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SUMMARY

In this study, a solar-wing-array concept is investigated which meets space-station requirements for minimum fundamental frequency, component modularity, and growth potential. Several of the design parameters for this array concept are varied, and the resulting effects on array dynamic characteristics are assessed. The wing-array concept is incorporated into a free-free space-station model, and the transient response to an applied step load of various durations is studied.

Study results show that by proper design of the wing array, any change in the space-station center-of-gravity location caused by rotation of the solar arrays can be eliminated. The transition truss, which connects the solar arrays to the space-station rotary joint, and the solar-array support truss can both be designed so that the fundamental frequency requirement chosen for the space station is met. The space station grows from its initial configuration by the addition of more modules and solar arrays. Substantial reductions in the space-station frequencies and changes in mode shapes occur during the growth phase. Thus, knowledge of the final configuration must be available if the minimum vibration frequency requirement is to be met for the life of the space station.

INTRODUCTION

Now that the Space Transportation System (STS) is fully operational, the aerospace community has begun considering projects which require large structures in space. As a result, the possibility of orbiting a permanent U.S. space station by the early 1990's is receiving increasing attention. (See ref. 1.) A system consisting of the STS and a permanent space station would permit efficient and continuous operation in the space environment. (See ref. 2.)

Figure 1 shows an initial space-station study configuration which is described in reference 3. The space station consists of two basic units - a central collection of logistics, habitat, utility, laboratory, and docking modules, and two large solar wing arrays. For maximum energy collection efficiency, it is desirable to keep the solar arrays pointed directly at the Sun during Earth orbit. The central modules, however, may need to be pointed at the Earth, or in some other direction, depending on mission requirements. Therefore, the wing arrays are attached to the central module by rotary joints which provide the required rotational degree of freedom between the modules and the arrays.

The large amount of power required for space-station operations has a great impact on its design. When fully operational, the space station must receive 150 kW of continuous power at the bus bar. This requirement leads to the design of large solar wing arrays as shown in figure 1. Each wing is 76.2 m (250 ft) long, and 24.4 m (80 ft) wide, has 1858 m^2 (20 000 ft^2) of silicon solar cells, and generates enough power while in sunlight to provide 75 kW to the space station and to charge the storage batteries. (See ref. 3.) The batteries provide the required 150 kW of power when the space station is in the Earth's shadow.

The solar arrays shown in figure 1 are large flexible solar blankets attached to a coilable longeron beam. (See ref. 4.) The beam serves both to deploy the solar

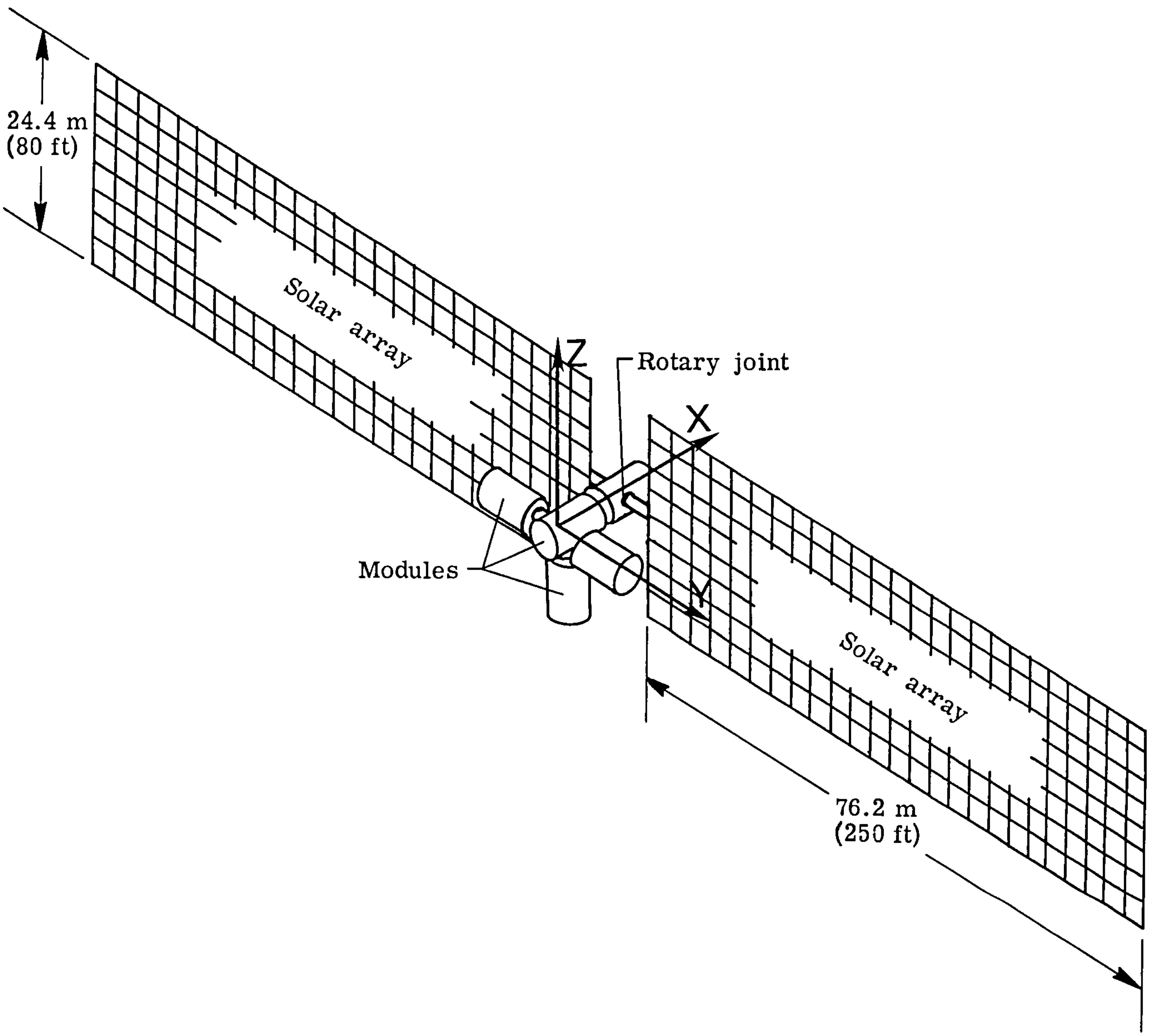


Figure 1.- Initial 150-kW space-station configuration.

blankets and to provide the wing structural stiffness once deployment is complete. Because of the extreme length of the solar arrays and the low stiffness of the coilable longeron beam supporting the wings, this configuration has a very low fundamental frequency (0.008 to 0.012 Hz). Increasing the space-station fundamental frequency is a desirable goal which, if attained, would reduce the mass and complexity of the space-station attitude control system. This goal can be achieved by increasing the stiffness of the wing-array structure. Since the maximum diameters of coilable longeron beams built to date are less than 1 m, these deployable truss beams are not currently able to provide adequate stiffness for the large solar wing arrays of the space station shown in figure 1.

Important aspects of the space-station design are modularity and potential for growth. Although the final space-station configuration will consist of many modules and enough solar arrays to generate 150 kW of power, it may be several years before that level of capability is attained. The initial space station may be smaller and may have a lower power requirement. Space-station growth, however, must be integrally designed into the structure from the start. Designing space-station structures and systems to be modular allows for easy replacement of defective components and upgrading of out-of-date components over the life of the space station.

The solar-wing-array design shown for the space-station concept in figure 1 suffers from two major drawbacks: a low fundamental frequency (0.008 to 0.012 Hz, depending on the size of the coilable longeron beam used); and because the arrays are deployed as a single unit, a lack of growth potential. The first objective of this study is to design a solar-wing-array concept which satisfies space-station requirements for modularity, growth potential, and fundamental frequency. Modularity and growth requirements can be met by using a combination of erectable- and deployable-structures technologies. The fundamental frequency requirement is met by making the arrays shorter and wider and by increasing the stiffness of the wing-array support truss. A fundamental free-free elastic frequency of at least 0.4 Hz is chosen as a design goal. The second objective is to subject the array to changes in several structural-design parameters and to assess the corresponding effects on the vibration modes and frequencies of the wing array. Finally, the wing-array concept is incorporated into a free-free space-station model so that the space-station transient response to applied loads can be studied.

SYMBOLS

A	area, m^2 (ft^2)
d	distance between origin and rotary-joint center of rotation, m (ft)
E	Young's modulus, Pa (N/m^2)
F	step-load allowable magnitude, N (lbf)
f	frequency, Hz
g_ξ	constant defined by Newton's second law $\left(\frac{\text{Force} = \text{Mass} \times \text{acceleration}}{g_\xi}\right)$
I	moment of inertia, m^4 (ft^4)
L	length, m (ft)
M	mass, kg (lbm)
P	strut axial force, N (lbf)
P_{cr}	Euler buckling load, N (lbf)
r	offset distance between array center of gravity and rotary-joint axis, m (ft)
t	time, sec
X,Y,Z	rectangular Cartesian coordinates

x	distance of center of gravity from origin, m (ft)
Δ	change in space-station center of gravity, m (ft)
ρ	material density, kg/m^3 (lbm/ft^3)

Subscripts:

A	arrays
B,C	array positioned at points B or C
M	modules
S	space station
s	strut
T	triangular support truss
1,2,3	mode numbers

DESCRIPTION OF SOLAR-WING-ARRAY CONCEPT

The concept devised to meet the space-station requirements for stiffness, modularity, and growth potential is shown in figure 2. The space station consists

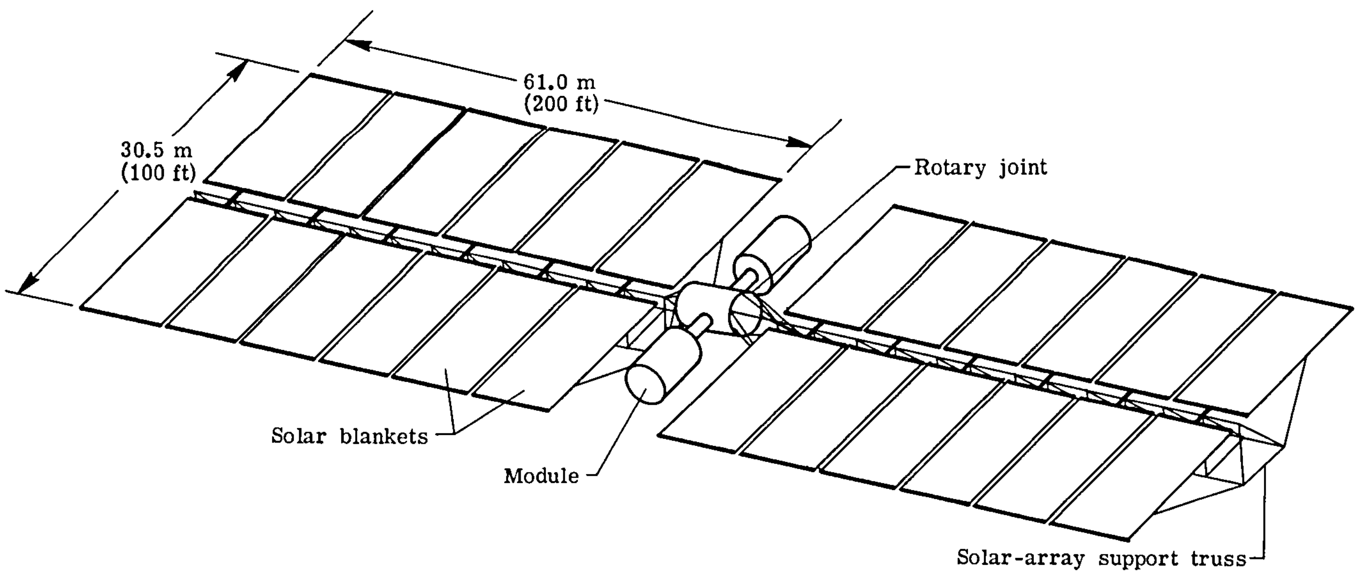


Figure 2.- Space-station concept with central-module cluster and solar wing arrays.

primarily of a central collection of modules and two large solar wing arrays. The modules are limited by the size of the Shuttle cargo bay to a 4.27-m (14-ft) maximum diameter. By mounting the rotary joints (which attach to the wings) on the ends of a module (as opposed to the side as shown in fig. 1), the depth of the support truss at the cantilever root can be increased to the diameter of a module (i.e., 4.27 m

(14 ft^2)). As in figure 1, the two solar arrays have a total collection area of 3716 m^2 (40 000 ft^2).

