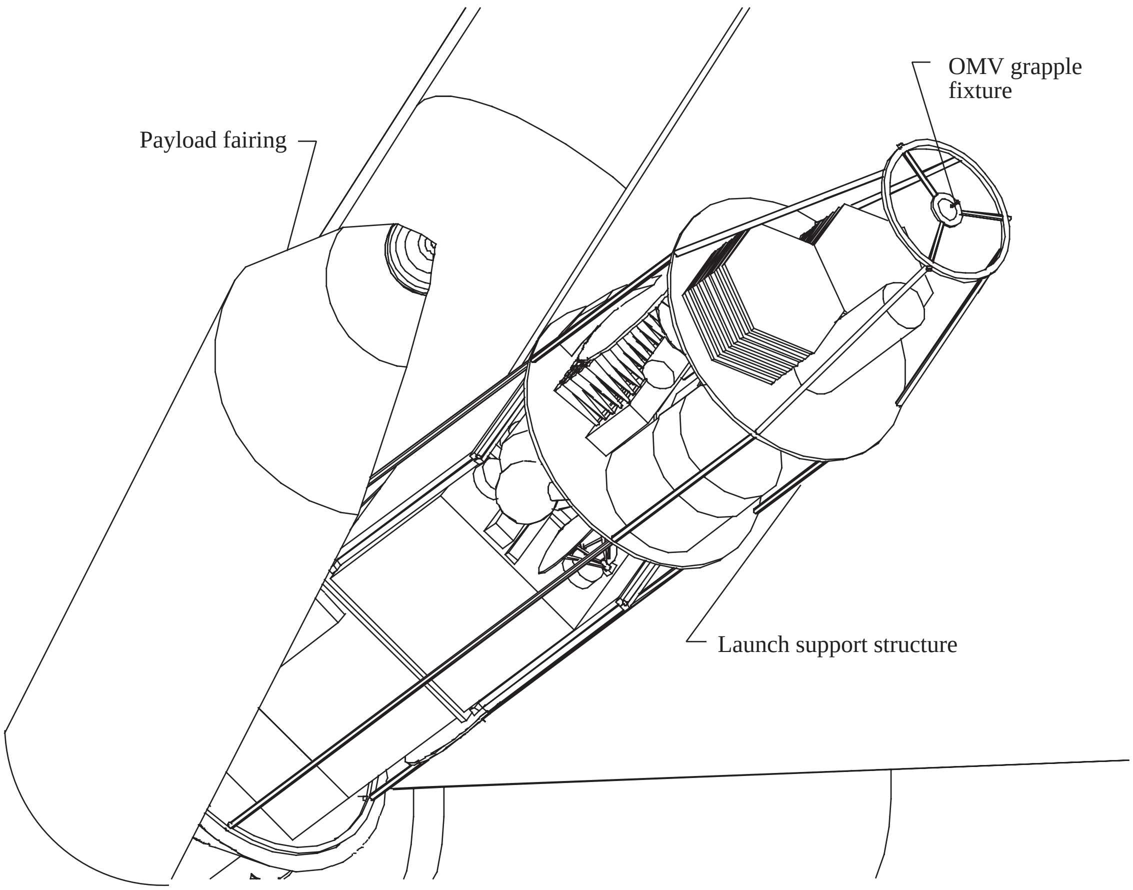


Figure 6. Titan 4 launch packaging for ESGP.

3.3. ESGP Assembly Sequence Definition

Assembly sequence design decisions are strongly influenced, if not dictated, by the specific spacecraft design under consideration. In any event, certain issues must be addressed (with mutually compatible conclusions) prior to determining an acceptable assembly sequence. These are as follows:

> Where is the assembly to occur? The options considered were the Space Transportation System orbiter, a free flyer (FF) arrangement where the spacecraft is its own assembly node, and the SSF. Although assembly at the STS does not depend on other elements such as the SSF or an FF, it does have limited on-orbit mission duration, leaving little, if any, contingency assembly time. The FF requires EVA support and/or astronaut transfer from either the SSF or the STS and active spacecraft control during at least part of the assembly period to point the large reflectors away from the Sun. The analysis assumes that the SSF has obstruction free area (OFA) available for both the assembly and the storage of the spacecraft in its launch-configured packages. EVA support is assumed to be available from either the orbiter or the SSF, and there is no limited-time constraint (before the shuttle's return to Earth) on the allowable time to build. Other factors, not quantifiable in a strawman scenario, such as disturbance of microgravity science experiments, should also be considered. Based on these considerations, the SSF was chosen as the assembly node.

Where and how large is the OFA? Sufficient OFA for assembly of the ESGP spacecraft must be found and allocated for both the assembly process and the storage of LCP's (until required for assembly). Although availability may change as the SSF design evolves, sufficient OFA for assembly was found along the back of the lower boom in the enhanced operations capability configuration of the evolutionary SSF (unpublished data from the Space Station *Freedom* Office at LaRC). (See fig. 7.) This configuration is scheduled to be within the time frame used for this study (before 2010). OFA used in assembly must also allow room for any special tooling required. OFA for storage of the spacecraft LCP's should be in near proximity to the assembly OFA and could be on either the port or the starboard keel near the lower boom. Any environmental protection needed for the LCP's must also be included. For this study, it is included in the launch packaging. Alternative solutions to the OFA include (1) along the inside of the starboard lower keel (fig. 7), (2) a hanger structure built on the evolutionary SSF dual beam truss structure (particularly if a protective enclosure is needed), (3) a temporary shelf-type truss structure (fig. 8) specifically designed for this spacecraft assembly, and (4) a special construction station capable of efficient assembly with the spacecraft oriented orthogonally to the SSF truss structure (fig. 9).

What will be the assembly sequence? Several options to this major question are (1) build the spacecraft truss structure first and then add the other components, (2) build the three-piece module stack first and expand from there, or (3) build the spacecraft in an end-to-end sequence with major deployments last. Although the first and second options may also provide acceptable assembly sequences, the third option was chosen to complement the special tooling discussed later and as a method of minimizing the EVA translational movements and resulting astronaut physical exhaustion as recommended in reference 14.

Figure 7. OFA locations as suggested for Space Station *Freedom* enhanced operations capability phase.

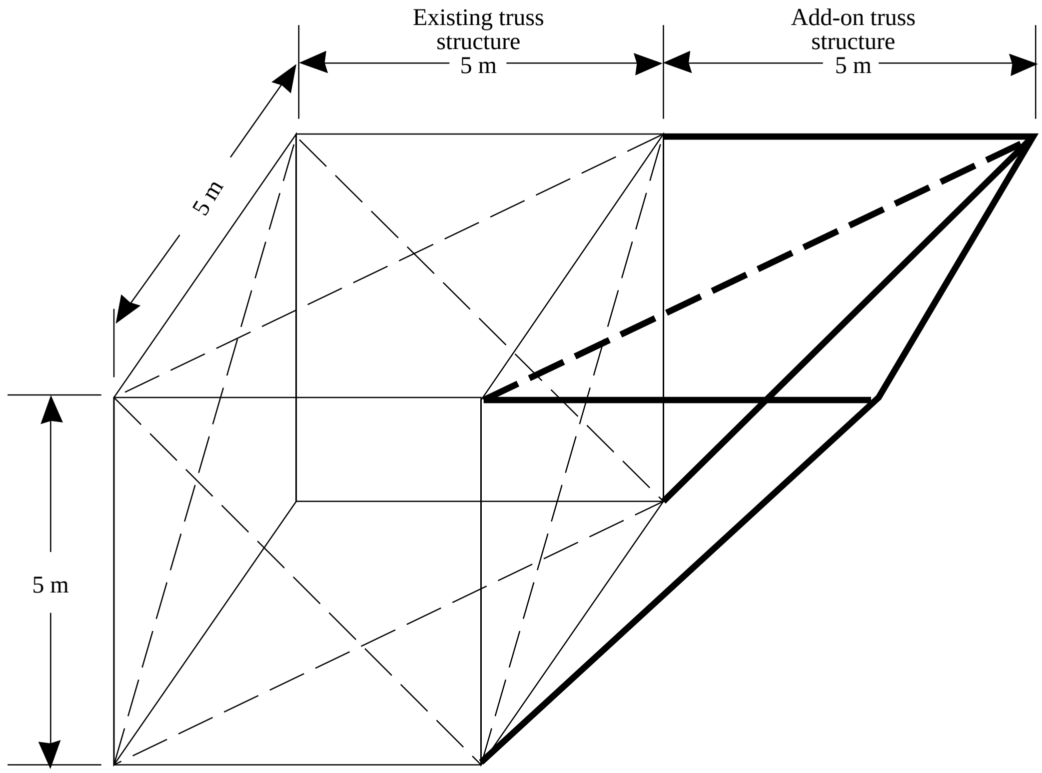


Figure 8. Space Station *Freedom* box truss section with proposed add-on shelf structure.