With this concept, astronauts, aided by a movable erector (not shown), would erect the large backbone truss two bays at a time in a manner similar to that described in reference 5. The space-station frequency requirement ($f_1 > 0.4$ Hz) would then be met by proper sizing of the truss. As the astronauts complete each two-bay section of the truss, they deploy two solar blankets and attach the outriggers. The installation of utilities, such as electrical bus bars, switch gear, and radiators would be integrated into the erection process. The erector could be used to add bays to the array as power requirements increase and would thus meet the requirement for modular growth capability. In addition, the erector would serve as a maintenance and logistics vehicle from which astronauts could service or upgrade the array.

Figure 3 shows details of the solar wing array, including the support truss, the transition truss, and a module with a rotary joint. The solar-array support structure is a single-laced three-longeron truss beam which would be erected by astronauts

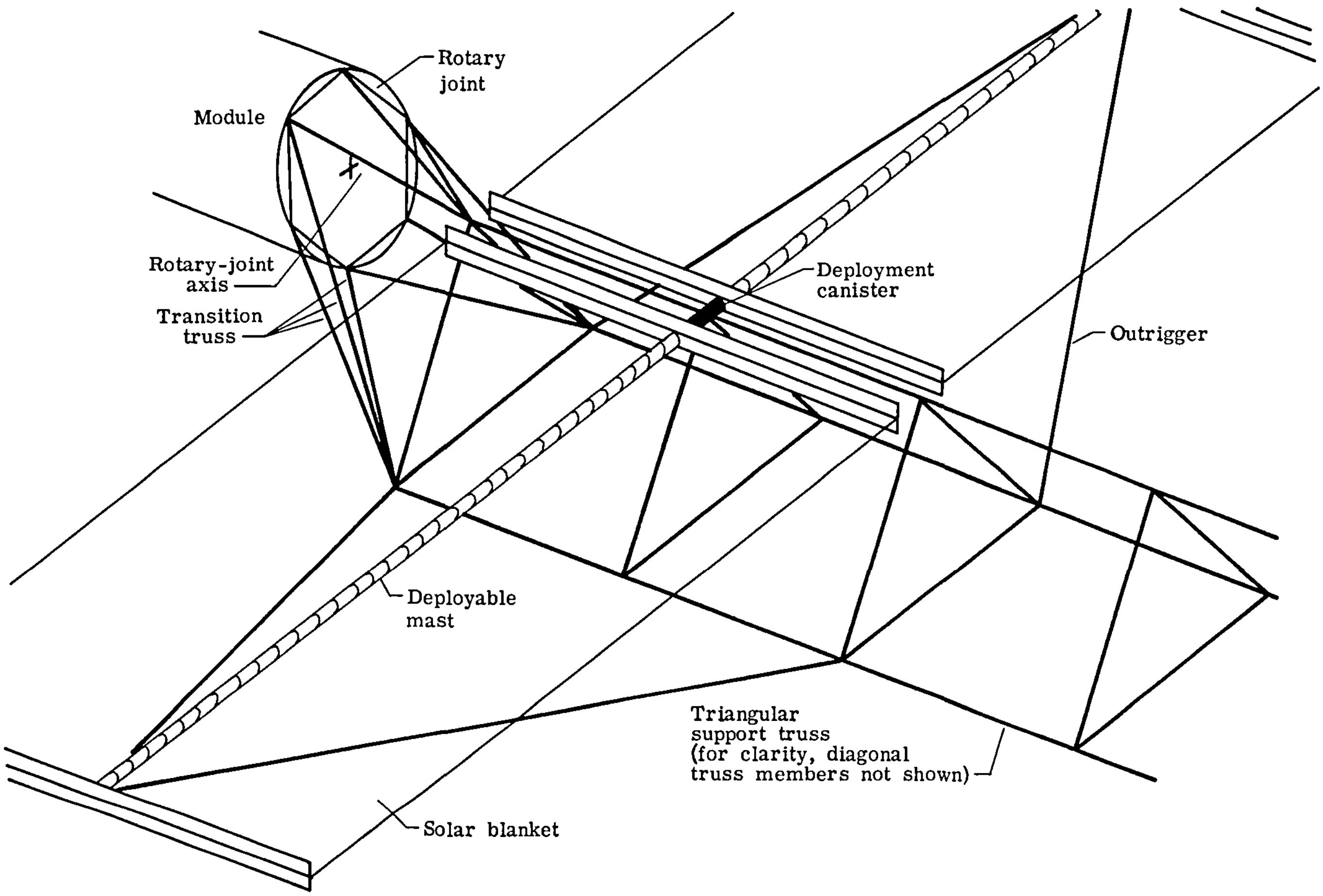


Figure 3.- Details of offset wing array.

riding a moving erector that runs along the truss interior. The solar arrays are packaged as folded solar blankets and are attached to a deployable mast. A deployment canister which contains the packaged mast is attached to the top longeron of the truss. As the mast is deployed perpendicular to the longeron and out from the truss, it unfolds the blankets. When fully deployed, the mast applies tension to the blankets and provides torsional stiffness. A pair of outriggers, extending from nodes on the bottom longeron of the truss beam, connect to the free end of the deployable mast to form a stiff tripod. Similarly, another blanket is deployed from the top longeron in the opposite direction. One complete section of the solar array, as shown in figure 3, consists of two bays of the triangular truss beam and two 15.2-m (50-ft) by 10.2-m (33.3-ft) solar blankets with associated support structure. Modularity and growth are inherently designed into this solar array so that bays can be added to the free end of the truss and more solar blankets deployed, or existing blankets can be replaced or upgraded.

One of the main space-station requirements is that the solar arrays be capable of tracking the Sun independently of the orientation of the central modules. In the configuration shown in figure 3, the rotary joints are mounted on the ends of one of the central modules, and the wing arrays are attached to the rotary joints. Therefore, the full 4.27-m (14-ft) diameter of the module can be used to support a rotary joint. A transition truss connects the three longerons of the solar-array support truss to six points to form a hexagon on the rotary joint. The solar arrays rotate about an axis which passes through the center of the module end. Since the solar blankets make up most of the solar-array mass, the center of gravity (c.g.) for the array lies very close to an axis which runs along the top longeron of the triangular truss. The total change in the space-station c.g. due to rotation of the solar arrays Δ is given by

$$\Delta = \frac{2r}{1 + \left(\frac{M_M}{M_A}\right)}$$

where r is the distance between the solar-array c.g. and the array center of rotation, M_M is the total mass of the modules, and M_A is the total-array mass. (See appendix.) To minimize the space-station c.g. shift caused by array rotation, as described in equation (A8), the top longeron of the solar array is aligned with the rotary-joint center of rotation (rotary-joint axis). The transition truss serves to connect the triangular truss to the rotary joint and provide the required array offset.

For purposes of sizing the graphite-epoxy three-longeron triangular truss shown in figure 3, it was assumed that the fundamental mode would be cantilever bending. Although the frequency requirement for the wing is 0.4 Hz, the value of 0.5 Hz was chosen for sizing to account for the possibility that a torsion mode might exist below the wing bending mode and to compensate for the flexibility which is introduced at the root of the cantilever by the transition section. The fundamental cantilever-bending frequency (ref. 6) is given by

$$f = \frac{3.52}{2\pi}\sqrt{\frac{EI_T\, g_\xi}{M_A L_A^3}} \qquad (1)$$

where

E Young's modulus

I_T moment of inertia of the triangular support truss

g_ξ = 1 kg-m/N-sec^2 (386 lbm-in/lbf-sec^2)

M_A array mass (blankets + truss)

L_A length of solar wing array

The moment of inertia of a three-longeron truss with an equilateral-triangular cross section is

$$I_T = \frac{A_s L_s^2}{2} \tag{2}$$

where

A_s individual strut cross-sectional area

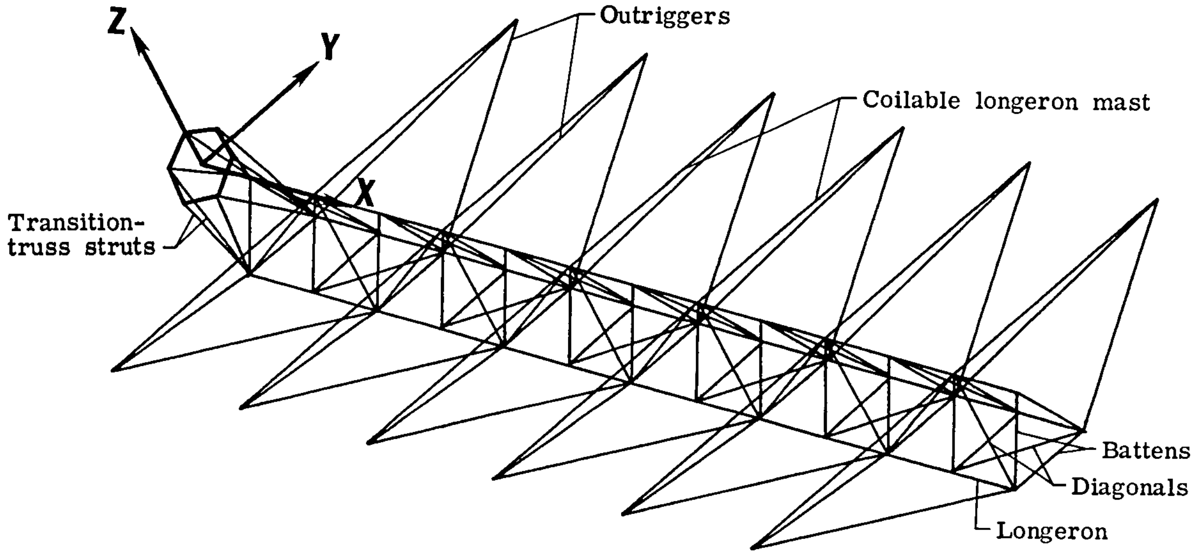
L_s strut length (length of one of the triangular legs)

The truss is constructed of 0.051-m (2-in.) diameter graphite-epoxy tubes which have a wall thickness of 0.0015 m (0.06 in.) and an axial modulus of 276 GPa (40×10^6 lbf/in^2). The corresponding strut area A_s is 2.36 cm^2 (0.3657 in^2).

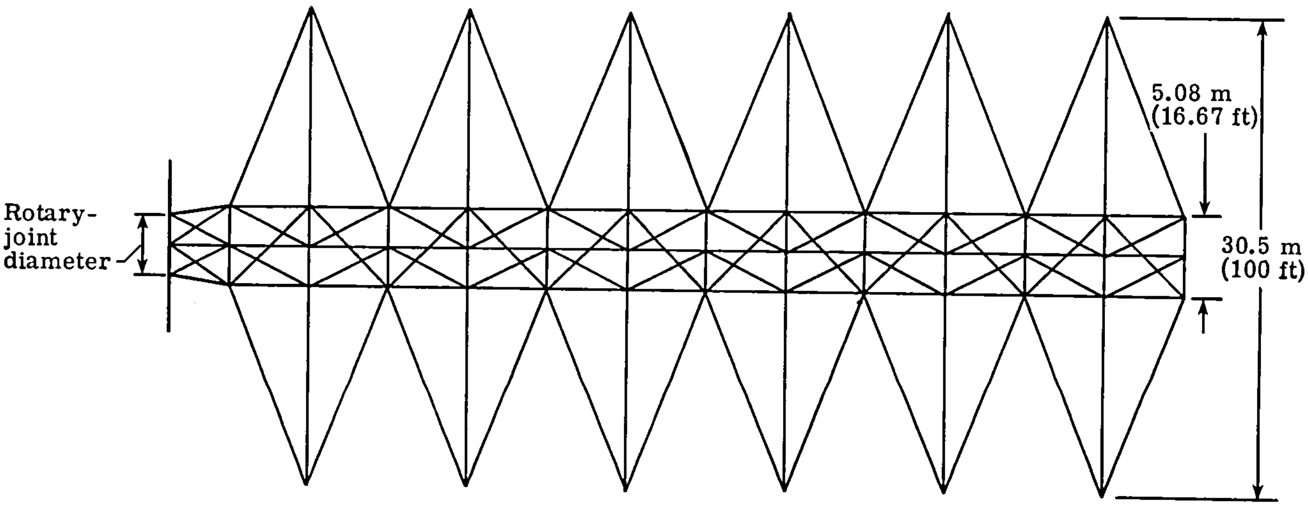
An estimate of the mass of an 1858-m^2 (20 000-ft^2) solar wing array was obtained by simply scaling up the numbers for the deployable array to be used in the Solar Array Flight Experiment. (See ref. 7.) This array has an area of 124 m^2 (1335 ft^2) and a mass of 225 kg (496 lbm). Thus, for purposes of this calculation, the mass of a solar wing array M_A is assumed to be 15 × 225 kg (496 lbm), or 3375 kg (7441 lbm). Since shortening the wing-array length helps to increase its fundamental bending frequency, the array is shortened from 76.2 m (250 ft), as shown in figure 1, to 61.0 m (200 ft). To achieve the proper array area, the array width is increased to 30.5 m (100 ft). Substituting equation (2) into equation (1), requiring $f > 0.5$ Hz, and solving equation (1) for the triangular truss strut length gives $L_s > 4.32$ m (170 in.). The transition truss (fig. 3), by adding length and an offset to the wing array, causes a reduction in frequency. Therefore, the quantity L_s was chosen to be 5.08 m (200 in.). This choice gives 12 bays in the support truss for each 75-kW wing.

FINITE-ELEMENT ANALYSIS

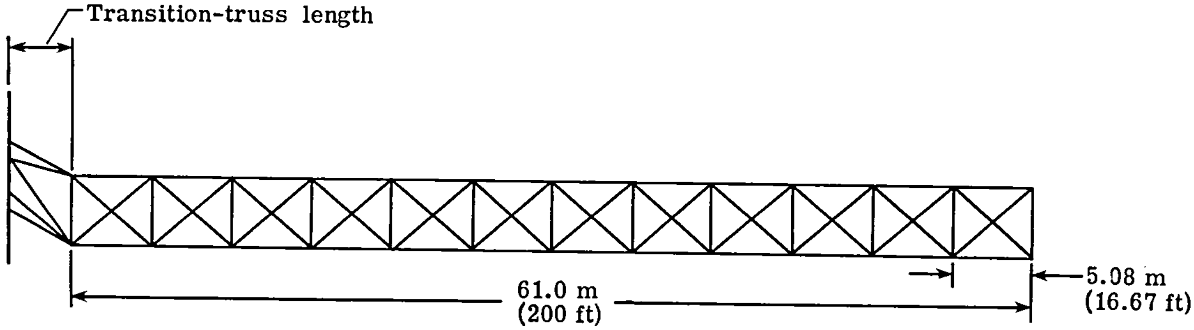
The finite-element model of the 12-bay cantilevered solar wing array used in this study is shown in figure 4. The various members which make up the wing, along with the wing geometry, are shown in figure 4(a). The longerons and battens are 5.08 m (16.67 ft) long and the solar arrays, which extend outward from the top



(a) Member definitions.



(b) Wing Y-dimensions.



(c) Wing X-dimensions.

Figure 4.- Cantilever solar-wing-array finite-element model.

longeron, are 15.24 m (50 ft) long. The total length of the 12-bay triangular support truss is 61.0 m (200 ft).

All the members in the model (longerons, battens, diagonals, solar arrays, outriggers, and transition-truss struts) are rod (axial stiffness only) elements. The member areas are 2.36 cm^2 (0.3657 in^2), and the modulus used is 276 GPa (40×10^6 lbf/in^2). The material density ρ is 1605 kg/m^3 (0.058 lbf/in^3). Although the solar blankets themselves are not modeled, their mass is accounted for. The mass of each solar blanket plus solar cells, deployable mast, and canister is approximately 240 kg (530 lbm). Thus, a nonstructural mass of 15.8 kg/m (0.8833 lbm/in.) is distributed along the solar-array members so that the solar-blanket mass effects are included in the model.

Three basic structural design parameters can be varied in this array concept. The stiffness (modulus times cross-sectional area) of the transition-truss struts (fig. 4(a)), although initially assumed to be the same as that of the other array members (outriggers, solar arrays, longerons, etc.), can be varied, as can the rotary-joint diameter (fig. 4(b)) and the transition-truss length (fig. 4(c)). As these three parameters are varied in the model, the vibration modes and frequencies of the array are calculated using the finite-element program.

Because of the level of finite-element modeling used, the following three limitations were placed on the analysis:

1. Joint mass was not included.

2. Since the array members were modeled with rod elements, the bending modes and frequencies of these members could not be predicted by the model. This would be of some consequence in short-wing arrays, for which the fundamental mode might be the bending of individual members rather than the bending or torsion of the entire array.

3. Although the blanket mass was included, the effects of blanket dynamics and prestress were not.

These limitations identify refinements which could be made in a more detailed study of a particular array configuration.

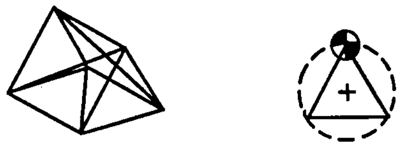
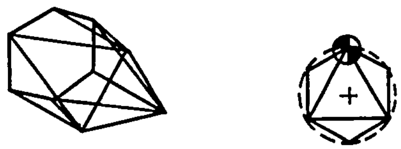
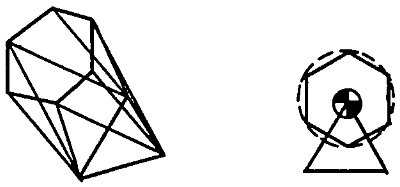
RESULTS AND DISCUSSION

Array Offset Effects

One of the main requirements for the space-station solar arrays is that they be able to rotate about their longitudinal axis. The corresponding effect on the wing-array design, as shown in figure 4, is to offset the array so that its c.g. is aligned with the rotary-joint axis. In table I, the effects of transition-truss geometry and offset on the transition-truss mass and the wing-array frequencies are summarized. The results of table I are for a transition-truss length of 3.05 m (10 ft) and a rotary-joint diameter of 5.87 m (19.25 ft).

In the first configuration shown in table I, the transition truss is one bay of a single-laced triangular truss and has the effect of adding one more bay to the support truss. However, this extra bay is shorter than those in the support truss, 3.05 m (10 ft) instead of 5.08 m (16.67 ft). The transition-truss longerons run straight into the rotary joint as shown by the sketch in the first column of table I.

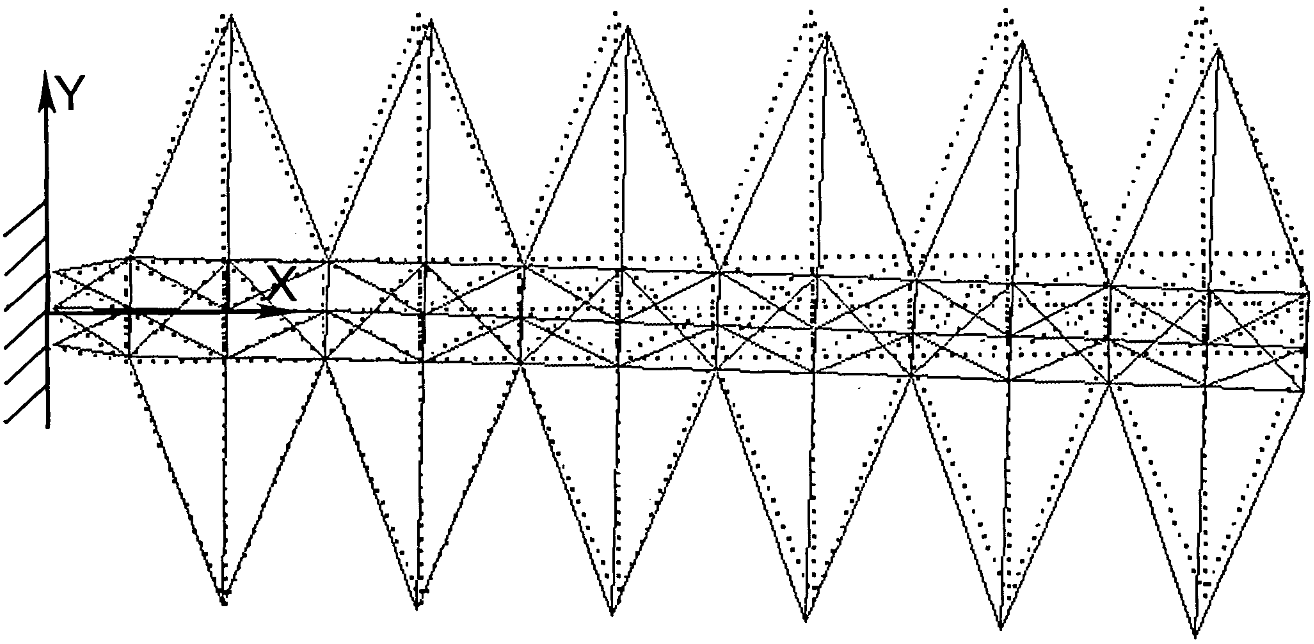
TABLE I.- EFFECTS OF TRANSITION-TRUSS GEOMETRY AND OFFSET ON WING FREQUENCIES

Transition-truss geometry[a]	Transition-truss mass, kg (lbm)	Solar-wing-array frequencies, Hz		
		f_1[b]	f_2[c]	f_3[d]
 Single-laced triangular truss	10.2 (22.5)	0.464	0.527	0.569
 Nonoffset, base hexagon	8.3 (18.2)	0.472	0.536	0.580
 Offset, base hexagon	10.5 (23.2)	0.382	0.491	0.566

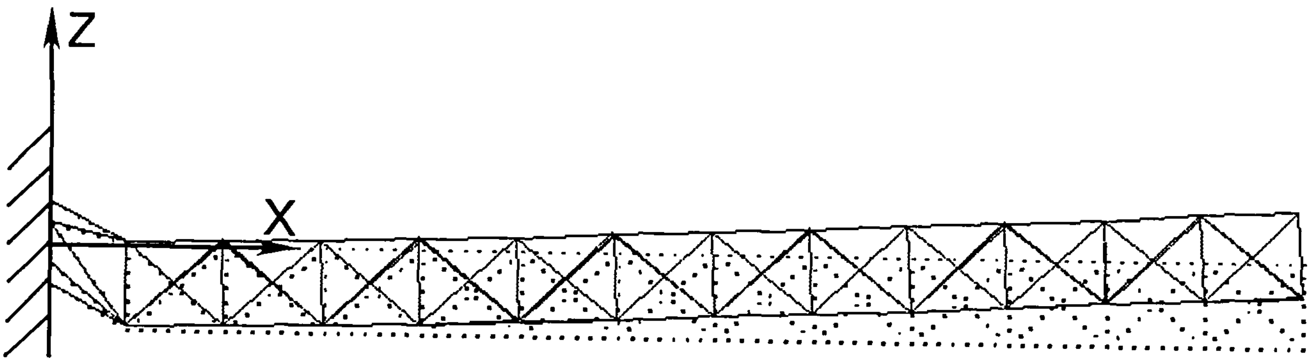
[a]Transition-truss length = 3.05 m (10 ft); Rotary-joint diameter = 5.87 m (19.25 ft).
[b]First mode is bending about Z-axis.
[c]Second mode is bending about Y-axis.
[d]Third mode is torsion about X-axis.

With this transition-truss arrangement, the center of stiffness of the support truss is aligned with the rotary-joint axis, but the array center of mass lies on a circle circumscribing the three truss longerons. The table also gives the transition-truss mass, 10.2 kg (22.5 lbm), and the first three wing cantilever frequencies. The corresponding mode shapes are shown in figure 5. The first wing frequency is 0.464 Hz and corresponds to bending in the plane of the solar arrays (about the Z-axis). The second frequency, 0.527 Hz, corresponds to wing bending out of the plane of the array. The third frequency, 0.569 Hz, corresponds to first torsion about the X-axis.

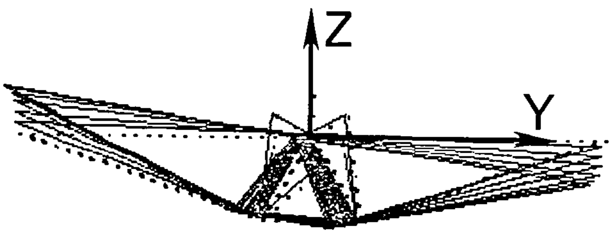
The second entry in table I shows the effects of changing the transition-truss geometry. Now the three longerons of the array support truss are connected by nine members in the transition truss to the six vertices of a hexagon on the rotary joint. A hexagonal arrangement was chosen in order to reduce the magnitude of the point loads to the structure. Once again, the center of stiffness of the array support truss is aligned with the rotary-joint axis. Using a hexagonal rotary-joint



(a) First mode - bending about Z-axis.



(b) Second mode - bending about Y-axis.



(c) Third mode - torsion about X-axis.