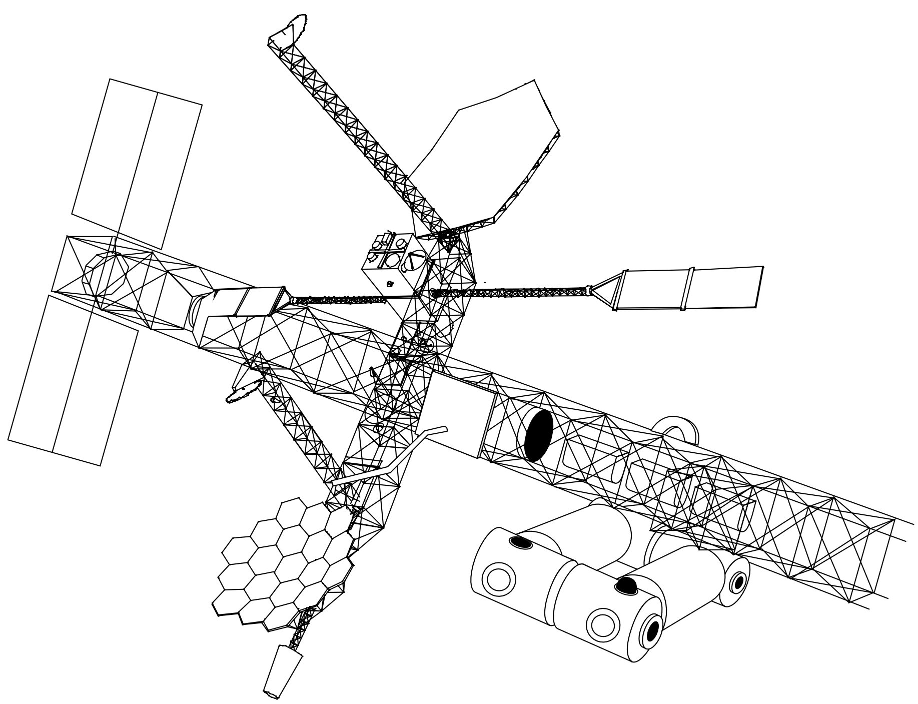


Figure 9. Orthogonal construction station at SSF.

How is the assembly implemented? The choices are EVA, telerobotics, automation, or combinations thereof. EVA requires much of a limited resource and provides significantly less astronaut safety than the other methods. Telerobotics may require as much or more assembly time than EVA but provides the desired safety.

Telerobotics is applicable to many but not all assembly tasks. Robotics may be faster than either EVA or telerobotics and also maintains astronaut safety but is usually best applied to repetitive tasks. In addition, its application to this type of assembly task is not a near-term option. (See ref. 13.) Therefore, for the near-term assembly techniques that were set up as ground rules for this study, a combination of EVA and telerobotics was chosen as a baseline assembly method. The assembly process, in whole or in part, could be automated as an enhancement when the necessary automation technology becomes available.

What assembly tooling is required? The unpacking and temporary stowage of the LCP items would require the use of a translatable remote manipulator arm device such as the SSF mobile servicing center (MSC). These one-time tasks would appropriately be done telerobotically to avoid EVA. For spacecraft truss assembly and instrument/appendage attachment via EVA, a build-extend turntable-type structure was devised. This special tooling was designed with similarity to ACCESS (ref. 11) and to a proposed SSF truss structure assembly tooling (ref. 12) to minimize astronaut EVA physical exertion. The build-extend descriptor refers to a truss section or bay being built on the tooling fixture, then extended outward but still held as the next bay is assembled. As each bay is being assembled, the turntable is partially rotated or rocked to give the astronauts access to all assembly nodes. This process is repeated until the complete truss is assembled. The same basic tooling apparatus that has been used to build the spacecraft truss can then be reconfigured to support the assembly of the 7.5-m solid reflector and its truss strongback. Another possible option, although not considered here because of unknown availability, would be to use the reconfigured SSF astronaut positioning system/mobile transporter (APS/MT) proposed for SSF assembly. (This system was discussed by D. R. Barron of McDonnell Douglas Mechanical Systems at a planning review meeting at Johnson Space Center.) This tooling configuration would have been used in SSF truss assembly and possibly could be downsized to accommodate the smaller truss of the ESGP spacecraft. A drawing of the assembled ESGP spacecraft and its tooling is shown in figure 10.

Restating from the above discussion, the strawman assembly sequence was defined within the following guidelines:

1. The assembly is to occur at SSF

2. The assembly is to use OFA along the back of the lower boom

3. The spacecraft is to be built in an end-to-end order

4. The minimum technology approach (EVA, telerobotics) was chosen as a baseline instead of an advanced technology approach (automation)

5. The basic tooling required was defined as a build-extend fixture incorporating a turntable

Once the basic approach has been selected, a more detailed review of the assembly scenario should be conducted. Examples of this more comprehensive review of the assembly design are now discussed.

This review should investigate the need for identifying other resources and their impact on the assembly node. This would include electrical power, propellant handling scenarios, other fluids usage, special checkout requirements, and any special instrumentation required for either assembly or checkout. The appropriate selection of either EVA or (tele) robotics for the implementation of particular assembly tasks must be considered. For instance, some tasks may be significantly quicker to implement via EVA although the risks associated with other tasks may be too high for EVA. Use and scheduling of other SSF resources such as the MSC, the SSF crew, if available, or an auxiliary crew must be taken into account. The requirements for environmental protection of the LCP while stored at SSF must be addressed.

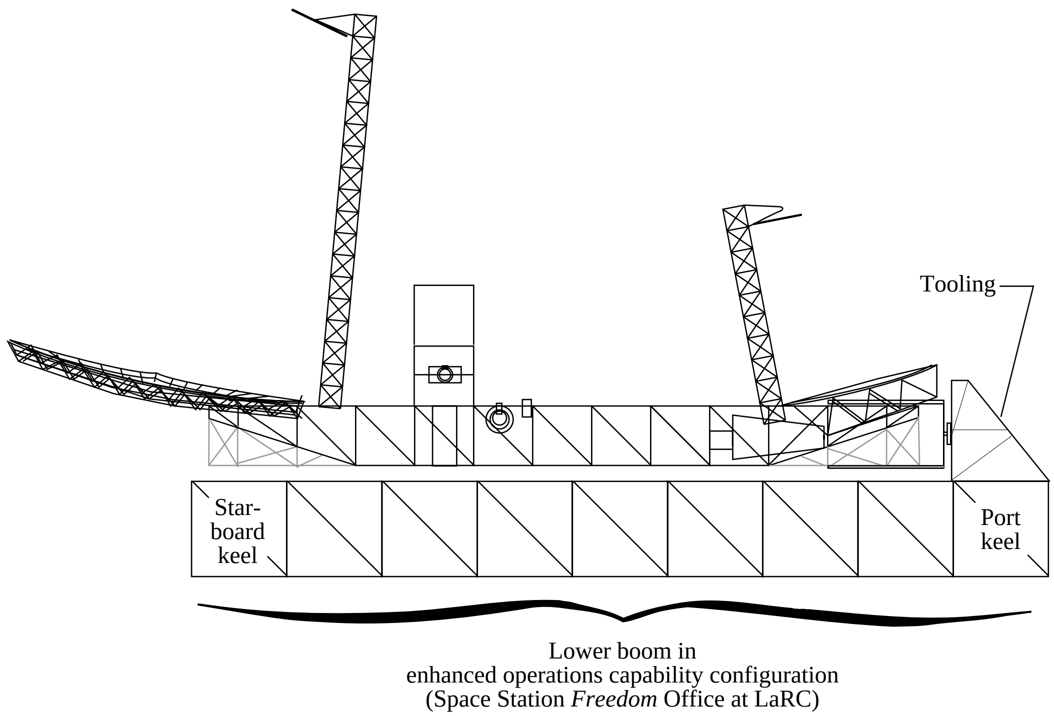


Figure 10. ESGP spacecraft assembled configuration (with tooling).

Furthermore, the minimization of assembly sequence times (or other selected parameters) can be significantly enhanced by attention to the following generic design considerations:

1. Design of the vehicle/node interface with possible incorporation of assembly requirements into the node design

2. Design for minimum spacecraft assembly (maximum prebuild on Earth)

3. Design for quick assembly (e.g., snap-lock fittings versus nuts and bolts)

4. Incorporation of self-alignment features

5. Incorporation of self-verifying locking features

6. The innovative design and use of assembly aids

7. Examination of the assembly sequence to minimize the need for time and other resource-intensive operations such as MSC translation and plane changes

8. The use of hybrid deployable/erectable designs that can yield significant resource savings or may give other unrelated benefits

9. The generation and use of a lessons-learned compilation to avoid previous mistakes (a study made by Rick Vargo, Fred Mitchell, Ken Flemming, and Maurice Willis of McDonnell Douglas Space Systems Company for Kennedy Space Center)

Each spacecraft element must be kept compatible with the design of the assembly sequence. As an illustration of this necessity, an incompatibility was found with the proposed design of the spacecraft propulsion system. The hardware that was selected in the spacecraft design (not the assembly design) was an all-titanium, all-welded design for both propellant storage tanks and propellant transfer lines. Thrusters were placed on the instrument module and its supporting truss bay. To minimize the potential complexity of in-space operations (the assembly and welding of this hardware), the option of prebuilding the module stack and its truss bay as a single assembly before launch was selected. Therefore, for a small Earth to LEO transportation volume penalty (i.e., the truss bay launched assembled vs launched as struts and nodes), the need for complex in-space assembly and welding of the propulsion system plumbing was avoided. This type of potential incompatibility illustrates the desirability of integrating the assembly sequence, as early as possible, into the total spacecraft design process.