Figure 5.- First three solar-wing-array cantilever mode shapes.

attachment instead of a triangular one gives a slight increase (≈2 percent) in the fundamental frequency for a sizable reduction (≈19 percent) in the transition-truss mass.

The third and final entry in table I shows how offsetting the triangular support truss affects the transition-truss mass and the array frequencies. The top longeron of the triangular truss (near the center of gravity) is now aligned with the rotary-joint axis. The resulting transition-truss mass is 27 percent more than in the nonoffset configuration in table I. Offsetting the array also causes a reduction in the array frequencies of 19 percent for f_1, 8.4 percent for f_2, and 2.4 percent for f_3.

Center-of-gravity calculations performed using the finite-element program indicated that the array c.g. was very near the top longeron as expected. The actual location was 0.284 m (0.933 ft) below the top longeron. The total change in the space-station c.g. Δ is plotted as a function of M_M/M_A for two cases in figure 6. In the first case (offset, base hexagon), the top longeron is aligned with

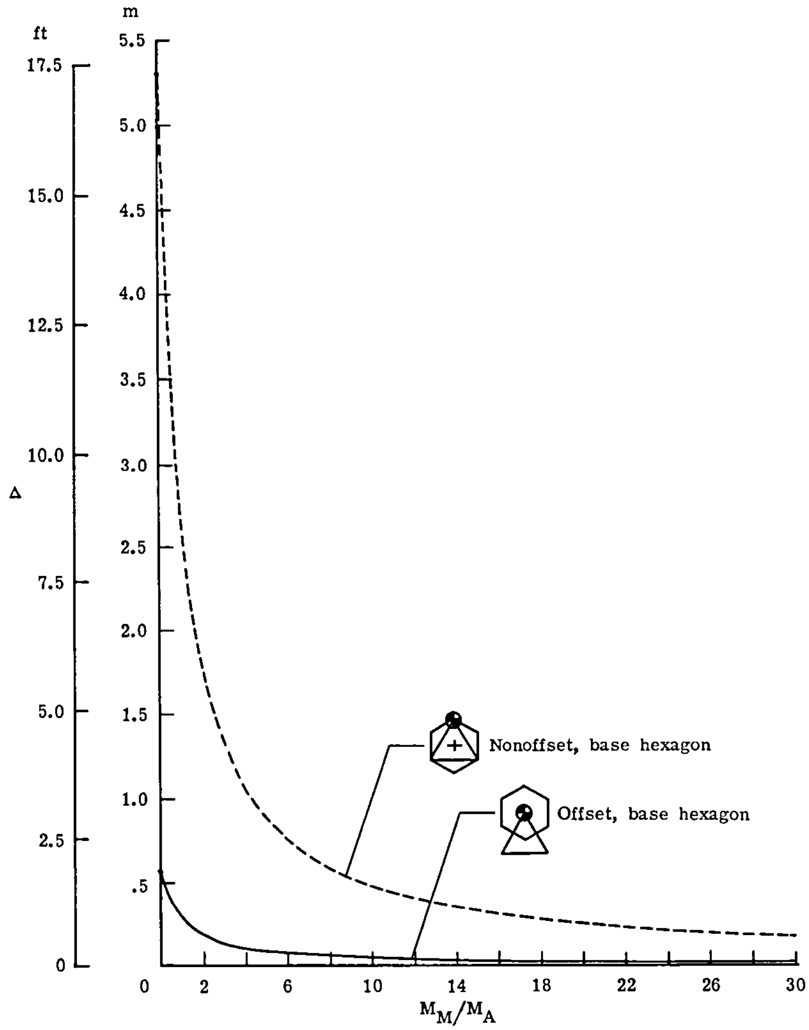


Figure 6.- Space-station center-of-gravity shift caused by rotating nonoffset solar wing arrays.

the rotary-joint axis and $r = 0.284$ m (0.933 ft) (see eq. (A8)). In the second case, (nonoffset, base hexagon) the support truss center of stiffness is aligned with the rotary-joint axis and $r = 2.64$ m (8.68 ft). The value of r in the second case is 9.3 times the value of r in the first case. The same is true of the values of Δ for fixed values of M_M/M_A. If the c.g. of the array is aligned exactly with the rotary-joint axes, then Δ is zero for all values of M_M/M_A.

The two wings making up the final space-station configuration each have 12 bays and a mass of 3345 kg (7374 lbm). The total central-module mass is estimated as 100 000 kg (220 500 lbm), which gives a value of 15 for M_M/M_A. When the support truss center of stiffness is aligned with the rotary-joint axis, $\Delta = 0.329$ m (1.08 ft) for this value of M_M/M_A. When the top longeron is aligned with the rotary-joint axis, the corresponding value of Δ is only 0.0357 m (0.117 ft). The effects of large c.g. shifts on such things as the space-station control system, onboard experiments, and flight attitude are not yet known. However, proper design can locate the c.g. of the rotating wing exactly at the rotary-joint axis and eliminate any space-station c.g. shift.

Variation of Transition-Truss Length

Figure 7 is a plot of wing-array frequency as a function of the transition-truss length for the first two cantilever modes (bending parallel and perpendicular to the

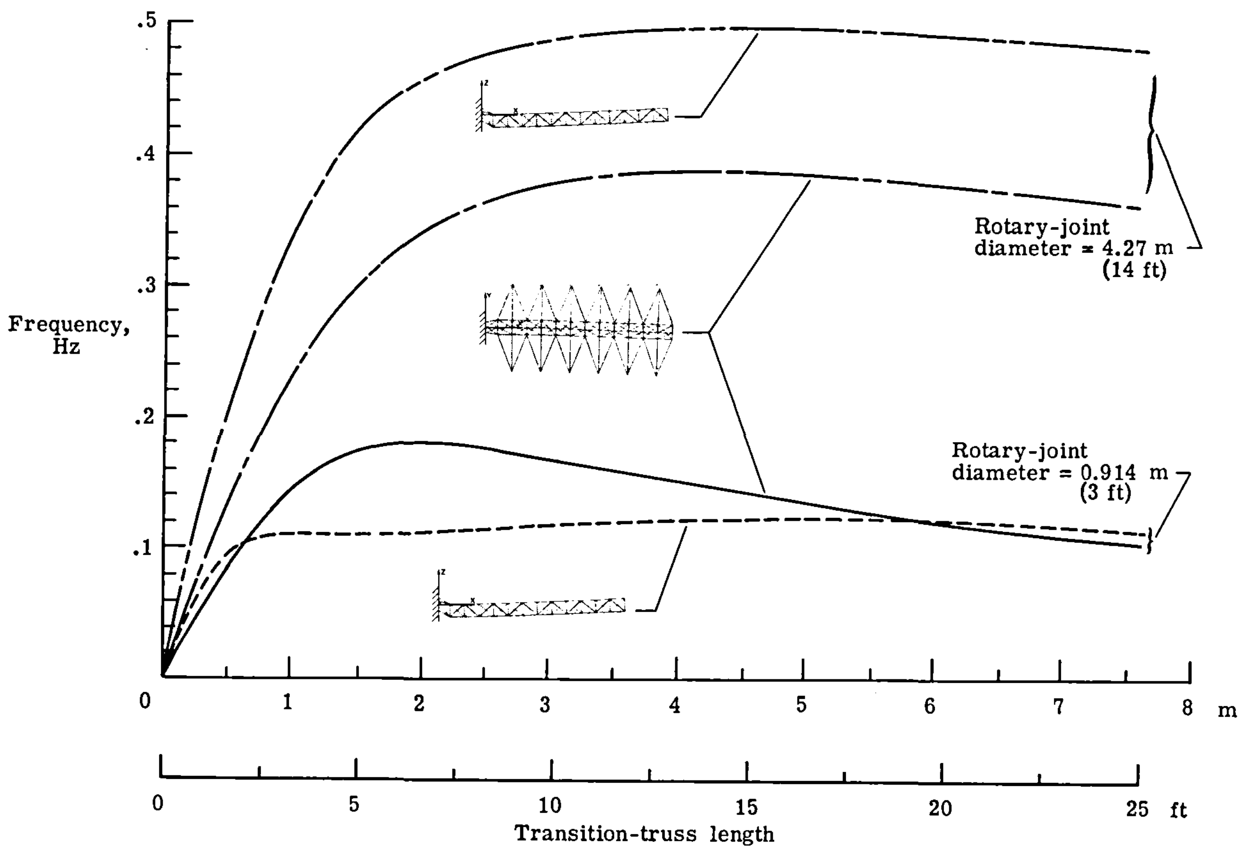


Figure 7.- Effect of transition-truss length on first two wing frequencies.

plane of the solar array). Results are presented for two rotary-joint diameters, 4.27 m (14 ft), which is the maximum module diameter, and 0.914 m (3 ft). The areas of the transition-truss members are 2.36 cm^2 (0.3657 in^2). For both diameters, the first two frequencies initially show a rapid rise starting from zero. This behavior can be explained by referring to figure 4(c), where the transition-truss length is illustrated. As the transition truss is shortened from the length shown in figure 4(c), the support truss moves in toward the wall. As the support truss nears the wall, the transition-truss members approach an orientation perpendicular to the truss, a position where their axial stiffness provides little bending stiffness to the array. Thus, the array frequencies begin at 0° for zero transition-truss length and rapidly increase with transition-truss length as the stiffness of the transition truss comes into play.

For the 0.914-m (3-ft) diameter rotary joint, the fundamental mode is bending about the Z-axis for transition-truss lengths up to approximately 0.61 m (2 ft). Between 0.61 m (2 ft) and 5.79 m (19 ft), the fundamental mode is bending about the Y-axis, and for transition-truss lengths greater than 5.79 m (19 ft), the fundamental mode is once again bending about the Z-axis. The figure shows that in the case of a 0.914-m (3-ft) diameter rotary joint, a transition-truss length between 0.61 m (2 ft) and 7.62 m (25 ft) gives a nearly constant fundamental frequency of 0.1 Hz for the solar wing array. In this case, since the fundamental frequency is nearly constant over a wide range, the transition-truss length may be chosen to maximize the second array frequency. This choice would result in a transition-truss length of 2.03 m (6.67 ft).

For the 4.27-m (14-ft) diameter rotary joint, the fundamental mode is always bending of the array about its Z-axis. The associated fundamental frequency reaches a fairly flat peak between 0.383 Hz and 0.387 Hz for transition-truss lengths ranging from 3.56 m (11.67 ft) to 5.59 m (18.33 ft). The second array frequency, corresponding to bending about the Y-axis, reaches a fairly flat peak at 0.480 Hz in approximately the same region as the fundamental frequency. This permits some flexibility in choosing the length of the transition truss without compromising the array fundamental-frequency performance.

Variation of Rotary-Joint Diameter

In figure 8, the effect on the first two wing frequencies of varying the rotary-joint diameter is presented. The transition-truss member area is again 2.36 cm^2 (0.3657 in^2), and the transition-truss length is now, on the basis of figure 7, chosen to be 3.93 m (12.9 ft). The fundamental mode is bending of the array about its Y-axis for rotary-joint diameters less than 2.03 m (6.67 ft) and bending about the Z-axis for larger rotary-joint diameters.