A time-sequence depiction of the construction sequence is shown in figure 11. The assembly begins with the +x end of the ESGP spacecraft (left end in fig. 4) and grows toward the opposite end. In all drawings, the build-extend tooling would be at the right end of the so-far-assembled spacecraft. In general, attachments are secured incrementally as construction progresses. The payload and housekeeping modules, thrust tube, and associated truss bay are constructed on Earth before launch and assembled into the spacecraft as a single entity. The 7.5-m reflector is constructed after all truss bays are completed. This requires reconfiguration of the tooling turntable. After construction, the 7.5-m reflector is attached to the spacecraft. Next, using the Minimast deployment mechanism, both Minimast subreflector booms are deployed and attached to the spacecraft with the aid of the MSC. The 15-m reflector canister (ID 7 in fig. 4), attached to the left end of the spacecraft, is then deployed. Individual checkouts occur as appropriate during spacecraft construction with full integrated checkout occurring last. A rough order-of-magnitude estimate for assembly time was desired. Based on previous experimental in-space construction timeline studies (refs. 12 and 14), a preliminary estimate for assembly of the ESGP spacecraft, using the specified baseline technology (EVA, telerobotics), is at least three and probably four standard 6- to 8-hour EVA periods. Contingency assembly time was not estimated for this study. Integrated checkout, being spacecraft and payload dependent, was not included in this estimate.

3.4. ESGP Summary Remarks

A spacecraft design process to accommodate on-orbit assembly design for an Earth Sciences Geostationary Platform was postulated. With this process, an information base was collected utilizing relatively near-term technology parameters. The information base was next applied in the specific definition of an assembly sequence for an already defined strawman vehicle (the ESGP spacecraft). The use of a strawman test case for an on-orbit assembly-required spacecraft design helped to identify and highlight the significant steps needed for this design process.

We recommend that the assembly tasks be planned as an integral part of the original design process and considered second in importance only to the spacecraft functional requirements. When this is not possible, they must be integrated as soon as feasible within the overall design process. The on-orbit assembly tasks must be verified as compatible with the spacecraft design and with the Earth to LEO packaging. This is done not only to assure compatibility between spacecraft design and assembly requirements but to increase the probability of optimal engineering design, cost and time savings, and safety. In addition, to assure compatibility between spacecraft design and assembly design, certain major issues should always be addressed. They are as follows:

1. Where is the assembly node?
2. How much assembly space is needed?
3. What is the basic build sequence?
4. What are the crew/EVA requirements?
5. Are there telerobotics/automation requirements?
6. What assembly tooling may be required?
7. How much overall time is required for the assembly?

A review of the assembly design followed an assessment of the major issues. Recommendations for assembly time minimization are made in the form of suggested generic design considerations.

Initial feasibility of on-orbit assembly for a strawman spacecraft was assessed. An assembly sequence for this spacecraft was designed and EVA requirements were estimated. Small spacecraft design modifications (in this instance, prebuilding the module/bay 2 stack) may significantly enhance (or even allow) the assembly task.

Analysis of alternative assembly sequences for the ESGP spacecraft could similarly be conducted by selecting any combination of available alternatives of the major questions discussed in section 3.3. The examination of resultant timeline scenarios from alternate assembly sequences can be an effective assessment tool for assembly sequence timeline definition and overall sequence selection based on comparative assembly times.

4. GHRMR Spacecraft

The geostationary high-resolution microwave radiometer spacecraft (fig. 2), which is named for its principal instrument, was conceptualized as a geostationary Earth science platform for geographic regional process environmental studies. A conceptual design of the GHRMR spacecraft was done as part of the Global Change Technology Initiative Architecture Trade Study. The GHRMR spacecraft is described in section 4.1.

The deployment, packaging, and assembly design concepts chosen for the GHRMR are discussed in section 4.3 and section 4.4. They were selected in parallel with and in conjunction with the spacecraft design. This follows the guideline established while studying the in-space assembly of the ESGP spacecraft, that is, to incorporate the in-space assembly requirement into the overall design process as early as possible.

4.1. GHRMR Spacecraft Description

Although the GHRMR is only one of seven instruments on this platform, it dominates the configuration of the spacecraft in that its large size, offset parabolic antenna design, and viewing requirements greatly limit the placement of other instruments as well as placement of the spacecraft bus. The GHRMR instrument was conceptualized during the GCTI trade study by LaRC researchers to meet the requirements of both high spatial resolution and high accuracy. Its Cassegrain multiple reflector antenna design provides wide-angle scanning to cover large portions of a hemisphere of the Earth from a geostationary orbit position. The concept is composed of a 15-m-diameter primary reflector, a 7.5-m-diameter secondary reflector, a moving tertiary reflector, and a phased-array feed system. The large scanning angle requirement necessitates long focal lengths for the GHRMR reflectors, the longest being around 30 m for the primary reflector. The overall spacecraft dimensions are approximately 40 by 24 by 15 m.

Spacecraft designers developed two options for the structure of the GHRMR. Option a (GHRMR-a) was a hybrid deployable/erectable concept which provided the greater capability of the two with a surface sufficiently smooth to operate up to 220 GHz, whereas option b (GHRMR-b) was a completely autonomously deployable concept which provided a surface accuracy sufficient for operation up to 90 GHz. Option b, however, completely eliminated the need for in-space construction. The first option was based on precision segmented reflector technology (ref. 9), which includes solid surface reflector panels and a stiff lightweight supporting truss. This option would be deployed/assembled into its flight configuration at low Earth orbit by using both EVA assembly and autonomous deployment steps. It subsequently uses a space transfer vehicle for transfer and insertion into its operational geostationary orbit. The total mass of the GHRMR-a spacecraft is 6159 kg (13 581 lb).

The second option is based on a solid hexagonal panel concept developed by the Harris Corporation (ref. 15) and was originally designed to operate up to 40 GHz with the technology extrapolated to 90 GHz for the GHRMR application. This option is designed for a single launch to geostationary orbit (GEO). It requires a propulsion stage (Centaur G$'$) for geostationary transfer and orbit circularization. After GEO placement, it will be fully deployed and made operational. The total mass of the GHRMR-b spacecraft is 5433 kg (11 980 lb).

The complexities of packaging and autonomous deployment of either of these very large spacecraft have a significant influence on the overall spacecraft design. The packaging and subsequent deployment at GEO of the GHRMR-b spacecraft was chosen as the primary strawman design task for investigating a large geostationary spacecraft having complex packaging and autonomous deployment schemes. A preliminary packaging-options examination for the partially erectable GHRMR-a spacecraft was also conducted.

Principal elements of the GHRMR spacecraft (whether a or b) are identified in figure 12. At the lower end of the spacecraft is the 15-m-diameter primary reflector and its supporting truss structure. It is connected via truss sections A and B to the feed and tertiary reflector support bay. This bay in turn is connected by truss section C to the spacecraft bus assembly. Each truss bay in a section is 2 m in length along each orthogonal direction. The bus assembly houses spacecraft subsystems (e.g., power switching, communications equipment) and also serves as a mounting platform for the other spacecraft science instruments. A one-piece hinged boom structure attaches the secondary reflector to the spacecraft bus.

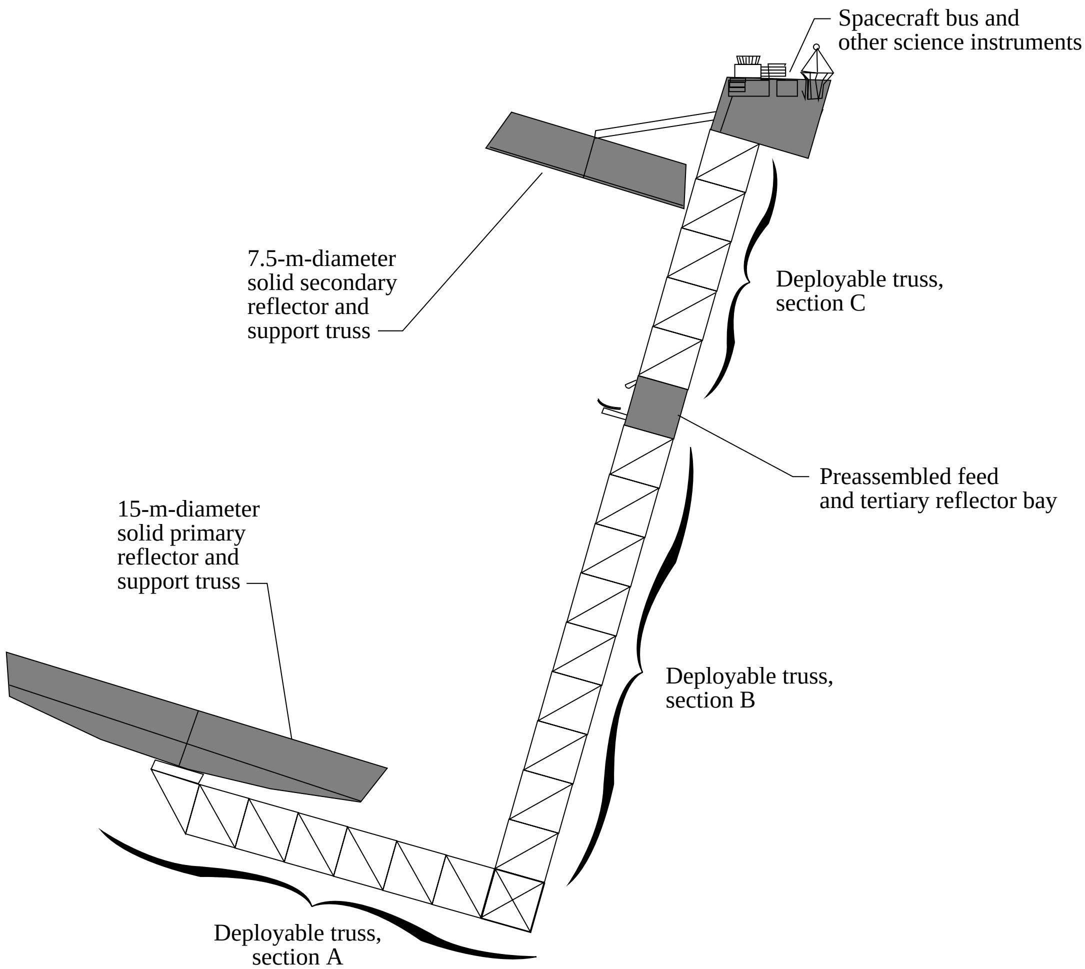


Figure 12. Principal elements of GHRMR spacecraft.