Both bending frequencies initially increase almost linearly with increasing rotary-joint diameter. This is consistent with the fact that the array frequencies depend strongly on the geometry of the transition truss at the root of the cantilever. The frequency of a cantilever is proportional to the square root of the truss moment of inertia which, in the case of the solar wing array, is proportional to the square of the truss depth at the root (rotary-joint diameter). Thus, the cantilever frequency should initially be proportional to the rotary-joint diameter. Figure 8 shows that the frequency associated with Y-bending is linear up to a rotary-joint diameter of approximately 2.13 m (7 ft). Also, the frequency associated with Z-bending is linear up to approximately 0.914 m (3 ft). As the rotary-joint diameter

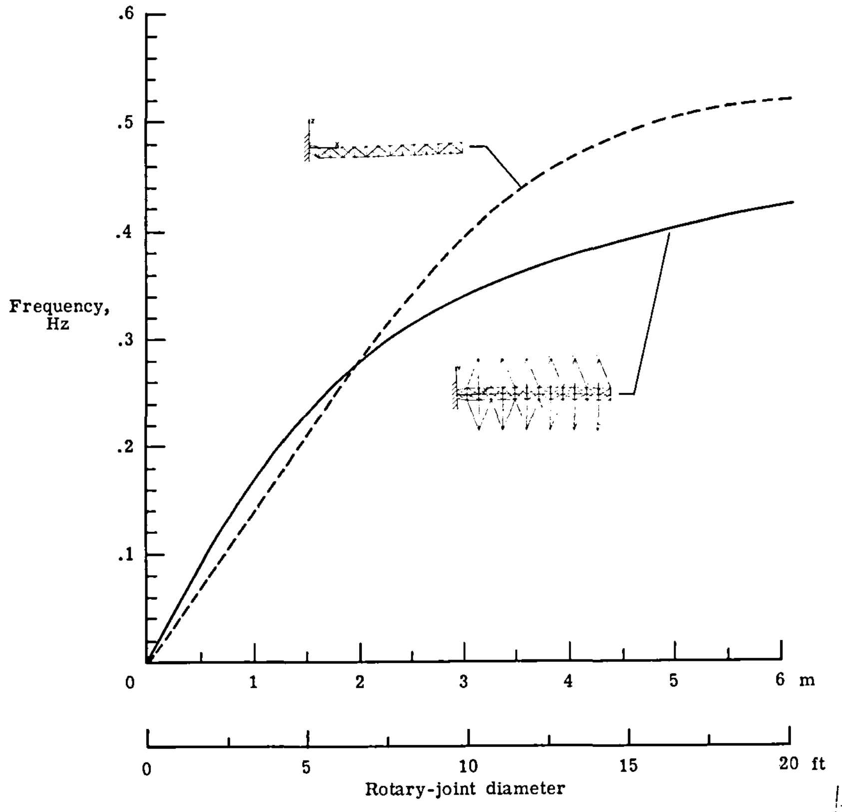


Figure 8.- Effect of rotary-joint diameter on first two wing frequencies.

is increased beyond the range shown in figure 8, the angle effects discussed previously will cause the transition-truss bending stiffness and, hence, the bending frequencies, to decrease. For the range of diameters studied, however, the general tendency is for the bending frequencies to increase with rotary-joint diameter. A rotary-joint diameter of 4.27 m (14 ft) would probably serve as a practical upper limit, since that is the maximum diameter of a space-station module. Decreasing the rotary-joint diameter from 4.27 m (14 ft) to 0.914 m (3 ft) would cause a 60-percent decrease in frequency for the Z-bending mode and a 75-percent decrease in frequency for the Y-bending mode.

Variation of Transition-Truss Strut Stiffness

The previous two studies have shown that the array frequencies are highly dependent on the stiffness of the transition truss located at the root of the cantilever array. In these previous studies, the transition-truss struts had the same axial stiffness (modulus times area) as the support truss struts. Because of the importance of the root stiffness, it is possible to increase the array frequencies by stiffening only the members in the transition truss. Figure 9 shows the effects on the solar-array frequencies of varying the transition-truss strut stiffness. The first two frequencies, corresponding to Y- and Z-bending, are shown for rotary-joint diameters of 0.914 m (3 ft) and 4.27 m (14 ft). Again, the transition-truss length is taken as 3.93 m (12.9 ft). For the solar array with a 0.914-m (3-ft) diameter rotary joint, the fundamental mode is bending about the Y-axis, and the second mode is bending about the Z-axis. For the array with a 4.27-m (14-ft) diameter rotary joint, the two modes are reversed.

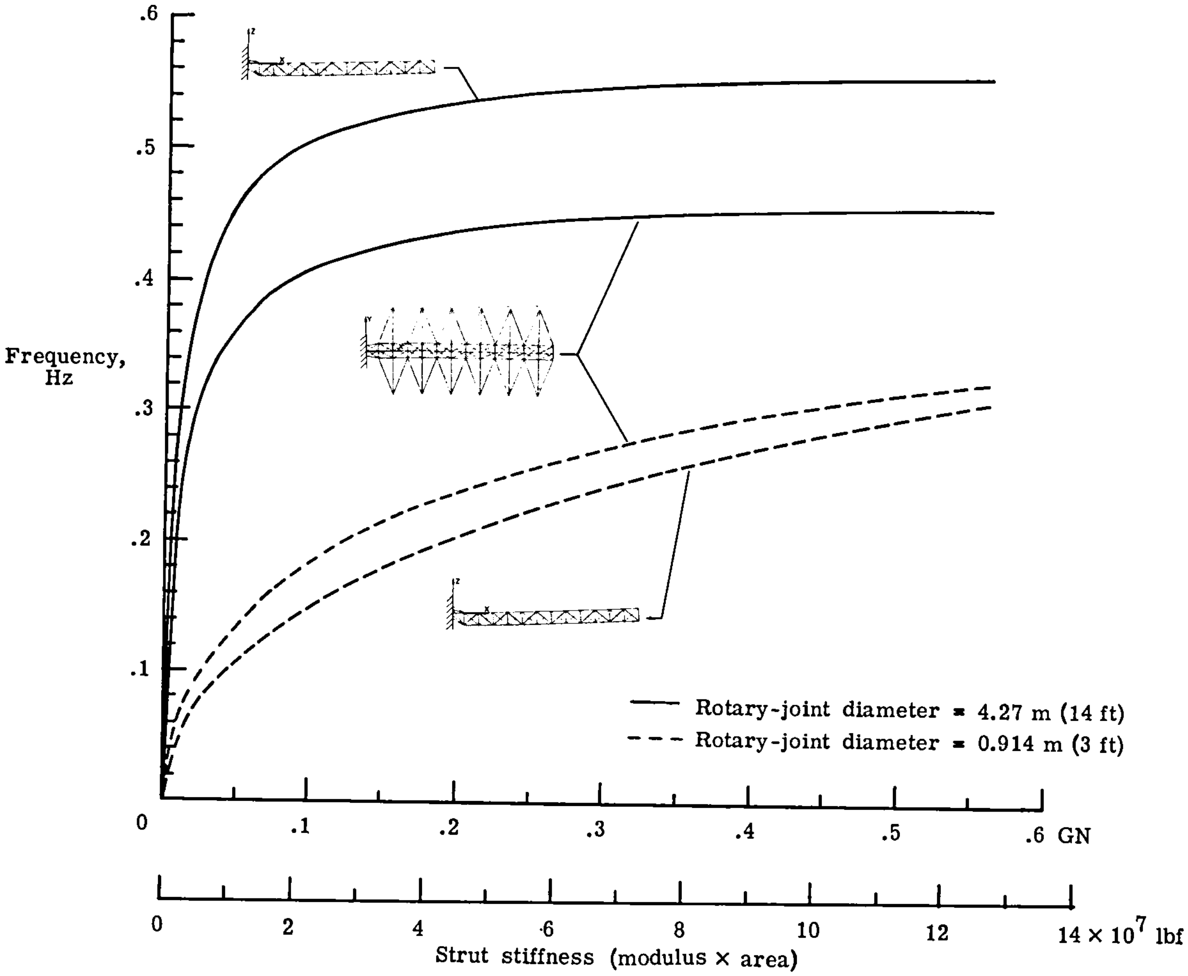


Figure 9.- Variation of wing frequencies with transition-truss strut stiffness.

For the array with a 4.27-m (14-ft) rotary joint, the two bending frequencies rise rapidly with increasing strut stiffness to a stiffness of approximately 89 MN (2.0×10^7 lbf), at which point the fundamental frequency is approximately 0.4 Hz and the second frequency is 0.5 Hz. The frequencies continue to increase from that point with transition-truss strut stiffness, but only gradually. Replacing the hollow 0.051-m (2-in.) diameter tubes described previously (EA = 65 MN (1.46×10^7 lbf)) with solid rods of the same diameter (EA = 560 MN (1.26×10^8 lbf)) results in an 18.4-percent increase in f_1 and a 16.0-percent increase in f_2, to final values of 0.457 Hz and 0.558 Hz, respectively.

For the array with a 0.914-m (3-ft) diameter rotary joint, the frequency rises gradually with transition-truss strut stiffness over the entire range studied. In this case, there is a significant increase in the first two frequencies when the hollow tubes are replaced by solid rods. The fundamental frequency rises from 0.122 Hz to 0.309 Hz, a 153-percent increase, and the second frequency rises from 0.153 Hz to 0.323 Hz, a 111-percent increase. Although replacing the hollow tubes with solid rods increases the transition-truss mass by a factor of 8.6, the resulting increase in the total wing-array mass is only 4 percent for both rotary-joint diameters studied. The increase in total space-station mass (assuming a central-modules mass of 100 000 kg (220 500 lbm)) is only 0.3 percent.

Point Design of 75-kW Wing

In table II, the modes and frequencies are summarized for the final solar-wing-array point design. The following values of the three parameters discussed previously were chosen: transition-truss length is 3.93 m (12.9 ft), rotary-joint diameter is 4.27 m (14 ft), and transition-truss strut stiffness is 65 MN (1.46×10^7 lbf). Since the transition-truss strut stiffness is the same as that of the support truss struts, only one type (cross-section) of strut would have to be manufactured. The array fundamental frequency of 0.386 Hz is slightly below the target value of 0.4 Hz. This frequency could be increased by increasing the transition-truss strut axial stiffness, which would require the manufacture of a second type of strut. In this study, further investigation was not pursued since the frequency is so near the required value.

TABLE II.- OFFSET WING-ARRAY FREQUENCIES

[Transition section length = 3.93 m (12.9 ft); Rotary-joint diameter = 4.27 m (14 ft)]

Mode	Frequency, Hz	Mode Description
1	0.386	First bending about Z-axis
2	.481	First bending about Y-axis
3	.561	First torsion about X-axis
4	1.50	Second torsion about X-axis
5	2.16	Second bending about Z-axis
6	2.41	Third torsion about X-axis
7	2.58	Second bending about Y-axis
8	3.17	Fourth torsion about X-axis
9	3.74	Fifth torsion about X-axis
10	4.09	Sixth torsion about X-axis

Wing Growth Effects

As the space-station power requirements grow, the solar arrays also have to grow. As shown in figure 10, increasing the array length lowers the array frequencies. For example, a 75-kW space station with six bays in each solar wing array would have a fundamental frequency (bending about the Z-axis) of 0.938 Hz and second and third frequencies (torsion about the X-axis and bending about the Y-axis, respectively) of 1.26 Hz and 1.54 Hz. Expanding to a 150-kW 12-bay configuration would decrease the Z-bending frequency by 59 percent to 0.386 Hz, the X-torsion frequency by 55 percent to 0.561 Hz, and the Y-bending frequency by 69 percent to 0.481 Hz.

Free-Free Space-Station Example

Figure 2 shows a 150-kW design incorporating the solar-wing-array concept shown in figure 3 and a central cluster of modules. In this concept, the two arrays are shown attached to rotary joints at each end of one of the modules. Also, the top longeron of the array is aligned with the rotary-joint axis.

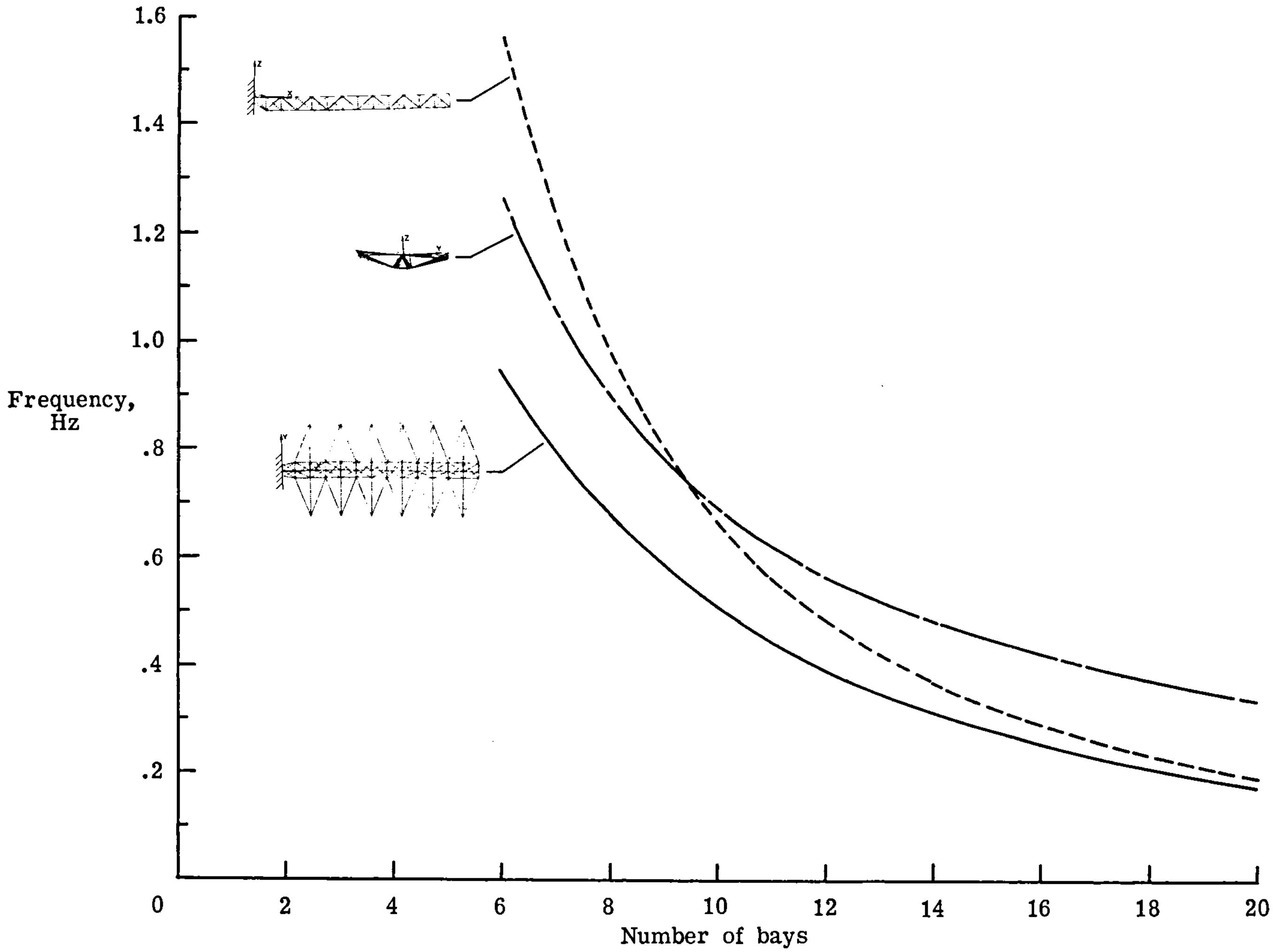
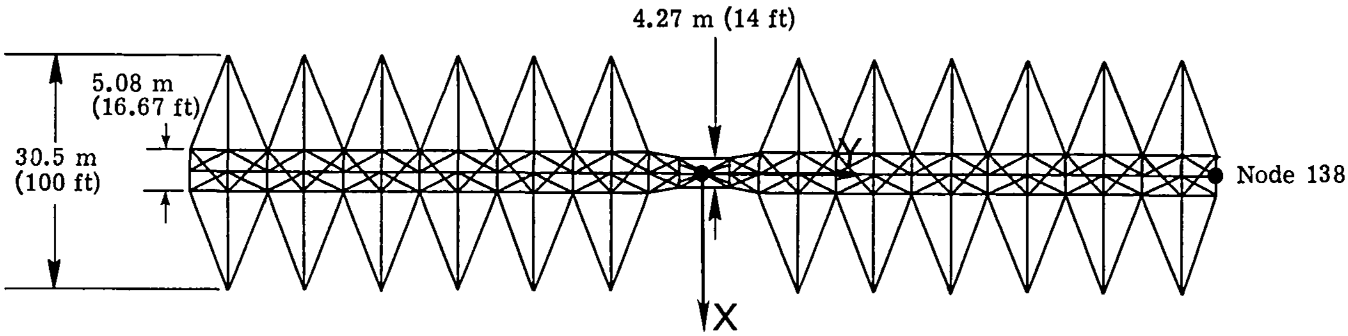


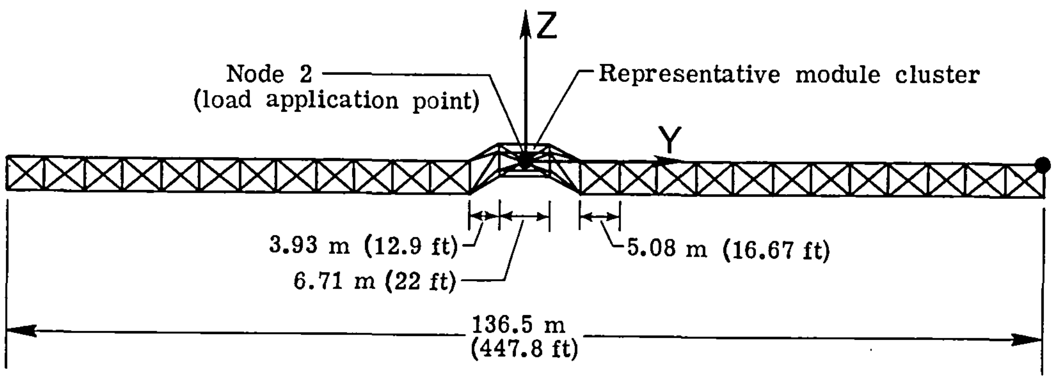
Figure 10.- Variation of wing frequencies with number of bays in wing. Bay length = 5.08 m (16.67 ft).

Modal analysis.- The finite-element model used in the analysis is shown in figure 11. The two solar arrays are modeled as in figure 4 with a transition-truss length of 3.93 m (12.9 ft), a transition-truss member stiffness of 65 MN (1.46×10^7 lbf), and a rotary-joint diameter of 4.27 m (14 ft). The module cluster is not modeled in detail. Instead, the module which connects the two wing arrays is modeled as a box-like truss with rigid beam and truss elements. The module is 4.27 m (14 ft) in diameter and 6.71 m (22 ft) in length. A lumped mass is located at the origin and assigned the weight and moments of inertia of the module cluster. The size of the space station (137 m (448 ft) long and 30.5 m (100 ft) wide) is due to the extremely large solar arrays; hence, major emphasis was placed on their design.

The frequencies and mode shapes of the free-free space station are calculated using a finite-element eigenvalue routine. The first six modes of the space station are zero-frequency rigid-body modes. The first elastic mode of the structure is first symmetric bending in the plane of the solar arrays. (See fig. 12(a).) The second elastic mode is first symmetric bending out of the plane of the solar arrays. (See fig. 12(b).) Table III is a summary of the first 19 frequencies and mode shapes

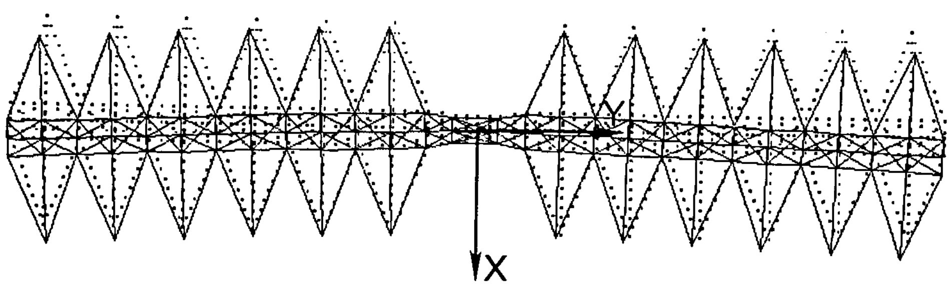


(a) X-dimensions.

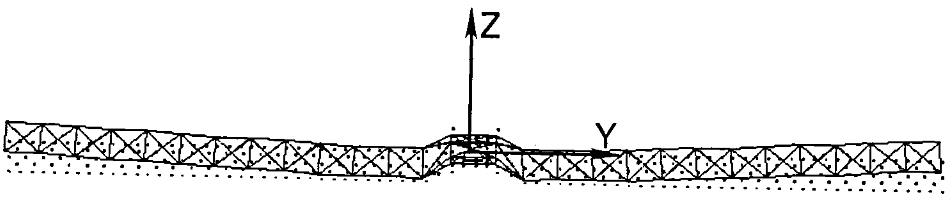


(b) Y-dimensions.

Figure 11.- Space-station finite-element model.



(a) First elastic mode - bending about Z-axis.



(b) Second elastic mode - bending about X-axis.

Figure 12.- Space-station mode shapes.

TABLE III.- FREE-FREE SPACE-STATION FREQUENCIES

[Central-module mass = 90 718 kg (200 000 lbm);
Transition section length = 3.93 m (12.9 ft);
Rotary-joint diameter = 4.27 m (14 ft)]

Mode	Frequency, Hz	Mode description
1 to 6	0.000	Rigid body
7	.403	First symmetric Z-bending
8	.495	First symmetric X-bending
9	.536	First X-torsion
10	.818	Second X-torsion
11	1.36	First antisymmetric Z-bending
12	1.56	Third X-torsion
13	1.60	Fourth X-torsion
14	1.63	First antisymmetric X-bending
15	2.23	Second symmetric Z-bending
16	2.42	Fifth X-torsion
17	2.50	Sixth X-torsion
18	2.62	Second symmetric X-bending
19	2.83	Second antisymmetric Z-bending

for the space station, assuming a module cluster mass of 90 700 kg (200 000 lbm). The total space-station mass, module cluster plus two wing arrays, is 97 000 kg (215 000 lbm). The first symmetric bending frequency of the free-free space station, 0.403 Hz, is 4.4 percent greater than the corresponding cantilevered array frequency and now meets the initially prescribed space-station frequency requirement.

Variation of central-module mass.- Growth of the space station includes the addition of modules, with their associated masses and inertias, as well as the addition of bays to the wing arrays. In figure 13, the variation of the first two space-station elastic frequencies with module cluster mass is shown. The space-station model used here is the same as that shown in figure 11 (i.e., one module and two 12-bay wings). When the central-module cluster mass is zero, the fundamental elastic frequency (associated with first symmetric bending in the array plane) is 0.651 Hz, the second elastic frequency (associated with first symmetric bending out of the array plane) is 0.835 Hz, and the space station behaves as a free-free beam. Both frequencies drop rapidly as the module cluster mass is increased from zero to 45 400 kg (100 000 lbm). As the module cluster mass is further increased, the frequencies remain essentially constant, approaching values of 0.397 Hz for the fundamental frequency and 0.488 Hz for the second frequency. A large central-module cluster mass effectively clamps the two solar arrays in the middle so that they behave as cantilever beams.

Frequencies for corresponding wing-array cantilever modes are shown by the two horizontal lines in figure 13. For a space station with a large central mass, the space-station frequencies can be predicted very accurately using only a cantilever model of the wing array. For the module cluster mass assumed in this study, 90 700 kg (200 000 lbm), the first elastic free-free frequency is 4.4 percent larger than the corresponding cantilever frequency, and the second elastic free-free frequency is 3.0 percent larger than the corresponding cantilever frequency.

Transient response analysis.- The space-station transient response studies were carried out on the full free-free space-station model shown in figure 11. Modal superposition, using the first 19 modes and frequencies (table III), was used to calculate the undamped response. The disturbance was assumed to be a step load applied at the origin (space-station center) in the positive X-direction. In the space-station model, a rigid mass having the mass and moments of inertia of the central-module cluster is located at the space-station center.