4.2. GHRMR Launch Vehicle Mass Considerations

The Titan 4/Centaur G′ mass capability to GEO is 4536 kg (10 000 lb)—not quite enough for the deployable GHRMR-b whose mass is 5433 kg (11 980 lb). The mass capability of the STS/Centaur G′ to GEO (before this program was canceled) was 6350 kg (14 000 lb). For this study, it was assumed that a Shuttle C (block 1)/Centaur G′ and a Shuttle C (baseline)/Centaur G′ would have at least the same or more mass capability to GEO than the already sufficient, though canceled, STS/Centaur G′. The STS, however, can easily carry either the GHRMR-a or GHRMR-b to LEO, where an STV could be utilized for GEO transfer. The GHRMR packaging task, therefore, becomes a volume-constrained problem, that is, to determine if GHRMR-b can be packaged in Shuttle C (block 1) or even in Shuttle C (baseline) and to also determine if GHRMR-a can utilize the STS orbiter for transfer of the assembly parts to LEO.

4.3. GHRMR-b Deployment/Assembly Concept

The deployment design concept for the GHRMR-b was undertaken first because portions of it would be applicable to GHRMR-a. The spacecraft design was examined, and the major functional elements were identified as the 15-m primary reflector, the 7.5-m secondary reflector, the tertiary reflector and feed assembly, the spacecraft bus including the other science instruments, and the supporting and connecting truss structures. Not shown in figure 12 is the final major element, the geostationary propulsion stage (the Centaur G′).

The descriptions of current and near-term launch vehicle payload envelope capabilities (table I), including the Titan 4, the Shuttle orbiter, the proposed Shuttle C (baseline) and Shuttle C (block 1), gave volume and shape bounds for the GHRMR-b packaging envelope. These envelope restrictions indicated that accommodation of this spacecraft by any current or near-term launch vehicle of reasonable size would require that deployable technology be applied, as a minimum, to the truss structure and the large primary and secondary reflectors.

Table I. Payload Envelopes

Launch vehicle	Length, m	Diameter, m	Payload to to LEO, kg	Payload[a] to GEO, kg
Shuttle orbiter	18.3	4.6	25 850	≈6000
Shuttle C (baseline)	25.0	4.6	70 000	>6000
Shuttle C (block 1)	27.0	7.6	60 000	>6000
Titan 4	18.9	4.6	18 600	4536

[a] In combination with Centaur G′.

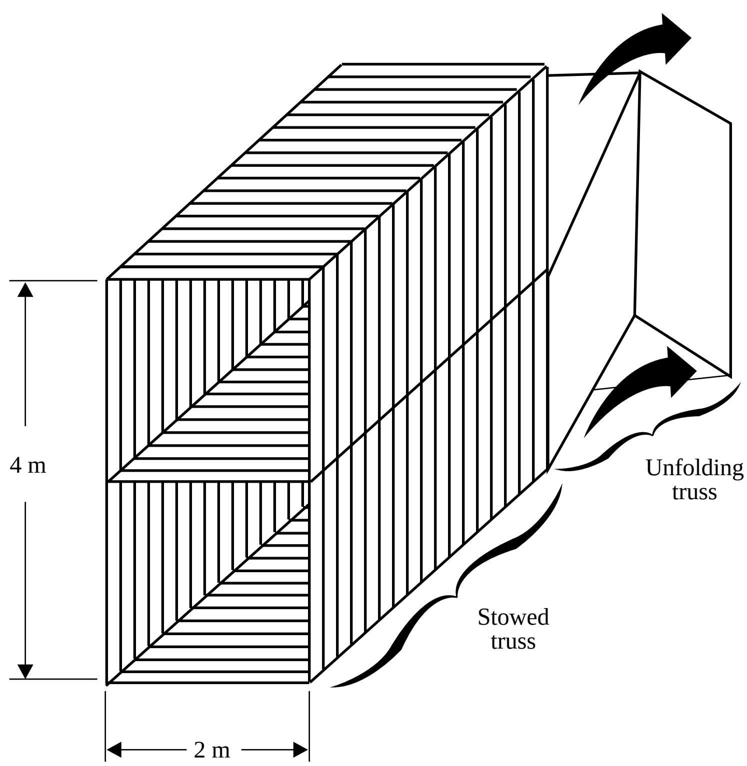


Figure 13. Single-fold, double-stowed truss concept.

After review of a number of deployable truss options, a deployable truss system described as a single-fold, double-stowed design (fig. 13) was selected for the spacecraft truss sections. Since this truss design is not self-deploying, it requires a deploying mechanism. Although this mechanism was not specifically designed in this study, it was verified that sufficient space was available for several different acceptable designs of a motorized deployer. For the 1.25-cm-diameter struts chosen for this spacecraft, a stowed truss length compression of approximately 98 percent was calculated (ref. 16) as achievable (i.e., a deployed 100-m-long truss section would have a length of 2 m in its stowed configuration). For the GHRMR, having truss bay lengths of 2 m, the 14-m truss section A (7 bays) becomes only 0.28 m long when stowed. Likewise, the 20-m truss section B (10 bays) becomes 0.4 m, and the 10-m truss section C (5 bays) becomes 0.2 m.

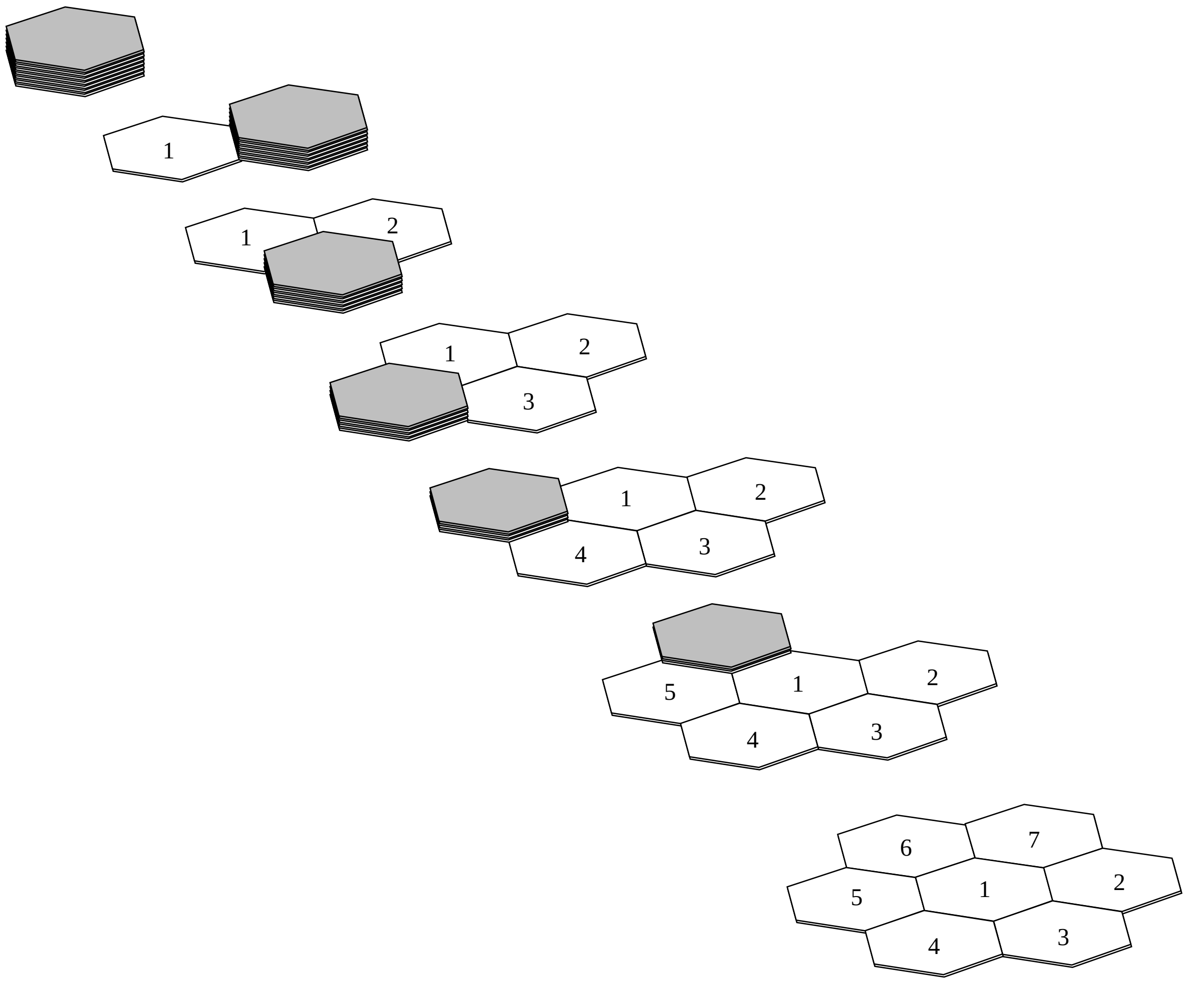


Figure 14. Hexagonal-panel solid-reflector deployment concept developed by Harris Corporation.