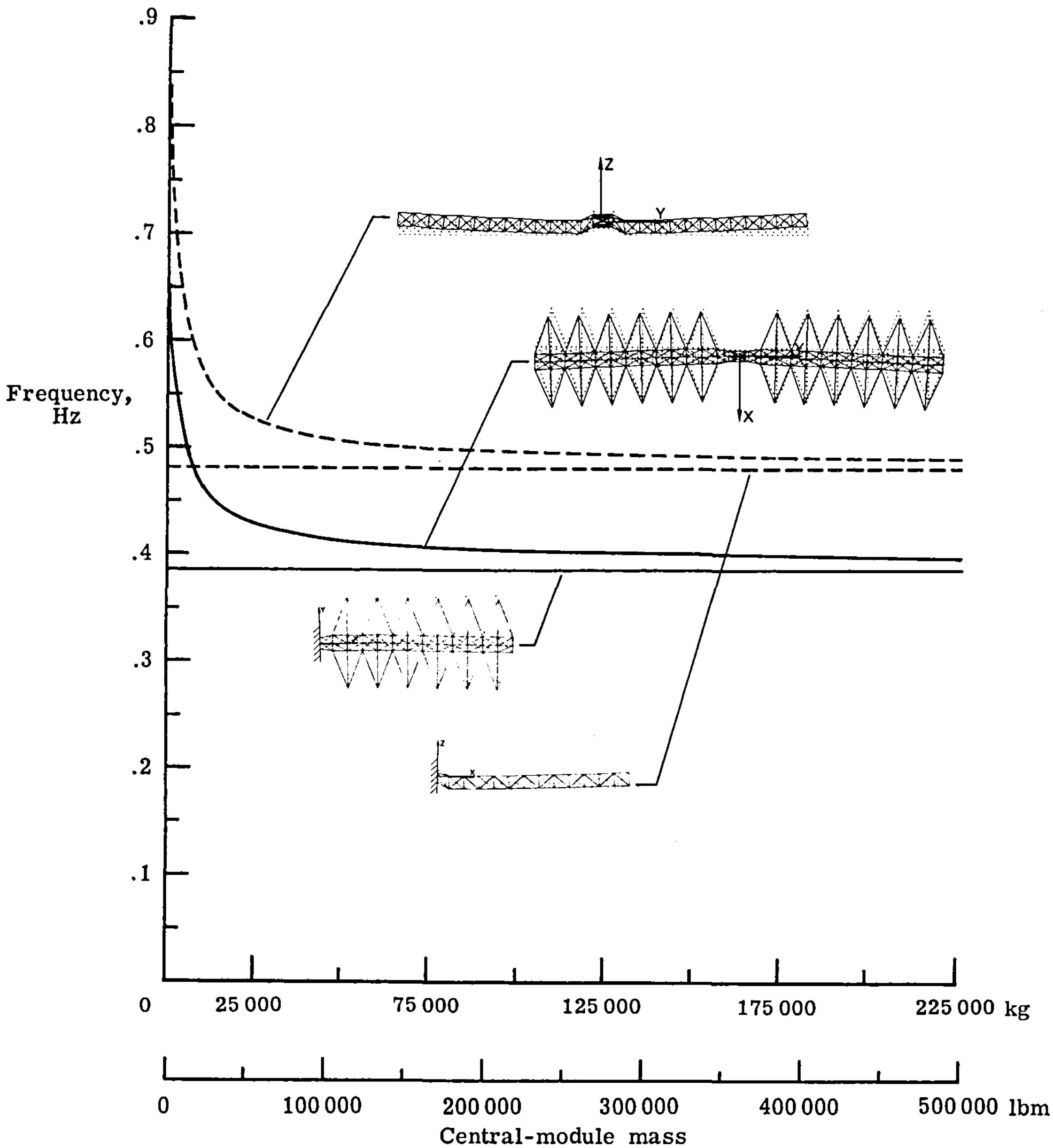


Figure 13.- Variation of first two free-free space-station frequencies with central-module mass. M_A = 6700 kg (14 700 lbm).

Figure 14 shows the acceleration response of the space-station center (node 2 in fig. 11) to a step load of 4.5 N (1.0 lbf) applied for a duration of 1.5 seconds.

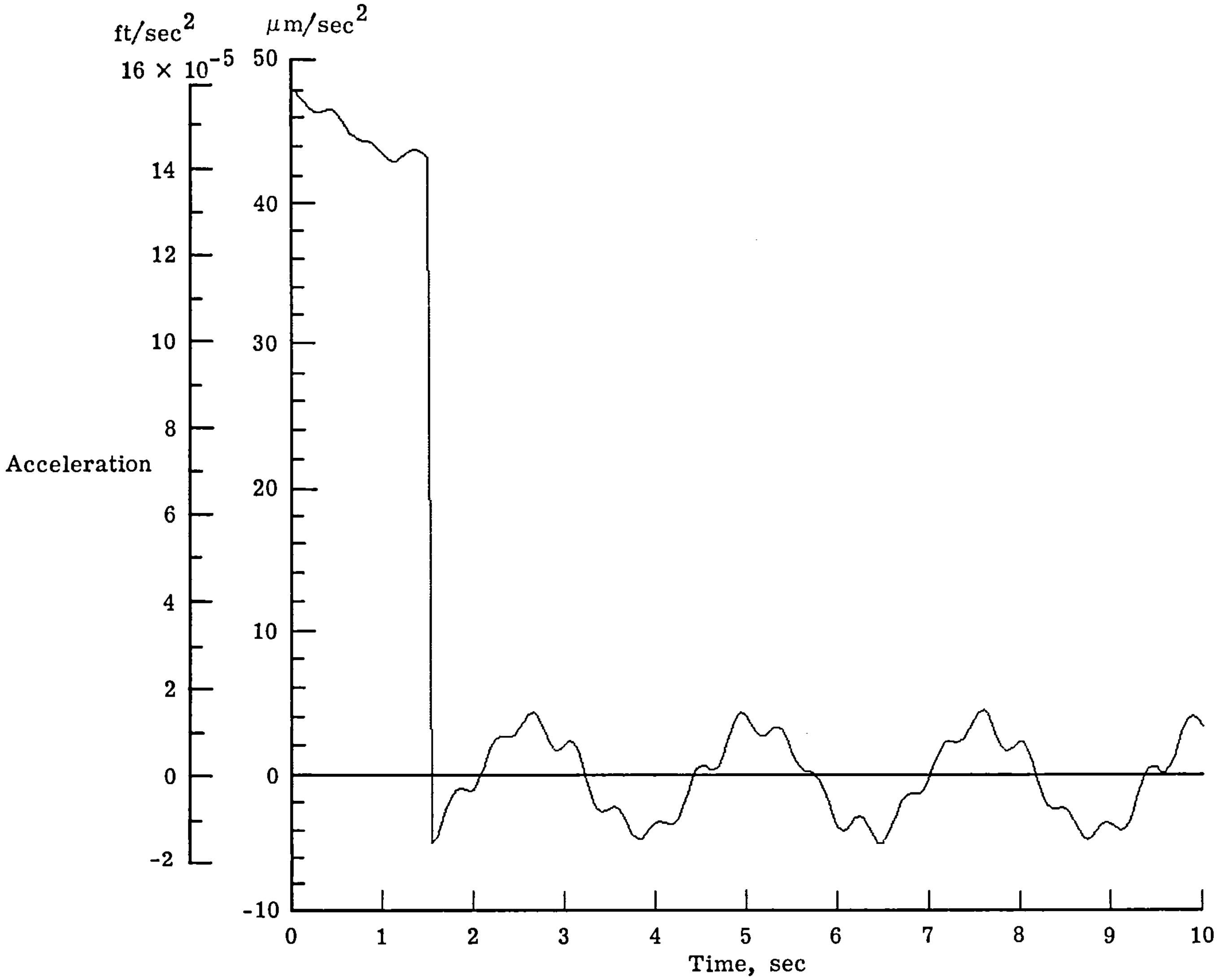


Figure 14.- Acceleration response in X-direction of space-station center (node 2) to a 1.5-second applied step load.

The acceleration of the space-station center consists of a rigid-body component and an elastic component during the time the force is applied. The instantaneous acceleration of the space-station center, found by dividing the applied force by the modules cluster mass, is 49.0 μm/sec^2 (1.93 × 10^{-3} in/sec^2). This compares closely with the computed acceleration at time zero (48.3 μm/sec^2 (1.90 × 10^{-3} in/sec^2)). For times greater than 1.5 seconds, the rigid-body acceleration is gone, leaving only the elastic component, and the acceleration at the space-station center ranges between ±4.60 μm/sec^2 (1.80 × 10^{-4} in/sec^2). Measurements taken from figure 14 show the main response of the space-station center to be at 0.40 Hz, with a much smaller response at 2.23 Hz superimposed. Table III shows the modes associated with these frequencies to be first and second symmetric bending about the Z-axis. The acceleration at the array tip (node 138 in fig. 11) is shown in figure 15 for the same step-load duration. Once again, the acceleration is composed primarily of the mode at 0.40 Hz with a small component of the mode at 2.23 Hz superimposed. The acceleration at the tip varies between ±164 μm/sec^2 (6.45 × 10^{-3} in/sec^2).

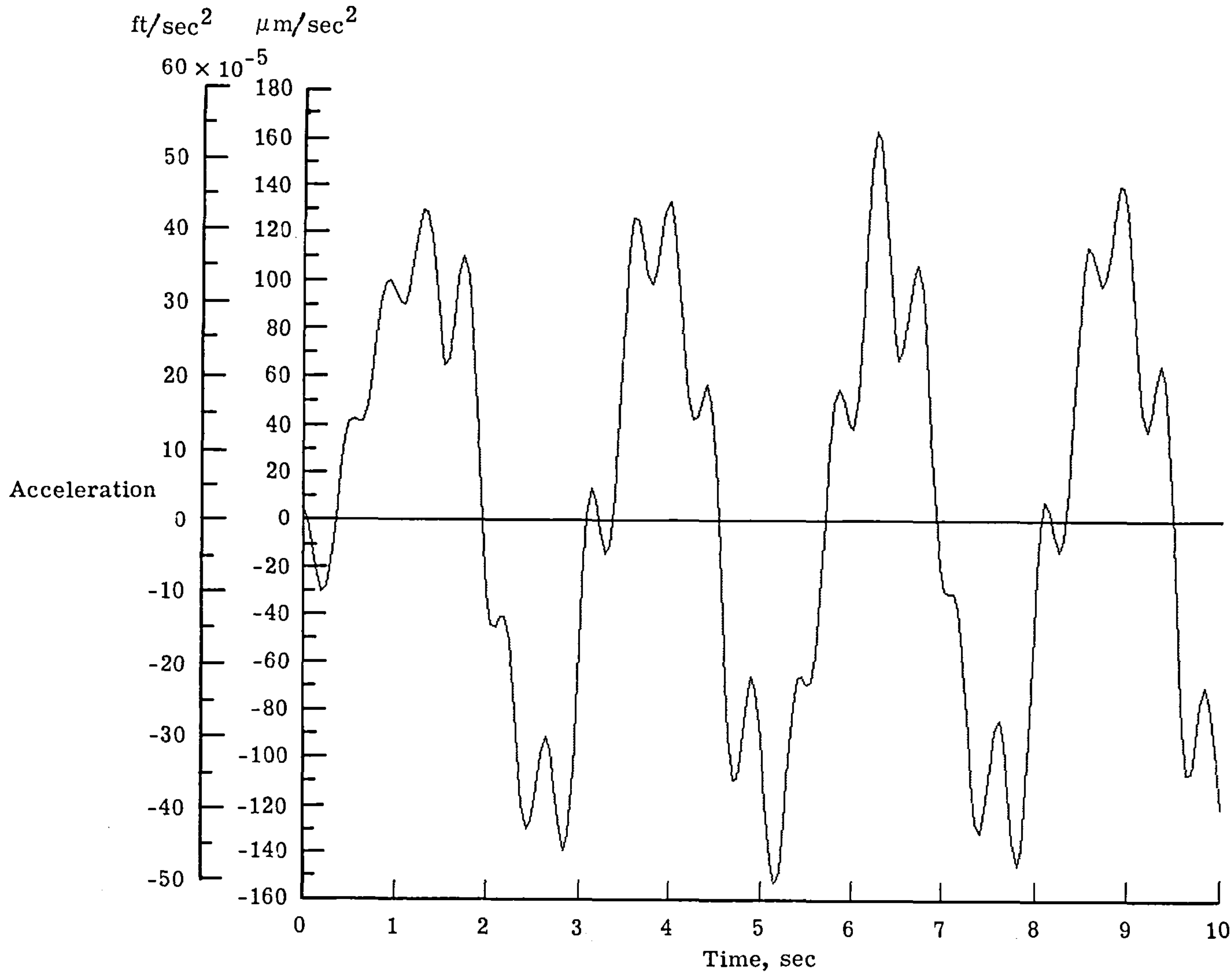


Figure 15.- Acceleration response in X-direction of array tip (node 138) to a 1.5-second applied step load.

One important result which must come out of a successful design effort is an indication of the forces which can be safely applied to a structure. One such set of results was obtained for this space-station concept in the following manner. First, the step force of magnitude 4.5 N (1 lbf) was applied to the space-station center in the positive X-direction for force durations ranging from 0.125 seconds to 3.0 seconds. A transient response analysis was then performed for each load case, during which the maximum and minimum forces P in each strut of the solar array were calculated. The critical member in the wing array was defined as the most highly loaded relative to its Euler buckling load P_{cr} for the given applied force and step duration. Finally, the allowable force of a given duration was calculated to be that which would be required to buckle the critical member. As an example, the case of a step load of 0.125-second duration is used. The applied force of 4.5 N (1.0 lbf) caused the critical member in the solar array (a batten) to reach a value of $P/P_{cr} = 4.35 \times 10^{-5}$. Thus, an applied force of 4.5 N (1.0 lbf)/4.35×10^{-5} = 102.3 kN (23 000 lbf) would be the allowable force magnitude for this step duration.

In figure 16, the magnitude of the allowable applied force as a function of the step duration is summarized. The allowable force magnitude drops off sharply as the step duration is increased from 0 to 0.5 second. As was shown by the acceleration response, this step load causes the most excitation of the first space-station elastic mode, which has a period of 2.5 seconds. Thus, a step load having a duration of half the first period, 1.25 seconds, is in phase with that mode and would be expected to cause the greatest loads in the array members. Hence, a step load of this duration would have the smallest allowable force magnitude, as is shown by figure 16. For longer step durations, the allowable force magnitude remains constant at 19 274 N (4333 lbf). Allowable force charts such as figure 16 could be obtained for a variety of applied loading situations and would be critical in determining the docking loads, thruster loads, etc., which could safely be applied to the space station.