The truss bay containing the tertiary reflector and the feed assembly was used as a starting point for the design of the deployment sequence (fig. 12). This bay, whether it is a truss structure or a solid structure, would be preassembled, aligned, and checked out before launch. The deployable truss sections B and C would be attached to opposite longitudinal sides of the feed assembly bay. The spacecraft bus and secondary reflector assembly would be attached to the opposite end of truss section C. At the opposite end of truss section B, truss section A must be attached in an orthogonal direction. Since the truss bays are 2 m in length on all sides, judicious selection of the truss section fold placements makes the direct attachment of truss section A orthogonally to section B possible with deployment in the required direction. The primary reflector assembly would then be attached to the opposite end of truss section A.

A hexagonal-solid-panel concept developed by the Harris Corporation (ref. 15) for extreme precision antenna structures was chosen as the method of assembly-compatible implementation for the large primary and secondary reflectors. This concept is presented in figure 14. Briefly described, this design is a stack of the necessary number of hex panels sitting atop a stowed strongback support structure. The deployment consists of the rotation of selected panels around corner pins, a displacement to the antenna plane, and the locking of these panels into place to form a rigid and accurate antenna surface. The Harris design concept was scaled to meet the requirements of the GHRMR large primary and secondary reflectors while fitting within the launch vehicle payload envelopes. In the stowed configuration, the primary reflector required a hex cross-section envelope 6.6 m long with a 3-m dimension across the hex flats. The secondary reflector envelope is 8 m long with a 2.5-m dimension across the flats.

The spacecraft bus, in addition to accommodating spacecraft subsystems and serving as a mounting platform for the other science instruments, has to fit within the launch vehicle payload envelope. An overall envelope of 4 by 4 by 3.5 m (one face is actually truncated by 15° to give a trapezoidal cross section) satisfied both the spacecraft design requirements of the GCTI study and the packaging constraints of the deployment concept. The GCTI study also indicated that the initial attempt to package GHRMR-b should use either the Shuttle C (baseline) or the Shuttle C (block 1) payload envelope (25 m by 4.6 m diameter or 27.4 m by 7.6 m diameter, respectively).

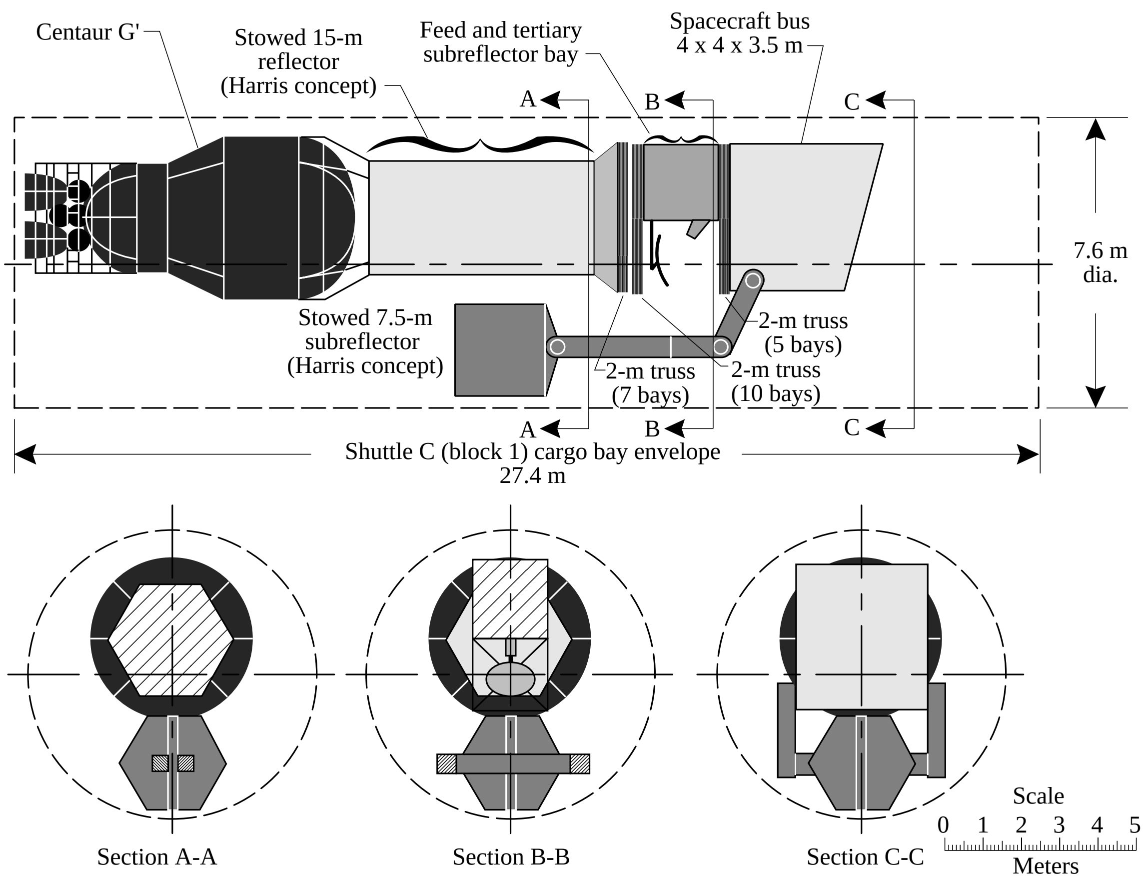


Figure 15. GHRMR-b stowed in Shuttle C (block 1).

The initial packaging design for the GHRMR-b (fig. 15) used the Shuttle C (block 1) payload envelope. Cross-sectional drawing details were also produced (fig. 15) at significant longitudinal positions to verify that the stowed spacecraft did not radially penetrate the payload envelope. A variation of this design incorporated a prelaunch-constructed 7.5-m secondary reflector fitting within the 7.6-m diameter Shuttle C (block 1) envelope enabling use of a more accurate (prelaunch built and tested) antenna surface (fig. 16). The apparent discontinuity in figure 16 between truss sections C and B is accomplished by changing the truss fold sequence. This would allow a different mass distribution in the payload envelope if required by yet-to-be-defined launch vehicle parameters.

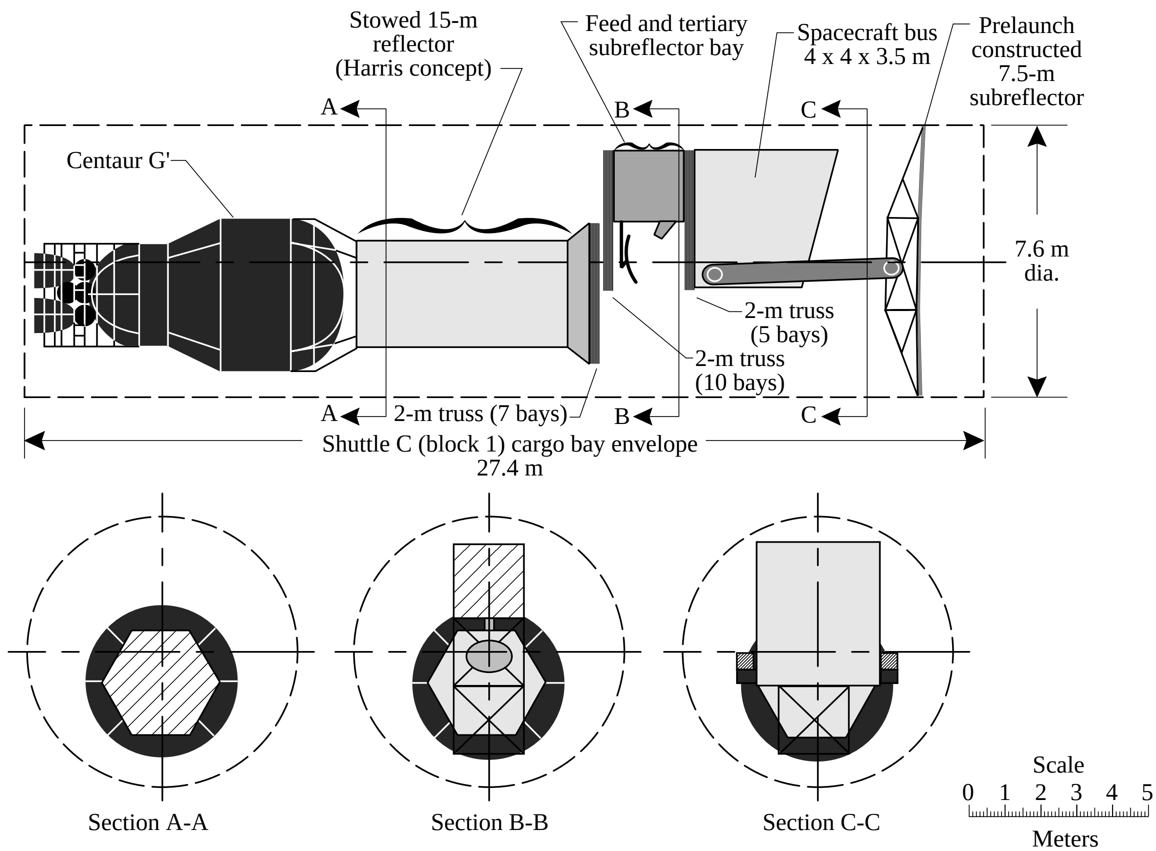


Figure 16. GHRMR-b stowed in Shuttle C (block 1) with prebuilt secondary reflector.