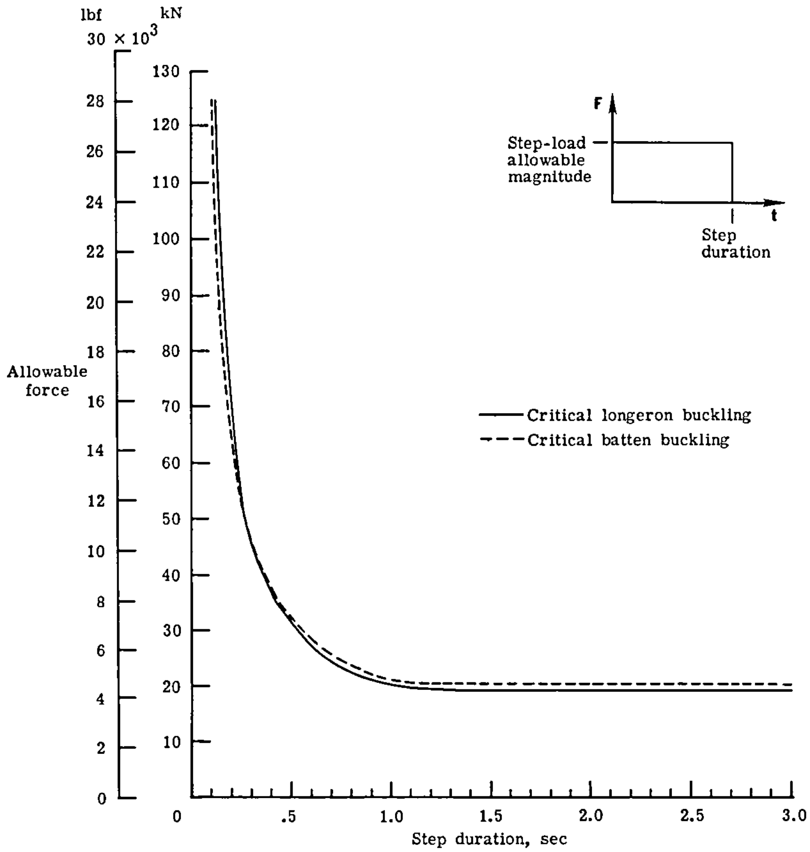


Figure 16.- Allowable force to be applied at space-station center in X-direction.

CONCLUDING REMARKS

The purposes of this investigation were to present a solar-wing-array design concept which meets space-station requirements on fundamental frequency, modularity, and growth potential and to study changes in the array vibration characteristics caused by variations in selected design parameters. A wing-array concept consisting primarily of a large, stiff, erectable backbone truss and deployable solar blankets was designed, and a finite-element model was developed. A parametric study was conducted in which modes and frequencies were calculated for a large number of different combinations of design parameters. Based on the parametric studies, a specific array point design was chosen and incorporated into a full space-station design. The transient response of the space station to a step load applied at its center was also studied. Results of all the studies lead to the conclusions which follow.

One of the major requirements placed on the large space-station solar arrays is that they be able to rotate in order to follow the Sun. As a result, the wing arrays are mounted to rotary joints on the ends of one of the space-station modules. This study showed that to avoid large space-station c.g. shifts during rotation of the solar arrays, the center of gravity of the arrays should be aligned with the rotary-joint axis.

For the wing-array design in this study, a transition truss connects the three-longeron solar-array support truss to the rotary joint. This transition truss should be long enough to minimize strut-angle effects on the wing-array frequencies. Also, since the transition truss is at the root of the cantilevered array, increasing the stiffness of only the transition-truss struts can substantially increase total-array frequencies. This procedure is an effective means of increasing wing-array frequencies, because large frequency changes are possible for small weight changes.

The diameter of the rotary joint to which the wing array was attached was another design parameter that was studied. In order to maximize array frequencies, the rotary-joint diameter should be made as large as practical. In this study, the practical limit on rotary-joint diameter was determined by the space-station-module diameter.

The space station is likely to evolve from its initial configuration through the addition of more modules and wing area. The space-station structural frequencies will change considerably, as, for example, more bays are added to the wing arrays. For the 150-kW concept studied here, the space-station frequencies initially decrease as modules are added to the central cluster. When the module cluster mass reaches seven times the mass of the two arrays, however, space-station frequencies remain nearly constant as more modules are added. The large changes in frequencies associated with space-station growth are likely to significantly influence the control-system design.

Finally, the transient response of the space station to a step load applied at its center was investigated. For this step load, accurate results were obtained using the first 13 elastic modes of the structure. The results suggest that allowable loads on the space station (due to thruster firings, shuttle docking, crew motion, etc.) may be dictated by buckling loads in critical wing-array struts.

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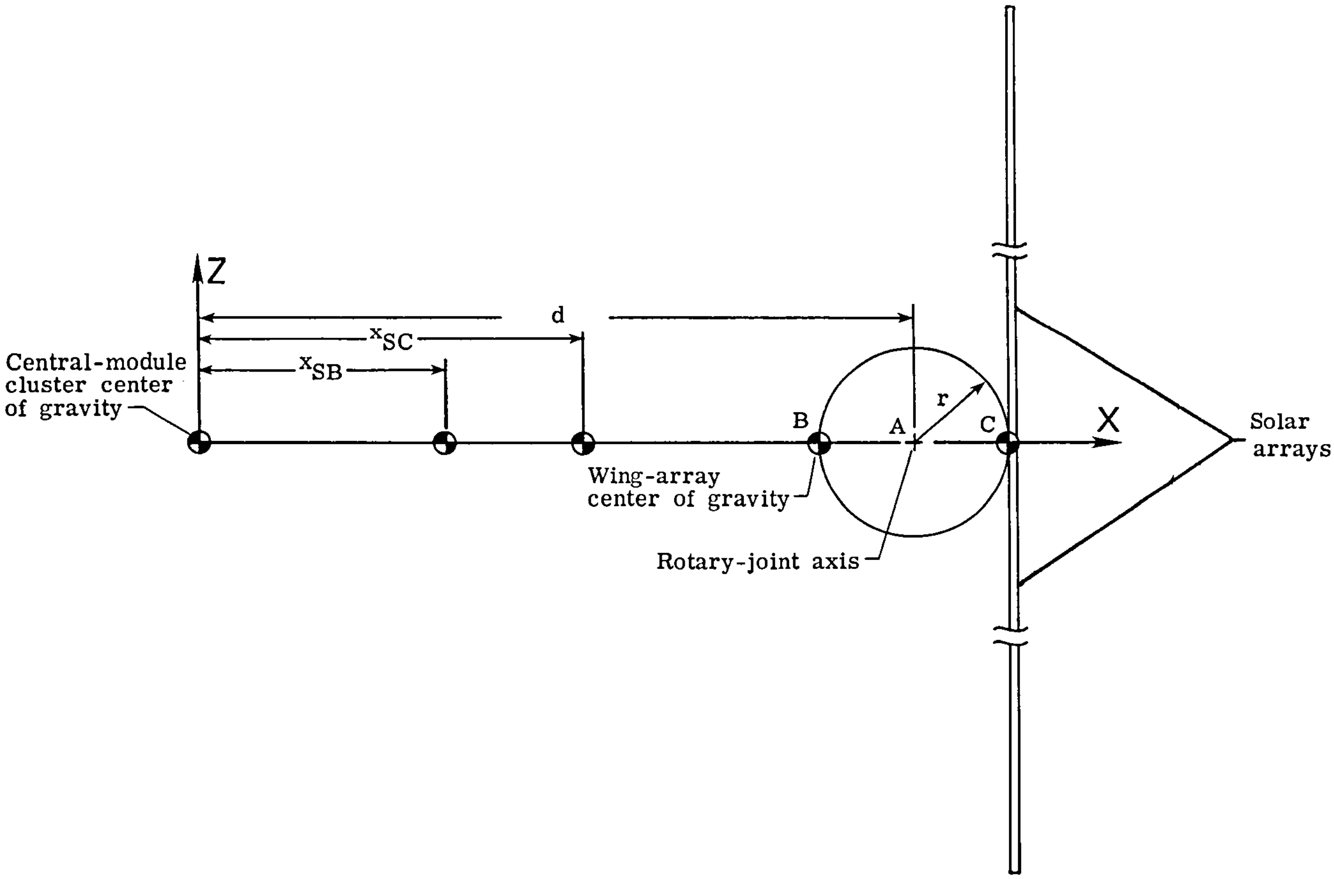
May 14, 1984

APPENDIX

SPACE-STATION CENTER-OF-GRAVITY SHIFT CAUSED BY SOLAR-ARRAY ROTATION

A major requirement for the space station is that the large solar-wing arrays be allowed to rotate independently of the central-module cluster. In this appendix, the influence of array rotation on space-station center-of-gravity shifts is investigated.

Sketch A shows a simple model which can be used for locating the center of gravity (c.g.) of a space station of the type shown in figure 1. This model corresponds to a view of the X-Z plane of the space station at $Y = 0$. (See fig. 1.) The two major components are the central-module cluster and the solar arrays. The c.g. of the central-module cluster is assumed to be at the origin, point O. The rotary-joint axis is located a distance d from the origin at point A. The wing array consists primarily of a support truss with attached solar blankets. Because the blankets are much more massive than the support truss, the c.g. of the arrays lies close to the plane of the blankets and is thus offset from the array center of rotation. In the model, the amount of this offset is r. As the arrays are rotated through 360°, the array c.g. travels in a circle of radius r about the rotary-joint axis.



APPENDIX

Since the c.g. of the space station is determined by the locations of both the module cluster c.g. and the array c.g., the total space-station c.g. moves as the arrays are rotated. When the arrays are rotated so that their c.g. lies on the X-axis at point B, the array c.g. is closest to the module cluster c.g., and the space-station c.g. is located a distance x_{SB} from the origin. When the arrays are rotated so that their c.g. lies on the X-axis at point C, the array c.g. is farthest from the module cluster c.g., and the space-station c.g. is located at a distance x_{SC} from the origin. The total change in the space-station c.g. along the X-axis Δ during a rotation of the solar arrays is $x_{SC} - x_{SB}$. Before calculating Δ, define the space-station mass M_S to be the sum of the module cluster mass M_M and the array mass M_A as follows:

$$M_S = M_M + M_A \tag{A1}$$

The X-coordinate of the space-station c.g x_S is given by

$$x_S = \frac{(M_M x_M) + (M_A x_A)}{M_S} \tag{A2}$$

Since the c.g. location of the modules is always defined as being at the origin,

$$x_M = 0 \tag{A3}$$

Thus, equation (A2) becomes simply

$$x_S = \frac{M_A x_A}{M_S} \tag{A4}$$

When the array is rotated to position B,

$$x_{SB} = \frac{M_A(d - r)}{M_S} \tag{A5}$$

and when it is rotated to position C,

$$x_{SC} = \frac{M_A(d + r)}{M_S} \tag{A6}$$

The total change in space-station c.g. is

$$\Delta = x_{SC} - x_{SB} = 2r \frac{M_A}{M_S} \tag{A7}$$

Recasting equation (A7) into the form

$$\Delta = \frac{2r}{1 + \left(\frac{M_M}{M_A}\right)} \tag{A8}$$

shows that, for a given value of r, Δ can be reduced by increasing the module cluster mass relative to the array mass. On the other hand, for a given space-station mass, the value of Δ can be reduced by minimizing the distance between the array c.g. location and the array center of rotation. If the array c.g. is made collinear with the rotary-joint axis, r is zero and the c.g. location of the space station does not change as the arrays are rotated, regardless of the cluster-to-array mass ratio. Since, from a controls point of view, it is desirable to keep the space-station c.g. location fixed, a proper solar-array design should minimize Δ by minimizing r.

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15. Supplementary Notes

16. Abstract

A solar-wing-array concept is described which meets space-station requirements for minimum fundamental frequency (0.4 Hz), component modularity, and growth potential. The basic wing-array design parameters are varied, and the resulting effects on the array vibration frequencies and mode shapes are assessed. The transient response of a free-free space station (incorporating a solar-wing-array point design) to a load applied at the space-station center is studied. The use of the transient response studies in identifying critically loaded structural members is briefly discussed. The final 150-kW space-station configuration has a fundamental elastic frequency of 0.403 Hz.

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