A series of time sequence drawings (fig. 17) show the deployment sequence for the GHRMR-b spacecraft (fig. 15). After achieving GEO, deployment events occur in the following order:

1. Release of the spent Centaur rocket
2. Deployment of truss sections in the order C, B, A
3. Deployment of the secondary reflector
4. Deployment of the primary reflector

Truss section A deploys in two phases. The second phase (the last one and one-half truss bays) is needed to orient the primary reflector in its required pointing direction. A deployment sequence of the GHRMR-b of figure 16 would vary only in that the secondary reflector is preconstructed and its connecting boom would have to rotate into operational position.

An attempt to fit the GHRMR-b packaging design (fig. 15) into the smaller payload envelope of Shuttle C (baseline) (25 m by 4.6 m diameter) was made. A small loss of payload envelope length was easily accommodated by all spacecraft elements. The significant

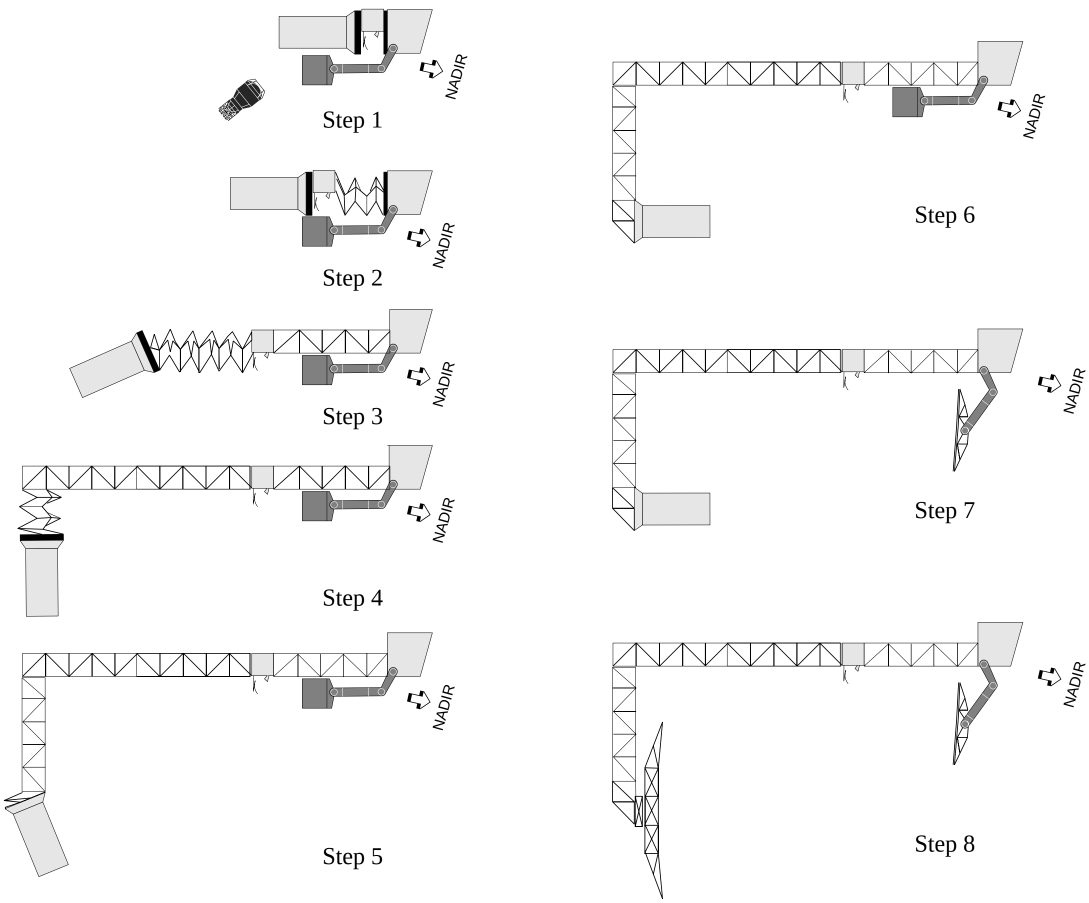


Figure 17. Deployment sequence for GHRMR-b spacecraft.

decrease in diameter was accommodated by all spacecraft elements except the spacecraft bus. A careful reconfiguration of the spacecraft bus (fig. 18) allowed the GHRMR-b to be stowed in the Shuttle C (baseline) launch vehicle (fig. 19).

Figure 20 shows the same GHRMR-b configuration, without the unallowed Centaur G′, in the STS orbiter payload bay. This option is viable in combination with a LEO-to-GEO space transfer vehicle.

4.4. GHRMR-a Hybrid Deployable/Erectable Concept

Finally, packaging for the hybrid deployable/erectable GHRMR-a was examined. The GHRMR-a design retains the spacecraft bus, the prebuilt tertiary reflector and feed assembly, and the deployable truss sections A, B, and C. The primary- and secondary-reflector designs are based on the PSR technology (ref. 9), which includes a stiff lightweight supporting truss and highly accurate solid surface reflector panels designed to operate up to 220 GHz. Although this technology offers higher frequency operational capabilities, the spacecraft must be deployed and the antennas erected at LEO and the spacecraft subsequently transported to GEO. (This study assumes near-term EVA capability at LEO but not at GEO.)

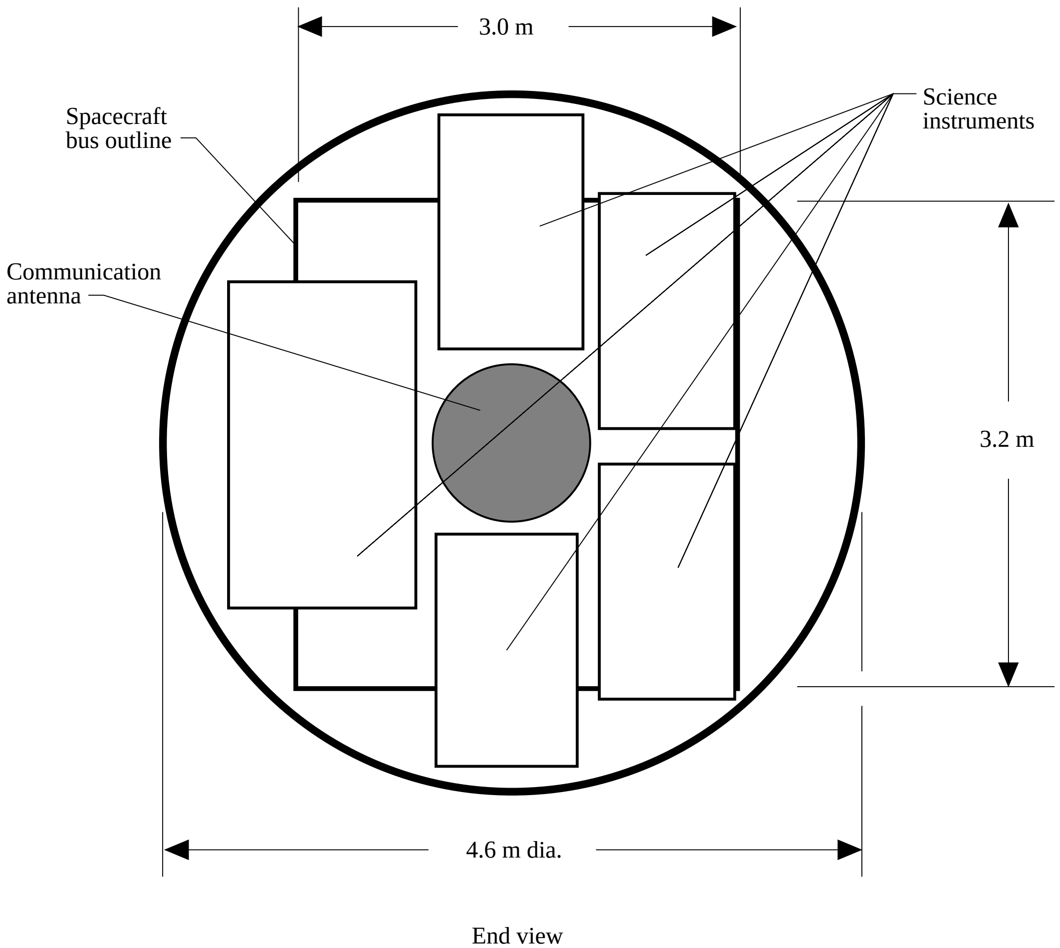


Figure 18. Reconfigured GHRMR spacecraft bus.

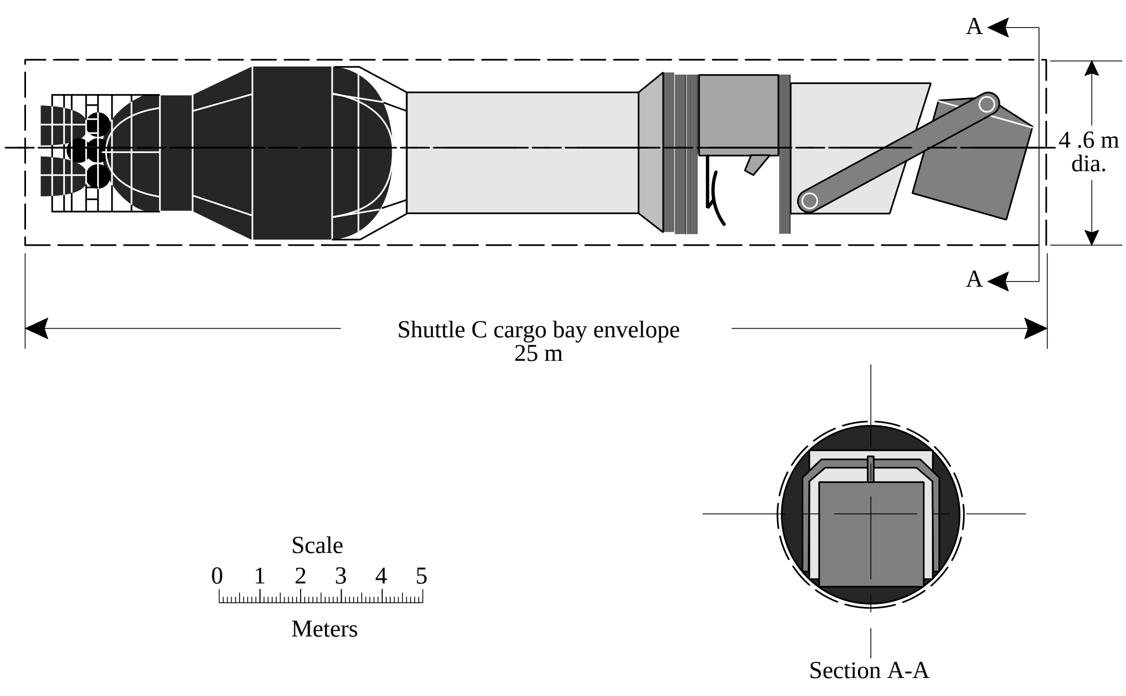


Figure 19. GHRMR-b stowed in Shuttle C (baseline).

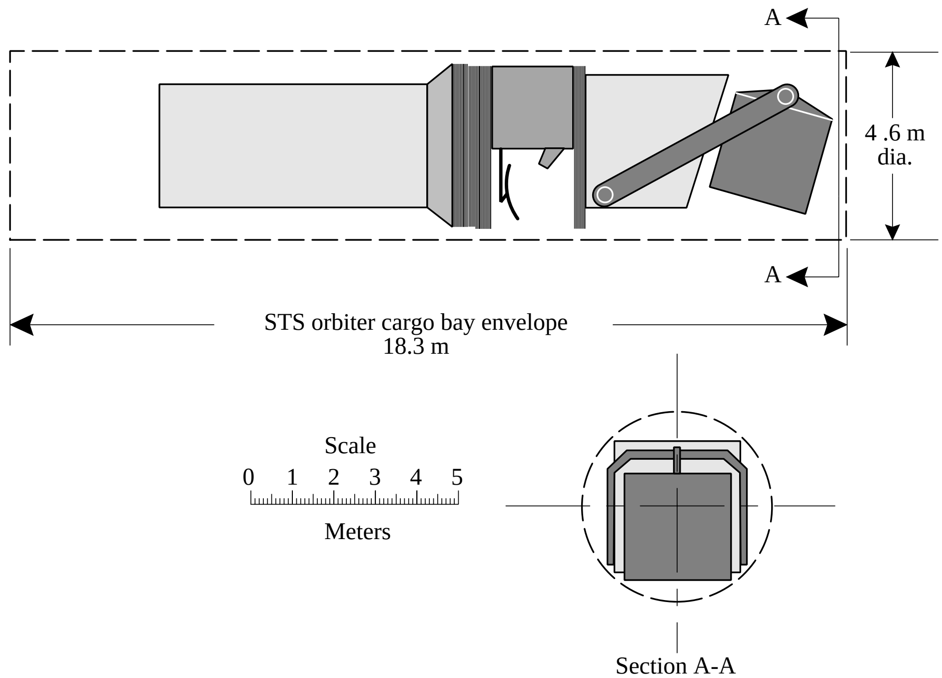


Figure 20. GHRMR-b stowed in STS orbiter.

The GHRMR-a deployment/assembly sequence begins with deployment of the truss sections C, B, and A in that order from the preassembled tertiary reflector and feed assembly bay. This is followed by the EVA assembly of the 15-m and the 7.5-m reflectors. Assembly technology and techniques for on-orbit assembly of the two large GHRMR reflectors can be assumed similar to that required for the 7.5-m reflector on the ESGP spacecraft. It was verified that, where applicable, the assembly choices made in section 3.3 for GHRMR-b could similarly be made for the GHRMR-a reflectors. The sizes and quantities of the struts, nodes, and hex-shaped reflector panels were scaled to meet GHRMR requirements. Assembly tooling devised by LaRC was attached to the Shuttle mission peculiar equipment support structure (MPESS) and used for reflector assembly. The MPESS and tooling must be transported to LEO.

Attention was then focused on the packaging of the GHRMR-a spacecraft in the STS orbiter. Conservative packaging rules, such as "all struts must be at least 1 diameter apart," were used to determine packaging needs for the hybrid deployable/erectable GHRMR. Two packaging options were examined. In the first option, the nodes and struts for the reflector support truss were packaged separately for more volumetric efficiency. This option (fig. 21) was found to require one STS flight for transport to LEO.

More rapid in-space assembly may be possible by packaging each node already attached to one of its connecting struts. The second option looked at what effect this could have on the total volume required for transport to LEO and, subsequently, on the number of STS flights necessary. This option (fig. 22) was found to require two STS flights to LEO.

4.5. GHRMR Task Summary

The seven options that were examined are summarized in table II. Five options (1a through 1e) were examined to support the fully deployable at GEO concept of GHRMR-b. Options 1a, 1b, and 1c (figs. 15, 16, and 19) accomplish the desired results with one launch. The GHRMR-b spacecraft is too heavy for option 1d (Titan 4/Centaur G′). Option 1e (fig. 20) can accomplish the desired results with one STS orbiter launch and the use of an appropriate STV. Options 2a and 2b meet the GHRMR-a concept requirements of minimum STS orbiter launches to LEO utilizing an STV for GEO transfer and placement.

Table II. Packaging Options for GHRMR Geostationary Platform

GHRMR concept	Packaging/launch vehicle option	Assessment
1. Fully deployable at GEO (Harris concept reflector and subreflector)	1a. Shuttle C (block 1)	1a. Viable fit
	1b. Shuttle C (block 1) with prelaunch-constructed 7.5-m subreflector	1b. Viable fit; requires one less GEO deployment
	1c. Shuttle C (baseline)	1c. Marginal (very tight) fit; no other problems at this level of analysis
	1d. Titan 4/Centaur G'	1d. Too heavy
	1e. STS orbiter	1e. Will fit (without Centaur G') but then requires STV to GEO
2. Hybrid deployable/erectable at LEO; STV transfer to GEO (PSR concept for reflector and subreflector)	2a. STS orbiter (structural nodes attached to struts)	2a. Can be delivered to LEO in two STS flights
	2b. STS orbiter (structural nodes and struts packaged separately)	2b. Can be delivered to LEO in one STS flight

4.6. GHRMR Summary Remarks

Building on the information base gathered in the Earth Sciences Geostationary Platform portion of this study, GHRMR strawman options for geostationary on-orbit deployment were devised along with a concept for on-orbit assembly at LEO with STV transfer to GEO. The study shows that very large geostationary spacecraft may be designed for geostationary on-orbit deployment with a single launch vehicle (available now or in the near term) by using and emphasizing the early integration of the on-orbit requirements of the design. The success of this effort was due in great part to the early incorporation of the on-orbit deployment/assembly design requirements.

By judiciously choosing the deployment, packaging, and assembly options, with the appropriate selection of available launch vehicles, a variety of alternatives exist for implementing the GHRMR in geostationary orbit. Most of these options require only one launch for geostationary spacecraft implementation, although some require a space transfer vehicle for geostationary transfer.

The study also indicated that hybrid deployable/erectable spacecraft may not only offer viable options but may present unique opportunities as well. In this case, the GHRMR-a concept allows use of a more accurate reflector surface (the precision segmented reflector technology versus the hex-panel deployable concept) for higher radio frequency utilization and subsequent improved instrument effectiveness. Option selection here could be based on mission requirements.

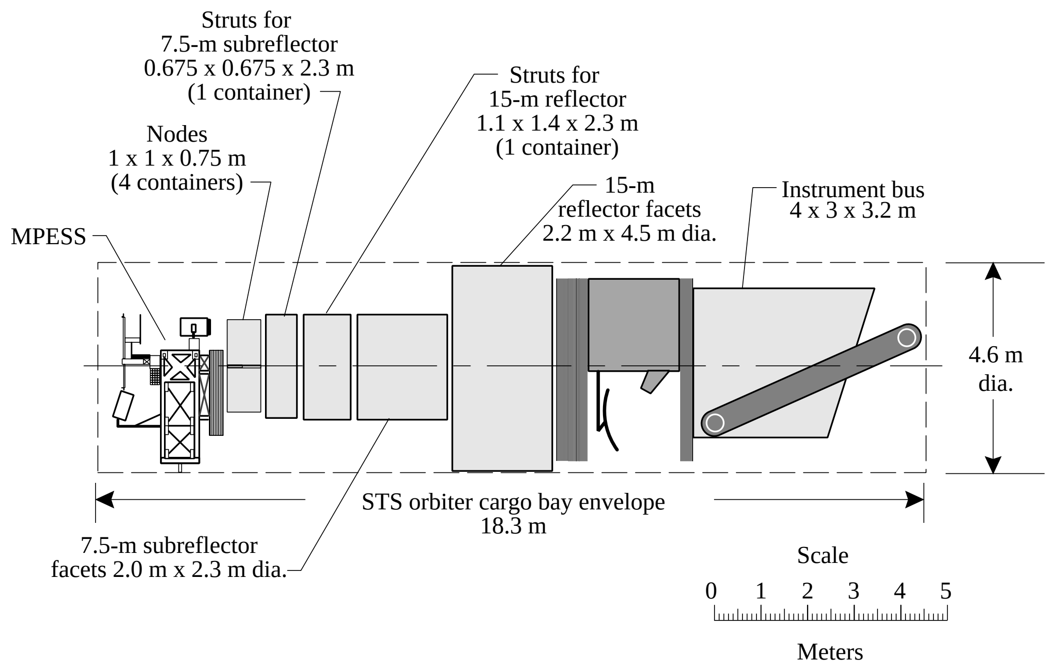


Figure 21. GHRMR-a hybrid deployable/erectable geostationary platform stowed in Shuttle orbiter cargo bay (struts and nodes packaged separately).

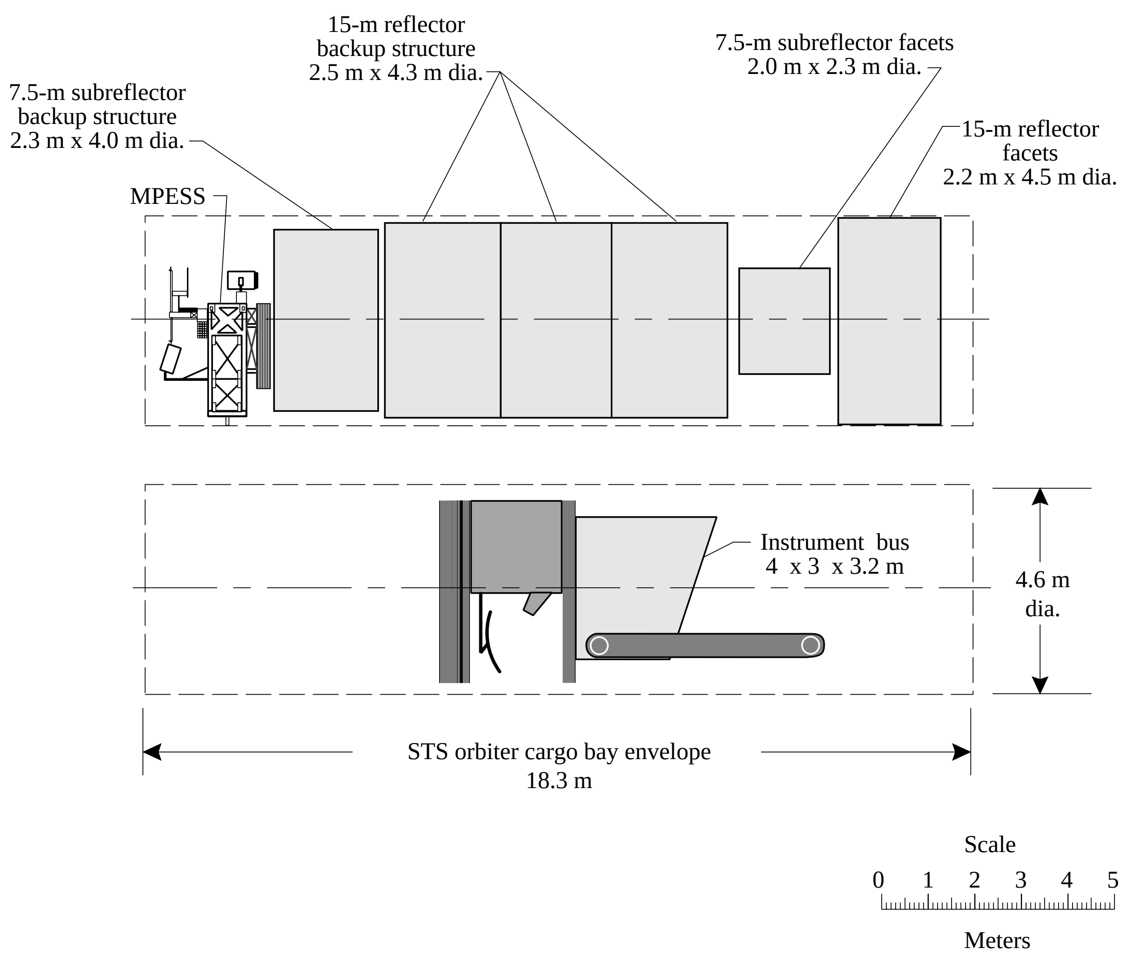


Figure 22. GHRMR-a hybrid deployable/erectable geostationary platform stowed in two Shuttle orbiter cargo bays (nodes preattached to struts).

5. Concluding Remarks

Previous studies have investigated various aspects of in-space assembly by applying experience and sound engineering judgment to a particular spacecraft or requirement. This study also used the assembly of specific spacecraft as a mechanism for on-orbit assembly investigation. In addition, this study treated two spacecraft as strawman test cases, postulating a spacecraft design process to accommodate on-orbit assembly design for geostationary spacecraft, and finding generic guidelines (for the near term) for the in-space LEO assembly of a large geostationary spacecraft.

As a result of applying the postulated design process and generic guidelines to the ESGP spacecraft, it is recommended that in-space assembly tasks be planned as an integral part of the spacecraft design. This assures compatibility between the spacecraft and its assembly requirements as well as enhancing final spacecraft design, cost and time savings, and assembly safety. If not an actual part of the spacecraft design process, the design of the assembly process should be incorporated as early as possible to optimize results. A particularly important step in this overall process is the assessment of actual compatibility between the spacecraft design and each of its assembly steps, including the order in which they take place.

This study went a step further than previous studies and used the GHRMR spacecraft to explore the possibilities for in-space deployment of large GEO spacecraft with positive results. Early incorporation of deployment/assembly design into the spacecraft design allowed tailoring of the spacecraft design to accommodate both critical launch vehicle packaging requirements and the capability for geostationary on-orbit deployment.

A significant portion of the GHRMR spacecraft effort involved its successful packaging, whether in its completely autonomous deployment configuration or in the modified hybrid deployable/erectable version. Here the available launch-payload-envelope volume (and its shape) was found to be of prime importance. Of nearly equal importance were innovative packaging techniques (in this case, the hex-panel reflectors and the single-fold, double-stowed truss design and their respective interfaces). The study results indicate that in-space deployment of a large spacecraft is possible, and various alternative options may be feasible.

The success of in-space assembly and deployment of any large spacecraft design will depend on how early and on how well the design of the deployment/assembly of the spacecraft is integrated into the overall spacecraft design and launch vehicle constraints.

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13. ABSTRACT *(Maximum 200 words)*
Large spacecraft, particularly in geostationary Earth orbit (GEO), require special attention to the design challenges of launch vehicle packaging, deployment, and/or on-orbit assembly. Design studies of two different GEO spacecraft required that packaging, deployment, and on- orbit assembly analyses be conducted to establish the viability of these concepts for future NASA missions. This study used these analyses as strawman concepts for an investigation of packaging, deployment, and on- orbit assembly techniques. It also revealed generic guidelines for in- space assembly and highlighted the importance of early integration of packaging, deployment, and on-orbit assembly requirements into the spacecraft design. The first of the study spacecraft were used to study the definition and analyses of on-orbit assembly options for large GEO spacecraft. The second study spacecraft required investigation of the feasibility of deploying large spacecraft at GEO. The second spacecraft was also used to examine the packaging requirements of a deployable spacecraft and the packaging requirements for a hybrid assembled/deployable version of that spacecraft. This investigation was done with attention to minimum volume (and minimum launches) and to the relationship between packaging and spacecraft deployment and final configuration.

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